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Variable Geometry and Multicycle Engines

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GROUPE CONSULTATIF POUR LA RECHERCHE ET LE DEVELOPPEMENT AEROSPATIAL

AGARD Conference Proceedings No.205

VARIABLE GEOMETRY AND MULTICYCLE ENGINES

Papers presented at the 48th Meeting of the AGARD Propulsion and Energetics Panel
held at the Ecole Nationale Supérieure des Techniques Avancées,
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The Propulsion and Energetics Panel wishes to express its thanks to the hosts, the French National Delegates to AGARD, for the invitation to hold the meeting in Paris, France, and for the provision of the necessary facilities and personnel to make the meeting possible.

PREFACE

This symposium was concerned with the improving possibilities available from various variable geometry components both in service and under study and test now.

High performance of aero engines in different phases of the flight missions cannot be attained by conventional engines with one design point only. Change from one design point to another is feasible by using geometry variation in the different components of engines like intake, fan, compressor, combustor, turbine, afterburner, and nozzle. Moreover, change from one thermodynamic cycle to another is possible if, e.g. by-pass ratio and/or energy inputs in the different flow passages are varied.

Both military and civil aspects of variable geometry engines were analyzed, a broad variety of cycle investigations presented as well as work on geometry variation of the individual engine components. Advanced performance both for supersonic conventional take-off and landing aircraft and for VTOL aircraft and for VTOL were shown. High speed turbo-props are likely to offer improvements in fuel conserving engine design as variable turbines do. Lower exhaust emission rates are to be expected when using variable combustors. But integrated control systems only will enable full exploitation of advantages and papers presented on this subject found great interest. Discussions of individual presentations and the final Round Table Discussion placed concern on weight and complexity problems of the engines under consideration and tried to identify what risks and costs are likely to be involved. Still, favourable SFC in reheat and dry conditions at subsonic and supersonic speeds, the improvements in the matching of engine mass flow to intake capacity in all flight conditions and the improvement of engine operating stability are considered to pay off sufficiently.

It was felt that the symposium provided a stimulating forum for debating the potential benefits offered by variable cycle engines, and for reviewing the progress in variable geometry component technology which will ultimately determine the extent to which these benefits will be achievable in practice. There was left no doubt that more extensive variability than is currently exploited will be a feature of future power plants for particularly demanding applications.

PREFACE

L'objet de ce symposium était l'examen des possibilités d'amélioration offertes par les divers composants à géométrie variable, soit déjà en service, soit actuellement au stade des études et des essais.

On ne peut obtenir de performances élevées dans les différentes phases des missions de vol avec des moteurs d'avions classiques présentant un seul point d'adaptation. On peut cependant passer d'un point d'adaptation à un autre grâce au concept de la géométrie variable appliqué à divers composants du moteur tels que entrée d'air, soufflante, compresseur, chambre de combustion, turbine, post-combustion et tuyère. D'autre part, le passage d'un cycle thermodynamique à un autre est réalisable si, par exemple, on fait varier le rapport de dilution et/ou les apports d'énergie dans les différents passages de l'écoulement.

Au cours des divers exposés, les aspects à la fois militaires et civils des moteurs à géométrie variable ont été analysés, et les résultats d'études nombreuses et diverses sur les cycles et sur les variations de géométrie des composants de moteur ont été présentés. On a montré les performances élevées obtenues tant sur des avions supersoniques à décollage et atterrissage classiques que sur des ADAVs. La conception des turbo-propulseurs à grande vitesse est susceptible d'être améliorée du point de vue consommation de carburant, tout comme l'a été celle des turbines à géométrie variable. De même, on obtiendra des taux d'émission de gaz d'échappement plus faibles grâce aux chambres de combustion à géométrie variable. Mais seuls les systèmes de commande intégrés permettront d'exploiter pleinement ces avantages et les exposés traitant de cette question ont suscité un vif intérêt. Au cours des discussions qui suivirent la présentation de chaque exposé, et pendant la "Table Ronde", l'accent fut mis sur les problèmes liés au poids et à la complexité des moteurs étudiés, et sur la détermination des risques et des coûts probablement impliqués. Une combustion spécifique favorable avec ou sans réchauffe aux vitesses subsoniques et supersoniques, une meilleur adaptation du débit du moteur à la capacité des prises d'air dans toutes les conditions de vol, et l'amélioration de la stabilité de fonctionnement du moteur sont toujours considérées comme des facteurs suffisamment valables.

De l'avis de tous, ce Symposium a donné lieu à des débats stimulants sur les avantages potentiels présentés par les moteurs à cycle variable, et a permis de faire le point des progrès qui ont marqué la technologie de ce domaine, progrès dont dépend, en fin de compte, la réalisation concrète de ces avantages. Il semble hors de doute que les systèmes propulsifs de l'avenir seront caractérisés par un degré de géométrie variable plus poussé qu'à l'heure actuelle en ce qui concerne les applications aux exigences particulières.

CONTENTS

OPPORTUNITIES FOR VARIABLE GEOMETRY ENGINES IN MILITARY AIRCRAFT

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SUMMARY

Studies of new mission opportunities and design concepts for military aircraft, both in short-range and long-range air-to-ground combat aircraft, suggest that the direction for new military engine developments could change course to accommodate multi-mission vehicles which are truly supersonic in their principal role. The paper discusses several concepts for engine development and the apparent opportunities such advancements offer in terms of either vehicle performance or total system cost.

Engine opportunities are discussed for both medium-range air-to-surface attack vehicles and for long-range, supersonic vehicles with high altitude, recon/strike capability.

1.0 INTRODUCTION

For the past decade or two, military engine development in the free world has centered around fixed cycle, afterburning (A/B) turbofan engines. The emphasis has been on simplicity with few variable components. The design condition for the military missions have been predominately at high subsonic speeds. The air-to-air superiority fighters have placed maximum emphasis on persistence (tight combat turns and high rates of climb) using the engine in full A/B mode, with the transonic cruise legs performed at maximum dry power. Supersonic flight has been a fallout in full A/B power. Technological goals for these engines have been aimed at maximizing engine thrust/weight ratio; hence, the core engines have been designed with a steady increase in combustor exit temperature, use of cooled, single stage high pressure turbines where possible, and core compressors with very high pressure ratio per stage. The overall engines are configured with very low bypass ratio fans. Additionally, design ingenuity with use of advanced materials, seals and bearings have aided greatly in the achievement of thrust/weight ratios, for reasonable military life, exceeding 8:1. These engine designs also allow for relatively simple, variable area nozzles.

Interdiction aircraft have and are being developed with varying radius mission capabilities based on somewhat similar engines. Engine requirements for this type of aircraft, compared to the fighter aircraft, result in higher compressor pressure ratios, somewhat higher fan bypass ratios, and more sophisticated con-di ejector nozzles which put more emphasis on installed drag because of the more difficult cruise mission range requirements. Once again, the aircraft supersonic flight capability is a fallout. There are an abundance of aircraft in development for these two roles; hence, it is natural that attention be given to at least one new role and perhaps one neglected area. These are the all supersonic, air-to-surface attack vehicle and the very high altitude, long range, all supersonic recon/strike vehicle, respectively.

Unfortunately, there are those who feel that designers' past efforts at supersonic cruising vehicles have been, at best, only marginally successful. This misconception can readily be corrected with dedication to the supersonic mode, as one needs only to point to the Concorde to see that excellent design with outstanding range factor can, in fact, be developed. The difficulty begins when one tries to combine good supersonic cruise capability with equally good off-design performance.

In the past, the supersonic performance has been degraded in favor of transonic maneuver and low altitude penetration. Furthermore, it is not clear what mission versatility will be required to make either of these suggested vehicles viable, but pressures are certain to be placed on the engine designer to offer multimode engine capability unlike that in operation today.

The idea of considering concepts for engines other than conventional turbofans with augmentation for military aircraft was first seriously expressed in the U.S. a few years ago (Reference 1, 3 and 4) as the result of considerable effort to define efficient dry-power engines for supersonic cruise that would meet transonic acceleration and long subsonic leg requirements on the U.S. SST Program. It was suggested at that time that the engine manufacturers should consider concepts which would provide both a multiplier in air flow as well as pressure ratio as a means to produce efficient matching for a variety of operating conditions, especially when efficient sustained supersonic cruise is desired. Figure 1, showing an array of ideas worth examination, is extracted from one such reference to illustrate the manner in which our thinking must be expanded to achieve good supersonic cruising vehicles which could also support secondary missions in an effective manner. It is recognized that the degree to which one can sell supersonic cruise depends on cost-effectiveness and, hence, multi-mission capability is fundamental to the consideration of such engine concepts.

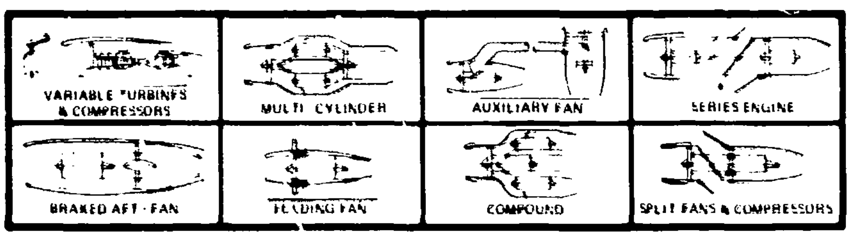


Figure 1 Some Variable Cycles

From those days, the U.S. technical community has moved slowly in achieving definition of practical engines for such missions. Figure 2 illustrates the varying degree of increased engine sophistication which can now be considered in examining the two missions in this paper. For the short-to-medium-radius mission of the air-to-surface attack penetrator, variable cycle engines which provide limited to moderate air-flow variation through variable turbine area and variable nozzles and compressor stator blades in combination with high throttle ratio (high flow at cruise), are being examined to identify their comparative merits on such missions.

MIXED BURNING (CURRENT)
- Variable geometry fans and compressor
- Conventional control

HIGH THROTTLE RATIO VGT
- Increased airflow control
- Variable area turbine

DUCT BURNING
- Increased airflow control
- Adjustable nozzles
- Variable compressors

VARIABLE PRESSURE RATIO
- Pressure ratio control
- Bypass ratio control
- Adjustable nozzles

REAR VALVED
- Pressure ratio control
- Bypass ratio control
- Adjustable nozzles

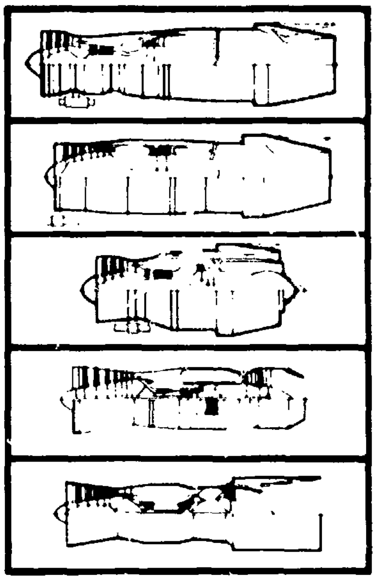


Figure 2 Engine Candidates

2.0 MEDIUM-RANGE AIR-TO-SURFACE ATTACK AIRPLANE

A possible primary and alternate mission profile for the air-to-surface attack aircraft is shown on Figure 3. Such a vehicle would be expected to operate a large percentage of its life at dry supersonic cruise near 40-50,000 feet altitude and at speeds in the neighborhood of Mach 1.6 - 2.0 depending on engine availability for dry supersonic operation.

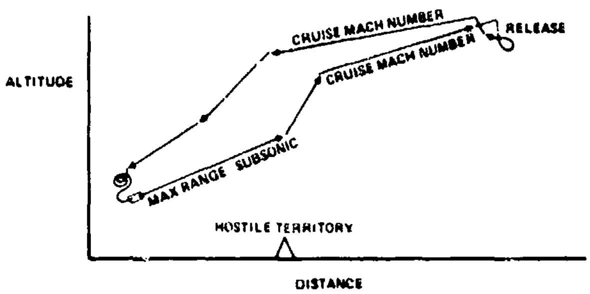
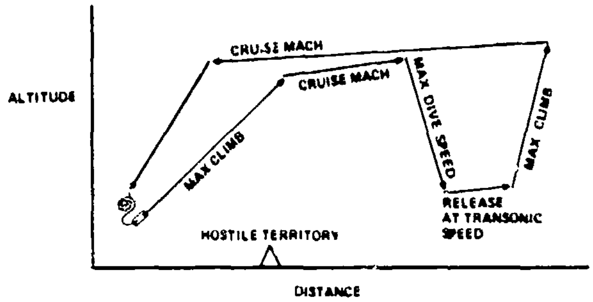


Figure 3 Air-to-Surface Medium Range Mission



Alternate Air-to-Surface Medium Range Mission

This aircraft may further be required to perform evasive maneuvers at these speeds though not engage in true air-to-air fighter activities. On alternate missions or as a part of the same mission, it may be required to operate at lower altitudes and high transonic speeds for weapon release with a much higher maneuverability requirement. The degree to which such activities can successfully occur will depend, in part, on the ability of the airframe and engine manufacturers to develop very low drag aircraft and a low total system weight with weapons installed.

To do justice to the supersonic cruise/surface attack mission, the aircraft must be stealthy (low I.R. radar cross-section, and be relatively small) and it must be able to maneuver at reasonably high altitudes. Hence, one task is to design an aircraft with very low drag characteristics, with weapons conformally installed, over much of the flight envelope and with minimum overall vehicle/weapon weight. Figure 4 shows an artist's conception of such a vehicle presented to illustrate the clear admission of supersonic cruise capability. Figure 5 is representative of the drag levels which may be possible for such a vehicle. Further, it must be the objective to cruise on dry power for reasons of low I.R. and to maintain small aircraft size. One major concern, therefore, is the engine design philosophy.

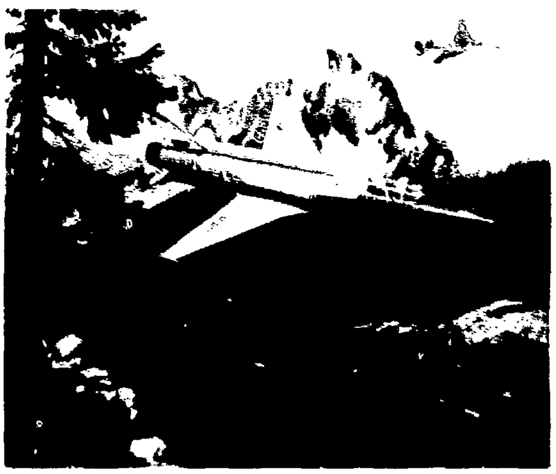


Figure 4 Medium Range Air-to-Air Surface Attack Aircraft

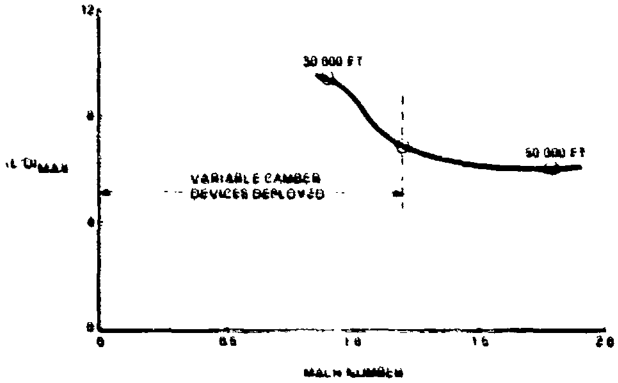


Figure 5 Typical Aerodynamic Cruise Efficiency for Strike Aircraft

2.1 High Throttle Ratio

A first considerat'on for a true supersonic a.r-to-surface attack vehicle, designed to cruise in the M = 1.6 - 2.0 region, is to point-design the engine for the supersonic condition and to make whatever adjustments are needed to the cycle to take care of other requirements. This thought process leads to the use of High Throttle Ratio in design, whether the engine is a "leaky" turbojet or a modest-bypass turbofan.

Traditionally, combat engines in the past have been designed at sea level static conditions (equivalent to M - 1.28 at 36,080 feet from $\sqrt{\theta}/\delta$ considerations) and as long as prolonged flight was in the high subsonic to transonic speed region, no apparent serious penalty was incurred.

Throttle Rat'o, ϕ, in this pape', is defined as the ratio of maximum cycle temperature at supersonic cruise to sea-level-takeoff temperature. In the past, the industry h's commonly used ratios less than unity. For the propo'ed aircraft, one must think of ra ios well ir excess of unity. The choice of design throttle ratio will depend on the drag of the aircr ft at cruise as well as th at transonic maneuver. Augmentation would be used as required for takeoff, climb and maneuver. Figure 6 illustrates the throttle ratio concept. Engines designed at sea level static and then operated at a Mach number of 2.0 suffer a considerable reduction in dry thrust due to requirements for compressor air flow matching at all inlet temperatures across the complete Mach number spectrum of the operational aircraft.

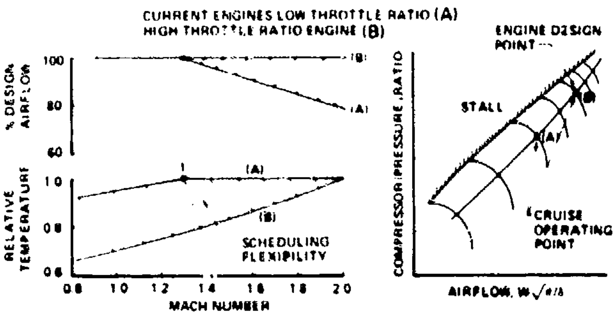


Figure 6 Throttle Ratio Scheduling

The HTR design principle has evolved over the years from the continual quest to obtain maximum pressure rise per stage and, hence, fewest number of stages resulting in the aforestated desire for highest thrust-to-weight ratio engines with the shortest length. With this type of design philosophy, the maximum turbine inlet temperature is a fallout as a function of the power requirement the turbine is called upon to deliver at the higher Mach number condition. The maximum usable power is limited by the disc structure again set by material technology and the ever present quest for minimum weight. Since turbine nozzle area has always been fixed, and the pressure supplied to the turbine is limited by available rotor speed, the power output is adjusted and limited by the turbine inlet temperature.

The result of this design philosophy has been that the nonafterburning thrust-to-weight of our engines at altitude/Mach number is quite low. As a rule, it is frequently not quoted. On the surface it would appear that any deviation from this approach would cause the engine to be extremely heavy because the engine would be forced to be long and all discs stronger and, hence, heavier. In fact, studies have suggested that the expected weight penalty may not be fundamental to an engine with a different basic design philosophy; namely, we are proposing to design the engine, including compressor and turbine aerodynamics at or near the planned cruise speed such that corrected air flow remains constant. When this is done, as shown on Figure 6, it is temperature derated at reduced Mach numbers to match the fixed turbine area and fixed compressor operating point. This now leads to reduced transonic and takeoff thrust, which must be offset with augmentation or by a combination of other concepts to be discussed shortly.

Studies suggest that the consequence of a High Throttle Ratio design can be enormous. In one such study, the dry turbojet size was reduced some 11% in diameter and 26% in frontal area for a composite structure Mach 1.8 airframe design when compared to a conventional engine with dry power. This is illustrated in Figures 7 and 8. For this design, we chose to use a 1.25 throttle ratio as shown in Figure 7. The installed cruise SFC was reduced by the HTR. Additional benefits may be possible with a simplified intake because the engine operates at constant corrected air flow across the Mach number spectrum, suggesting that the inlet might be simplified with the fixed throat requirement. At transonic conditions for this highly tailored aircraft design, the HTR engine offered marginally acceptable dry thrust for one "g" flight. The thrust could be corrected, in part, with application of augmentation. However, as shown in Figure 9, when the HTR feature is combined with a variable area turbine (Reference 2) the cycle can be rematched to allow a greater temperature rise in the combustor (before turbine area becomes limiting) and, hence, a large increase in transonic thrust.

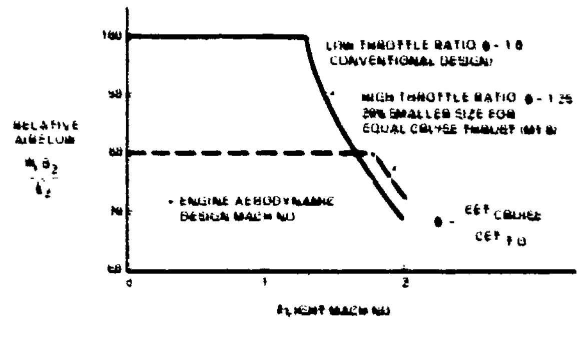


Figure 7 Turbojet Size Variation With Throttle Ratio

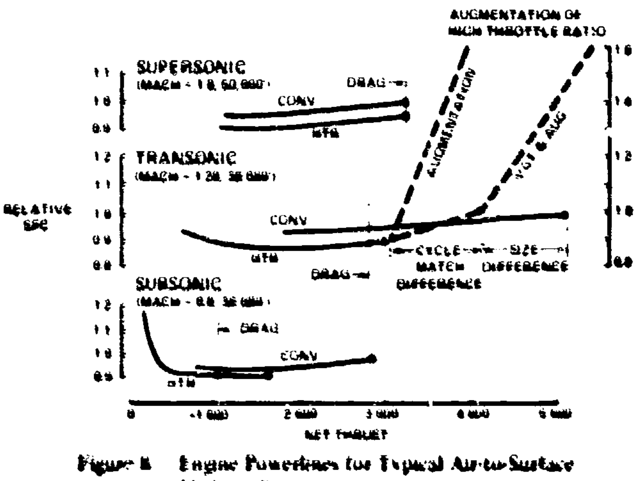


Figure 8 Engine Powerlines for Typical Air-to-Surface Medium Range Airplane

The effect of HTR on takeoff thrust is illustrated in Figure 9, showing that the merits of HTR are taxed where available field lengths are short, and some oversizing (reduction in throttle ratio) may be required to increase full A/B takeoff, if additional aircraft lift augmentation cannot be developed by aerodynamic means. It is compromises like these which may in the limit establish the top cruise speed which can be developed effectively with dry thrust. It is doubtful that the HTR principle can be considered for design cruise speeds much in excess of Mach 2.0.

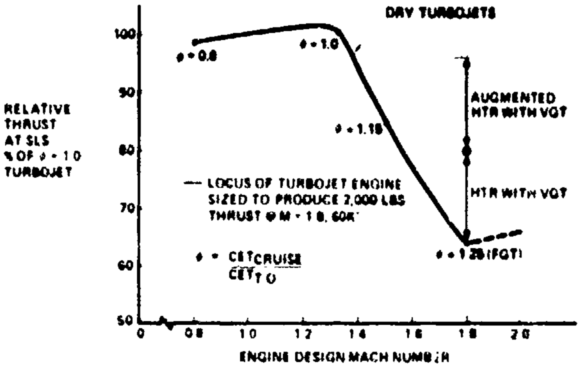


Figure 9 Takeoff Thrust Improvement for HTR Engines

2.2 Mission Requirements

For the purpose of illustration, a primary mission is defined as occurring from an airfield of fixed maximum field length followed by a climb to a maximum subsonic cruise range altitude. Enroute to the entrance to hostile territory, evasive maneuvers of substantial "g" force are considered a requirement. After following the flight over a fixed subsonic leg, the aircraft climbs with A/B power to the supersonic cruising altitude in minimum time. The aircraft levels off at supersonic design Mach number with the A/B shutdown, crosses into hostile territory and performs a radius mission. The aircraft returns to base, initially at supersonic cruising altitude, then at subsonic maximum range altitude, and finally loiters for a fixed period of time and lands. This mission is illustrated on Figure 10.

The analysis method used to compare engine concepts involve a fixed aircraft gross weight; the airframe variables are optimized consistent with the engine cycle concept; all elements of the mission were prescribed with the exception of the supersonic radius capability; the measure of goodness being the comparative supersonic ranges each engine concept offers.

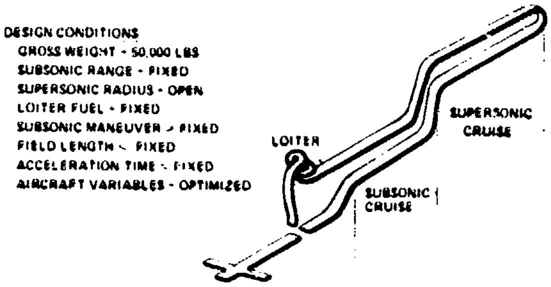


Figure 10 Medium Range Ground Attack Mission

	RANGE OF INTEREST
WING ASPECT RATIO	1.3 - 4.3
WING LOADING	50 - 150 LBS/FT²
WING THICKNESS TO CHORD	.030 - .080
WING LEADING EDGE SWEEP	34° - 74°
AIRPLANE THRUST TO WEIGHT	.70 - 1.40
TAKEOFF GROSS WEIGHT	50000 LBS

Figure 11 Considered Airframe Variables

The airframe range of variables used to describe each configuration are shown on Figure 11. It has been found that variation of airframe configurations while examining engine concepts is very important and that simply freezing an aircraft configuration for picking the proper engine concept leads to very erroneous results. It is often necessary to reconfigure the aircraft layouts when the analysis suggests the configuration has become far removed from that for which the basic drag polars and engine installation arrangement were originally developed. This is particularly true when the operations field length, minimum maneuver and acceleration capability are varied widely in the study.

2.3 Propulsion Systems

Five engine types were chosen for this paper. The design bypass ratio, pressure ratio, and cruise throttle ratio are shown on Figure 12. All engines were specified to have the same technology in terms of combustor exit temperature, cooling air flow and component efficiencies. A complete study requires variations on each of the given engine definition values, but for purposes of illustration, they are omitted from this paper.

Figure 13 shows the relative air flow lapse rate for the engines being evaluated. It must be pointed out that the break in air flow schedule is quite arbitrary for the purposes of this study and would be

	BYPASS RATIO	OVERALL PRESSURE RATIO	MAXIMUM COMBUSTOR EXIT TEMP	THROTTLE RATIO*	AUGMENTOR TYPE
LOW THROTTLE RATIO (LTR) JET	20	16	CONSTANT TECHNOLOGY	1.06	AFTERBURNER
HIGH THROTTLE RATIO (HTR) JET	20	16		1.34	AFTERBURNER
VARIABLE GEOMETRY TURBINE (VGT) JET		16		1.60	AFTERBURNER
MIXED FLOW TURBOFAN (AB FAN)	1.30	20		1.16	AFTERBURNER
SEPARATE FLOW TURBOFAN (DB FAN)	1.30	20		1.17	DUCT BURNER

*THROTTLE RATIO = $\frac{CET_{CRUISE}}{CET_{T.O.}}$

Figure 12 Selected Engine Cycle Definition

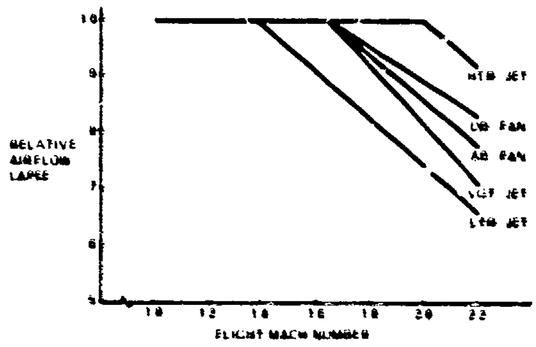


Figure 13 Comparative Engine Airflow Demand

examined in great depth for an actual engine/airframe investigation. In fact, this parameter must be made a variable so as not to preclude the answer with the original assumptions. Figure 14 shows the comparative thrust lapse of the resulting engines and Figure 15 shows the relative dimensions of these engines at a constant airplane T/W.

2.4 Airplane-Engine Evaluations

Figure 16 shows the effect on optimum airframe variables for the given engines of this paper, when maximum supersonic radius is computed at a fixed takeoff thrust/weight ratio and a fixed, relatively long subsonic leg with field length and acceleration not considered (unrestrained). Different airframe design variables are suggested for the assumed fixed T/W to achieve maximum supersonic range. It is important that the airframe variables be considered as major drivers as they will affect both field length and the basic configuration as well. The other major driver is the required subsonic radius. It will be shown later that T/W is also a major variable that must be optimized along with the cycle concept and the airframe variables

Figure 17 shows the relative supersonic radius versus T/W. Airplane variables are optimized at each T/W until a minimum T/W is reached where the airframe no longer meets the selected "g" maneuver or acceleration time (in full A/B). Hence, the primary conditions of interest are the solid points on Figure 17, reconstructed on Figure 18 for comparison. Each engine requires a considerably different design T/W and, hence, airframe configuration. Note that the HTR-turbojet produced the best range at the lowest thrust-to-weight ratio; however, field length requirements were still unrestrained. Figure 19 shows results at a T/W of 0.90 selected to illustrate an aircraft with better short-field takeoff capability. By comparing at the fixed T/W, a whole new set of engines appear to provide the best range which illustrates the extreme importance of properly selecting the study assumptions. It was shown earlier that the HTR-turbojet would not be expected to have best takeoff thrust by comparison to a conventional design; however, the Figure 18 comparison shows that where operating field length is not as critical, the HTR turbojet can perform with best supersonic radius considerably in excess of the competing engines. However, where field length considerations dictate the design (Figure 19), this concept appears to suffer considerably. There may be a solution to this difficulty. The Variable-

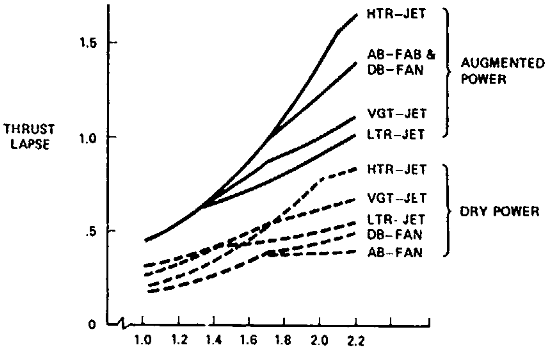


Figure 14 Comparative Thrust Lapse

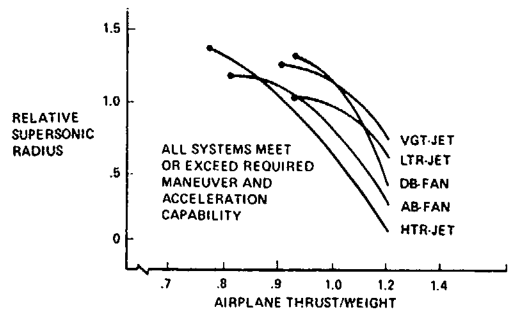


Figure 17 Maximum Supersonic Radius

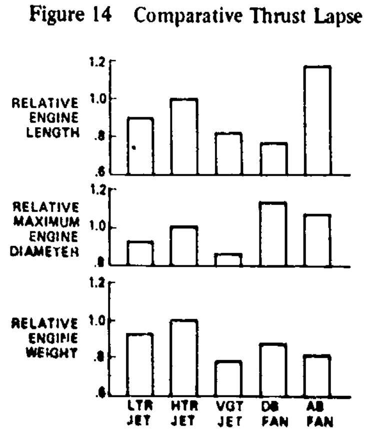


Figure 15 Engine Dimension and Weight Comparison at Equal SLS Maximum Thrust

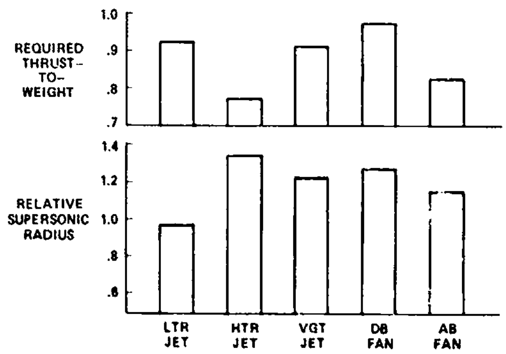


Figure 18 Optimum System Performance at Constant Acceleration and Maneuver Requirement

		LTR JET	HTR JET	VGT JET	DB FAN	AB FAN
AIRPLANE	AR	4.3	4.2	4.3	4.3	4.3
	T/C	030	.030	.030	090	.030
	W/S	120	120	135	115	120
	SWEEP	40°	45°	47°	34°	43°
SUPERSONIC	PS	AUG	DRY	DRY	AUG	AUG
	SFC	1 87	1 37	1 49	1 64	2.09
	L/D	3 7	4.9	4.9	4.6	4.8
SUBSONIC	PS	42	62	44	70	71
	SFC	1 25	1 15	1 49	.94	1 04
	L/D	11 1	10.2	9.9	16 5	10.7

Figure 16 Maximum Supersonic Radius System Definitions With No Performance Requirements
(T/W = .90 TOGW = 50,000 Lbs)

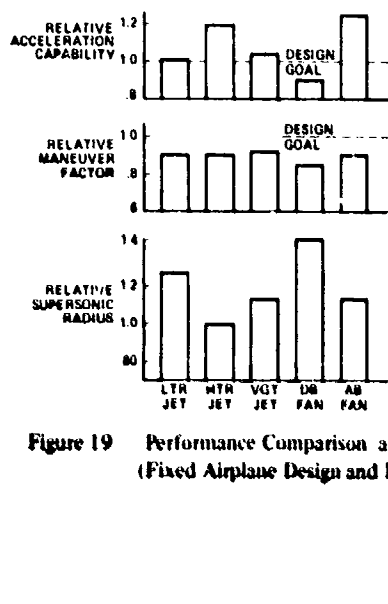


Figure 19 Performance Comparison at T/W = .90 (Fixed Airplane Design and Equal Field Length)

Geometry Turbine (VGT) turbojet appears to offer an exciting means to offset this deficiency, thus suggesting that a second iteration of these engines may include a compromise involving an engine where both HTR and VGT are incorporated. Figure 20 shows a comparison of these same engine concepts when the engines are sized to meet maneuver margin and acceleration time requirements with a fixed airplane design. Note that, again, the relative performance of the engines show a considerable shift which illustrates the large sensitivity that study ground rules, airplane variables, and mission requirements can play on engine concept evaluations.

In all the results thus far, the degree of afterburning has been allowed to float. Therefore, in the case of the turbofan, a large amount of A/B takes place and less for the turbojet. What would be the case if the engines were constrained to dry supersonic cruise to minimize IR signatures? Figure 21 shows the results of such a study with the same basic mission assumptions as Figure 20. The fan engines fail to perform the supersonic mission and the jet engines are best. Both the HTR and the variable geometry turbine jet (which also has HTP capability) perform well. Hence, to perform an adequate study, not only should the engine concepts contain an array of variables, but also those of the aircraft, and finally the mission rules should be sensitized to determine where the best opportunities exist.

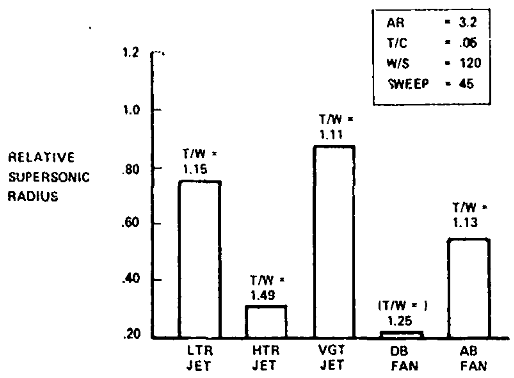


Figure 20 Supersonic Radius with Required Maneuver and Acceleration Capability (Fixed Airplane Design)

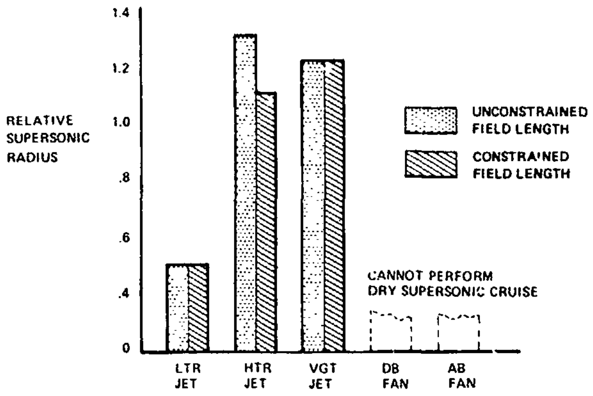


Figure 21 Optimum Systems for Dry Supersonic Cruise

In the examples shown, several interesting engine concepts appear to be competitors. The use of VGT in combination with HTR shows up as a primary candidate although further refinement of the process is necessary before a strong final position can be taken.

3.0 LONG RANGE SUPERSONIC AIRPLANE

A second military system which may soon get increased attention is either the SR-71 replacement or a scaled-down super-Concorde. The technology which has evolved since such programs were initiated some 15-20 years ago, by ongoing developments, suggest that much larger ranges are currently possible with these vehicles at much lower weights and, hopefully, at greatly reduced systems cost.

An apparent major reason the military have shied away from expanding the role of such a global system is cost. The importance of engine size and cruise fuel consumption (airplane drag and engine SFC) at very high altitudes suggests that the engine concept may want to be entirely different than that for the air-to-surface tactical vehicle. Extensive high altitude loiter and long subsonic legs may additionally be a requirement that could be met to add interest in this vehicle. Since such systems have been costly in the past, every effort must be taken to make such an aircraft small. Figure 22 suggests possible primary and alternate missions for such a vehicle.

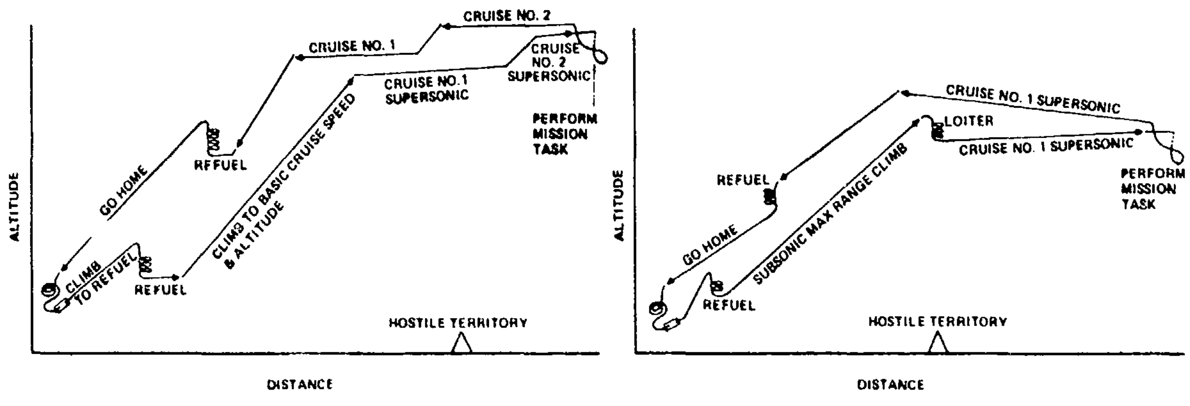


Figure 22 Basic Long Range Mission — Alternate Long Range Mission

With the great strides which have been made in the past twenty years in titanium sheet, brazed and bonded honeycomb structure, advanced high temperature composites, sealants, intakes, digital flight controls, and aerodynamics, it is now possible to design an outstanding global recon/strike system. Flight in the Mach 2.7 - 3.5 region can now be combined with reasonable maneuverability, good loiter and outstanding subsonic range, to provide far better military cost effectiveness than that offered by a derivative SR-71 or a scaled Concorde.

The long-range recon/strike vehicle requires an additional examination of more sophisticated engines because of the much higher cruise mach number and altitude spectrum. These engines may need to provide large changes in operating pressure ratio as well as air flow to achieve the substantial gains in fuel consumption and specified ranges necessary for the widely diversified use of the aircraft. Hence, true dual-cycle engines, such as shown in the bottom of Figure 2, will be examined. In all cases, augmentation can be added either in the main stream, the fan duct stream, or both to provide thrust for overloaded takeoffs, transonic climb, maneuver, and for very high speed, high altitude dash.

3.1 Mission Requirements

Figure 22 illustrated two of many possible missions for such a vehicle. It is essentially a very high altitude global penetrator. It must be small and, hopefully, simple to maintain to keep its life cycle cost down. Because of these goals, it may require one or more aerial refuelings to perform the longer missions. The design will minimize the number of required refuels as a part of the low life cycle cost goal.

For such an aircraft to be considered as viable, the engine is once again a major factor. Simply combining variable compressor and turbine geometry with some form of augmented turbojet or turbofan will probably not be enough. Dry cruise is essential for good range factor. However, in the cruise range of M = 2.7 - 3.5, the design will require outstanding internal compression intake performance in combination with a very distortion-forgiving compressor/fan system. Ideally, a low pressure ratio turbojet is what is wanted at these cruise speeds. The problem gets even more complex when one wants to combine this requirement at more than one supersonic cruise speed (best range speed plus flight at specified altitude), loiter at high altitude, and subsonic engine-out return to base. Such aircraft would be expected to have no operating field length restrictions.

3.2 Propulsion Systems

Extensive study on supersonic commercial transports has developed prejudices on engine cycles of interest which are closely allied to those of the long-range recon/strike aircraft (namely, long supersonic range, long subsonic range, extended loiter). NASA AST studies have caused engine and airframes to consider engine types more typified by those shown in the bottom of Figure 2. In designs which try to satisfy these different mission objectives HTR, variable area turbine, and compressor discharge bypass have only a minimal benefit; mainly what is needed is a full convertability from a relatively high bypass ratio turbofan to a turbojet. Airframe trades in such areas as aspect ratio, sweep and wing loading are made to support the engine assessment. Due consideration of intake and nozzle performance, weight and complexity must further be included. Engine manufacturers often overlook the intake and nozzle problems in performing these studies because of the difficulty for them to include the system integration effects in adequate depth. The performance, weight, air flow matching, control and drag of such components are at least at the same level of importance as the gas producer. In deciding the complete propulsion configuration (intake, gas producer, nozzle) all factors must be considered with equal importance in the design of a Mach 2.70 - 3.50 engine system.

Such concepts as multiple bypass engines with flow diverters, and fan duct burning are currently being investigated by the engine industry. The schemes shown in Figure 2 are in addition to these to indicate there is plenty of room for thinking. Figure 23 shows an enlarged view of two potential alternate schemes for producing bypass ratio variability (RVVCE) and bypass and pressure ratio variability (VPR).

In this study five engines were compared, all incorporating the same technology, but with different configurations. The dry-cruising turbojet, with a short A/B for acceleration use only, is used as the frame of reference. The duct-burning turbofan (DHTF), after-burning turbofan (ABTF), rear-valved variable cycle (RVVCE), and the variable pressure ratio turbofan (VPR), are the chosen concepts. Basic cycle data is shown in Figure 24, and comparative performance data are shown in Figures 25 and 26.

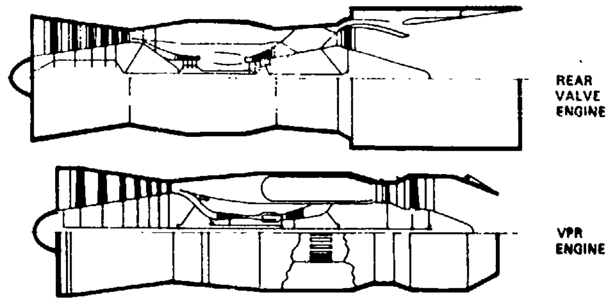


Figure 23 Typical Variable Cycle Engines for Long Range Penetrator

MACH 3.0 DESIGN

ENGINE	BPR	OPR	FPR	CET
TURBOJET (TJ)	0	12	-	CONSTANT TECHNOLOGY
VARIABLE PRESSURE RATIO (VPR)	2.2	32	6.5	
DUCT HEATING TURBOFAN (DHTF)	1.3	13	3.3	
REAR-VALVE VCE (RVVCE)	2.5	13	5.8	
AFTER-BURNING TURBOFAN (ABTF)	1.9	15	2.5	

Figure 24 Engine Cycle Characteristics

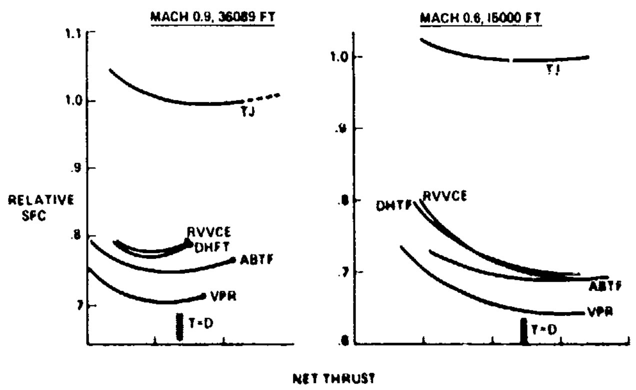


Figure 25 Subsonic Engine Performance Comparison

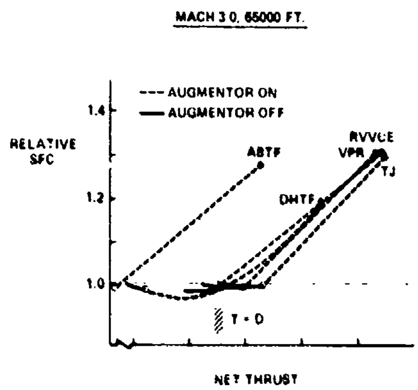


Figure 26 Engine Performance Comparison

3.3 Engine-Airplane Evaluations

The study was conducted by sizing the engines to fit a twin-engined aircraft incorporating aircraft study variables consistent with a long-range vehicle at fixed gross weight, see Figure 27. Figure 28 shows the capability of the dry turbojet to perform a design mission (A) and selected alternate missions (B, C and D). The design mission was selected with takeoff from an unrestricted field, followed by climb and cruise at Mach number 0.90 until the aircraft requires refuel. It then refuels and climbs to supersonic best cruise speed and altitude (in this case, M = 2.7 and 65,000 feet average). It cruises out to the hostile territory at which time it climbs to 80,000 feet and Mach 3.0 and performs a fixed radius mission of 500 nautical miles. It then returns at Mach 2.7 to the refueling station, decelerating to Mach 0.90 for refuel and heads home. The open condition is the range at M - 2.70 available assuming fixed aircraft and engine size. As shown on the top of Figure 28, an unrefueled subsonic range of 2,800 nautical miles was obtained for this aircraft with the baseline turbojet engine.

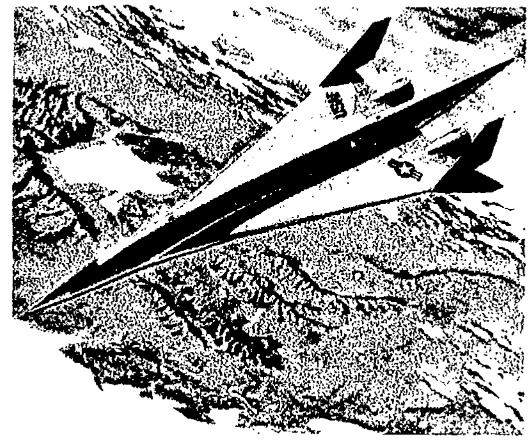


Figure 27 Long Range Penetrator

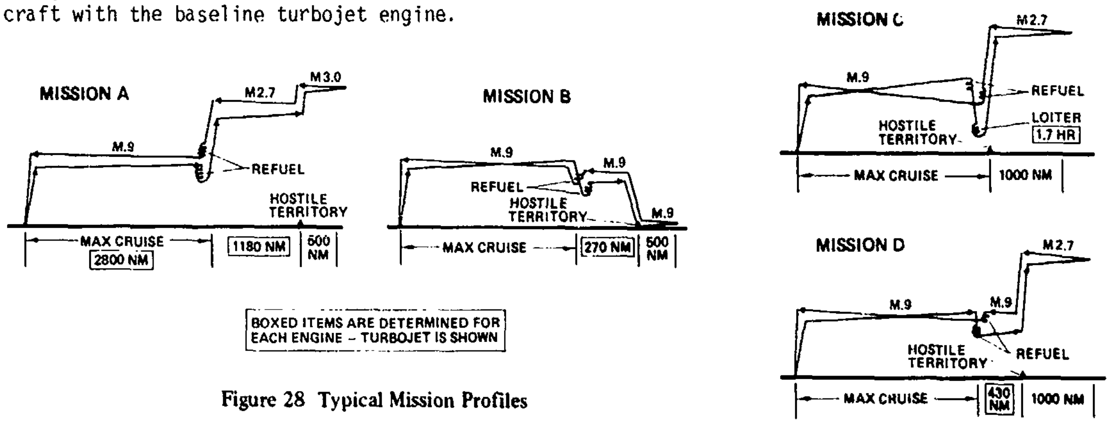


Figure 28 Typical Mission Profiles

Using the same aircraft and engine, Mission B demonstrates a low altitude penetration of 500 nautical miles radius with refueling taking place as far away from hostile territory as possible, which for the turbojet is 270 miles. Mission C shows that this same aircraft can loiter for 1.7 hours prior to a high-altitude penetration of 1,000 miles radius at M = 2.70. Loiter fuel can also be used to increase the refuel distance from the hostile territory as shown in Mission D (430 nm). These alternate mission choices were arbitrarily selected to indicate potential benefits only. The maximum subsonic cruise range (up to refuel point) included in missions A, B, C and D varies for each engine type. The relative length of the subsonic cruise leg for each engine is shown in Figure 29 and applies for all missions.

Figures 29 and 30 show a comparison of the figures of merit for the five engines considered. The baseline dry-cruising turbojet compares favorably with the other engine concepts in Mission A; however, the variable-cycle engines appear to be somewhat superior. In the other three missions, the VCE schemes are considerably better.

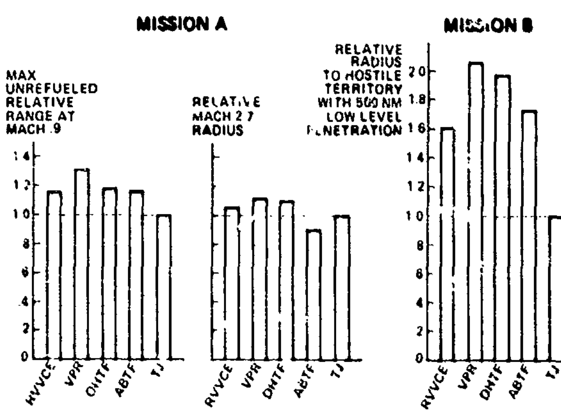


Figure 29 Long Range Penetration Mission Comparison

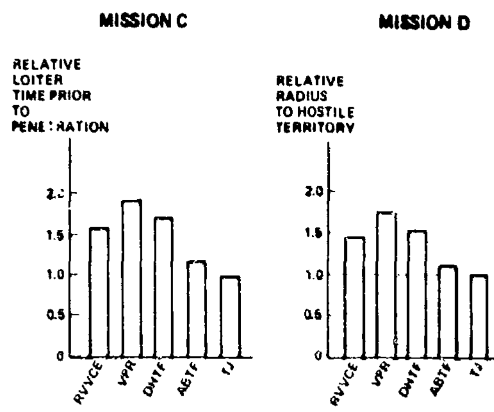


Figure 30 Long Range Penetrator Mission Comparison

The principal difference in engine concepts is that the RVVCE and the DHTF are configured to provide best air flow match on all mission segments with the optimum throttle ratio. These engines also have best specific thrust at all conditions, but are limited in that cycle pressure ratio is not optimum for all mission segments. The VPR on the other hand shows the best performance of all schemes. This results from the good air flow match combined with the added capability to optimize cycle pressure ratio. These comparisons show that considerable reason for further development of variable cycle engine components is justified, primarily because of the much better mission versatility inherent with these engines.

4.0 CONCLUDING REMARKS

In this paper, two examples of potential military aircraft have been considered and study results on the effect of selected variable cycle engines have been illustrated. Many other missions in such areas as V/STOL and logistics carriers reveal a similar interest in variable geometry engines. The proper solutions to such matters can only come about if fully integrated studies of airframe variables and engine variables are simultaneously conducted. These studies require a concentrated, combined activity of the airframe and engine contractors. It is hoped that this paper reflects a contribution to the needs that exist if we are to achieve the maximum potential at a minimum airframe size and cost.

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3. "Performance Problems Related to Installation of Future Engines in Both Subsonic and Supersonic Transport Aircraft," W. C. Swan, The Boeing Company, Royal Aeronautics Society - 2nd International Symposium on Air-Breathing Engines, March, 1974.

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DISCUSSION

J.F.Chevalier

Vous avez démontré qu'il fallait prendre en compte tous les paramètres variables pour faire une bonne optimisation, et vous avez montré comme exemple caractéristique de l'erreur qu'on peut faire le cas d'un HTR jet installé sur un avion à rapport poussée/poids au décollage fixé. Or, nous n'installerions jamais un HTR jet dans un avion à rapport T/W donné. En effet, si nous améliorons la poussée à grande vitesse de ce moteur, c'est pour réduire la taille du moteur; donc nous aurions automatiquement étudié un avion au rapport T/W au décollage réduit. Nous n'aurions pas fait l'erreur que l'on commet si l'on suit la deuxième méthode de votre communication.

You have mentioned that, to achieve good optimization, all the variable parameters have to be integrated, and you have given as an example, the installation of a HTR jet on an aircraft with a fixed thrust to take off gross weight ratio. Now, we would never install a HTR jet on a fixed T/W ratio aircraft. As a matter of fact, if we improve the thrust of an engine at high speeds, it is for the purpose of reducing the size of this engine. Therefore, we would automatically consider an aircraft with a reduced thrust-take off weight ratio. We would not make the mistake which would be made if we adopted the second method discussed in your paper.

Author's Reply

In airplanes which have been designed today, which cruise with M = 1.2 with dash up to M 2.0, there is no reason to talk about throttle ratios greater than 1 because engine design at sea level is exactly the same as the engine design at Mach No.1.28 and if you are only going to dash to Mach No.2 the time you are at supersonic speed is very short and therefore the use of an afterburner makes sense because of its light weight. But the plane which will dash subsonically and cruise supersonically, one should be careful not to design the engine at sea level static or it will result into a very large engine for supersonic cruise. This is the essence of what I was trying to express.

R.M.Denning

Is not the Olympus engine in Concorde an example of the variable geometry turbo jet with afterburner used at off-design conditions?

Author's Reply

Concorde, being a Mach 2 airplane, is right at the upper limit as to whether high throttle ratios make sense according in our studies. I feel high throttle ratio makes a lot of sense if you are cruising at Mach 1.6. If you get up to around Mach 2 the advantage begins to disappear. I do believe, however, if the Olympus engine had a variable area turbine, you could have cut down the amount of augmentation at takeoff for the same total thrust and it would not be as noisy an aircraft. But again I recognize this was not a fixed weight aircraft during the design phase and the way it was changing force you to use augmentation for takeoff.

R.M.Denning

On the variable pressure ratio engine with the bypass duct turbine is there not a mismatch in air angles on this turbine when the mode of operation is changed?

Author's Reply

In the case of the variable pressure ratio engine you will note first that the fan is one of very high pressure ratio. It is of the order of 3.5 to 4. This is something we have never built before in an engine and which then minimises the difference in the air angles the turbine sees. Secondly, I am not precluding the use of a first stage stator variability in the low pressure turbine. As a matter of fact we do not use a stator now in order to get around the problem you are talking about.

SOME ASPECTS OF VARIABLE CYCLE
PROPULSION SYSTEMS

by

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SUMMARY

The paper first considers the incentives which encourage the study of variable cycle gas turbine powerplants. In addition to achieving performance gains at off-design running conditions, and increasing safety and flexibility by providing greater operating margins, there are now significant attractions from the environmental point of view in regard to the control of both exhaust emissions and noise.

The gains potentially available in a number of civil and military aircraft applications are outlined, together with the resulting requirements for cycle variation and the implications for the configuration of the gas turbine engine itself and its intake and exhaust systems. The substantial component and system design problems posed by advanced variable cycle powerplants are discussed.

1. INTRODUCTION

A 'variable cycle' propulsion system might be defined as one which incorporates provision for the significant variation of the thermodynamic cycle at selected operating conditions. This definition clearly embraces a wide spectrum of possibilities. At the same time, it is intended to exclude effects purely associated with the use of component matching aids for low-speed running in conventional engines.

Deliberate cycle variation may be achieved by a variety of means, ranging from variable propelling nozzles, through more complex design features such as variable capacity turbines, to systems involving gross airflow switching to obtain alternative series or parallel operation of major powerplant components. Thus, some degree of added complexity is an inevitable characteristic of any variable cycle solution to a powerplant project requirement. The level of technical uncertainty is generally increased, as are the risks of reduced reliability and high maintenance costs. For a variable cycle solution to be adopted, such factors need to be out-weighed by very substantial incentives. This paper offers a broad discussion of the incentives, design features, problems and prospects of variable cycle gas turbine powerplants, illustrated by a number of potential applications considered in studies at NGTE.

2. INCENTIVES

As a form of prime mover, the gas turbine offers great design flexibility. Due to the steady flow nature of the engine, each major component has to handle only one thermodynamic process. It is therefore possible to assembly appropriately designed compressors, combustion systems, turbines and ducting in various arrangements to provide - within limits - any desired thermodynamic cycle. Thus the current aeronautical scene features jet engines having design pressure ratios ranging from less than 5 to about 30, by-pass ratios from zero to about 8, and turbine entry temperatures from less than 1100K to over 1600K.

Notwithstanding this basic design freedom however, the very large flight envelope possessed by many types of modern aircraft inevitably leads to compromise in the choice of the powerplant cycle, and operation far from the design point at some flight conditions. The thermodynamic cycle and airflow may then be relatively badly matched to the duty required. At low power, cycle efficiency may be poor. If engine airflow demand is low, installation drag may be high. Both effects worsen fuel economy. An alternative possibility is that there may be difficulty in achieving the required thrust level due to a stress, temperature or aerodynamic limit being reached within the powerplant.

In such situations there is clearly scope, at least in principle, for achieving performance gains by altering the cycle to improve the matching between the powerplant and its duty. The nature of the required cycle change depends on the particular performance improvement desired. Assessment in the context of the aircraft application, making proper allowance for changes in installation drag, is always necessary. It should be noted that although variable cycle features add complexity to the powerplant and its control system, an improvement in thrust capability at a critical flight condition could allow an engine of smaller size and possibly lighter weight to be used for a given aircraft mission requirement.

In addition to performance benefits in terms of steady-state fuel economy or thrust, some variable cycle features can be deployed transiently to improve engine handling or provide faster thrust response capability. Such techniques may also be useful for temporarily lowering the working line on a compressor characteristic to improve operating margins under extreme flight conditions where severe inlet airflow maldistribution is liable to occur.

Environmental requirements, now of major importance in civil aircraft design, constitute another class of incentives for the study of variable cycle powerplants. Internationally agreed legislation to limit the noise of new types of subsonic aircraft has been in force for a number of years, and is likely to become more stringent in future. In addition to a lowering of the noise levels permitted for subsonic aircraft, such legislation may well be extended to cover civil supersonic transports. The noise produced by an aero-engine arises from a number of sources, whose relative magnitudes depend very much on the design of the engine. Cycle variation may be used to reduce a dominant noise source, for example by decreasing low pressure rotor speed to lessen fan noise, or alternatively by increasing total airflow at a given thrust level to reduce jet velocity and hence jet noise.

The need to reduce certain exhaust effluents is the other environmental constraint now affecting engine design. Much attention is currently being devoted to this problem, largely under the stimulus of requirements framed by the US Environmental Protection Agency (EPA). These requirements cover several pollutants, with the reduction of oxides of nitrogen (NO_x) presenting the greatest technical challenge. Although the factors which influence the production of NO_x in engines are still the subject of study and debate, it appears that NO_x levels may be expected to increase as combustion chamber entry temperature rises - ie the NO_x problems tend to be accentuated by the modern trend to high pressure ratio engines. A variable cycle capability which allows engine pressure ratio to be somewhat reduced at the take off condition, may therefore be worth consideration as one means for reducing NO_x emission on the ground and at low altitude. At the same time, it is important from a fuel economy viewpoint that high pressure ratios should still be available for cruising flight.

Summing up regarding the current incentives for the study of variable cycle powerplants, these may be stated in terms of the following broad headings:-

(a) Incentives directly associated with the steady-state performance capability of the powerplant - improvements in fuel economy or thrust capability; relief of stress or temperature in high duty components at extreme flight conditions, etc.

(b) Improvement in environmental conditions - noise and/or exhaust emissions.

(c) Improved thrust response and other operational factors.

3. TECHNIQUES AND DESIGN FEATURES

This section first considers the principles which govern the matching and operation of a gas turbine powerplant at conditions away from its design point. With this as a basis, the means by which controlled cycle variation may be achieved are then outlined and discussed.

For equilibrium running at any flight condition and power setting, the propulsion system must operate such that various work, heat and pressure balances, and flow continuity, are maintained. As a result, each component of a conventional powerplant is constrained to operate along a 'working line' or within a restricted region, departing only during transient non-equilibrium operation when the steady-state work balances no longer hold. In this overall matching process, a dominant part is played by the effective flow areas of the powerplant expansion and exhaust systems.

At the same time, it is essential for satisfactory running that each component shall always be working within its boundaries of stable operation. It is usually the compression system which gives problems in this respect, and therefore considerable attention has always been paid to the off-design performance characteristics of compressors. An important outcome has been the successful development and widespread use of variable stagger inlet guide vanes and stator blading. In modern high pressure ratio compressors, for instance, variability may be incorporated in perhaps five or six stator rows to obtain the 'corridor' of stable operation required for satisfactory starting and handling. Thus, considerable experience has already been gained in the aerodynamic, mechanical and control aspects of compressor stator variability. This existing body of experience provides a useful basis for the study of powerplants where deliberate control of the cycle at off-design conditions is envisaged, although in many such cases much larger variation of the compressor performance characteristics would be needed. For example, some concepts imply considerable pressure ratio variation, which might require the extension of variable blade stagger to include one or more rotor rows.

Returning to the downstream components which exert a controlling influence on the matching of the powerplant as a whole, it can also be noted that considerable experience exists on variable-area propelling nozzles. The development of such nozzles was given great impetus by the reheat or afterburning principle - itself a dramatic 'variable cycle' concept of fundamental importance for military aircraft with supersonic flight capability. More limited experience exists on variable by-pass mixing systems, which constitute a form of co-axial nozzle, operating unchoked, within the main jet pipe. Although involving some mechanical complication, these should pose no fundamental design difficulty.

Of much greater significance is the problem of achieving variation of turbine swallowing capacity. In view of the high temperature operating environment, it is fortunate that the swallowing capacity of a turbine can be changed very substantially by variation of its nozzles alone, without any need for manipulation of the rotor blading. Even so, the engineering of a variable capacity turbine so as to maintain high efficiency and satisfactory mechanical characteristics in long-term use presents a considerable technical challenge. Turbines with variable nozzles have been used in a number of engines, as well as for experimental purposes in component test rigs. Experience is therefore building up, and some encouraging results have been reported. For example, in a recent 'demonstrator' programme in which a small turbo-fan engine was modified to include a variable low-pressure turbine, performance comparisons with normal production engines "showed no significant penalties associated with the incorporation of variable geometry" (References 1 and 2). Close attention to detail design is clearly required to avoid efficiency losses due to low leakage via the root and tip clearances of pivoted nozzle vanes. Up to now, most if not all applications have been to turbines whose nozzles do not require an internal cooling airflow. The added complication of combining a pivot system with a cooling air supply is formidable and acts as a powerful deterent to the use of this means of capacity variation on high temperature turbines. As an alternative to pivoting of the nozzle vanes, some variation of swallowing capacity may be produced by arranging for portions of the annulus wall to be moveable. So far, only very limited experience of this technique appears to have been gained.

Another approach, analogous in matching terms to increasing turbine swallowing capacity, is to bleed off a proportion of the flow upstream of the turbine. Bleeding or 'blow-off' from the later stages of compressors is already an established method for improving compressor handling in the low/medium speed range. If applied between the high pressure (HP) and low pressure (LP) turbines, a similar effect to increasing the LP turbine swallowing capacity would be obtained - ie, the HP system working line would be lowered on its compressor characteristic. This bleed technique could offer a more straightforward design solution than direct turbine nozzle variability, particularly for engines where the LP turbine operates at temperatures such that nozzle cooling air is required. To avoid excessive circumferential flow disturbance, bleed ports distributed around the whole outer annulus wall would probably be needed. The bleed flow could either be reintroduced into the jet pipe or simply exhausted through rearward-pointing nozzles into the by-pass duct. Although the latter might appear to be a somewhat inefficient process, the gain obtained from cycle re-matching might far outweigh a local inefficiency which applies to only a small proportion of the total engine flow.

The more radical concepts for variable cycle or 'dual mode' powerplants involve arrangements for the switching of major airflows. Such schemes have been envisaged for applications where a very large change in total powerplant airflow between one flight condition and another could be beneficial. In addition to large valve systems within the engine itself, auxiliary intakes and in some cases propelling nozzles also, are needed. A variety of configurations has been discussed in the literature, most designs having the basic aim of achieving parallel operation of fans or compressors in the higher flow mode, and series operation for normal flow. Such schemes clearly present major engineering challenges, not only in terms of the design of the engine itself, but also in regard to the intake and exhaust systems, and powerplant/airframe interactions generally. There are also dangers of considerable performance losses due to leakages and tortuous airflow paths. These numerous difficulties will not be discussed in further detail here; suffice it to say that very strong incentives are necessary for such radical design proposals to merit serious consideration.

4. SOME EXAMPLE APPLICATIONS

This Section discusses examples of potential or actual applications of cycle variability, some of which have formed the subject of performance studies at NGTE. They are classified under the general objectives of (i), increasing thrust capability; (ii) improving fuel economy; and (iii), reducing noise.

4.1 Increase of thrust capability

In some cases the mission performance of an aircraft would be significantly improved if its powerplant could provide an increased thrust at a certain flight condition. On a conventional engine, an increase in thrust rating simply implies running the engine at a higher rotational speed and turbine entry temperature, leading to shortened life of high temperature components. It is therefore worth examining the possibility of achieving increased thrust without raising turbine entry temperature, by cycle variation. Now the majority of modern aero-engines employ the by-pass or turbo-fan principle, whereby downstream of the fan the total airflow is divided between an outer fan duct and the central core engine. Thus there would seem to be some scope, at least in principle, for increasing thrust by means of a cycle change which directs a greater proportion of the total airflow through the core. This is considered below for the classes of engine used in civil transport and military combat aircraft.

4.1.1 Subsonic civil transports

The congestion which occurs on some of the busier airways in the altitude band 20,000 to 35,000 feet has led to suggestions (for example, Reference 3) that a commercial advantage would be available to aircraft having more flexible operating characteristics - in particular, a capability to cruise efficiently over a wide altitude band. For high altitude cruising, a rapid climb is needed, demanding a high thrust at the engine climb rating. In such circumstances the availability of a 'high thrust mode', to be used during climb and perhaps also for take-off, would have attractions - particularly if turbine entry temperature increases could be avoided.

The proportion of the total engine airflow which passes through the core can be raised in either of two ways:-

(a) by increasing HP compressor non-dimensional flow capacity.

(b) for a given swallowing capacity, raising HP compressor entry pressure - ie 'supercharging'.

Method (a) has been discussed recently by Hourmouziadis[4], who draws attention to the need for a significant increase in HP system rotational speed capability. As a consequence, the weight of the HP system would be increased. In addition, the flow capacity increase obtainable from over-speeding might be limited by choking of the HP compressor. The method chosen for study here is (b), which avoids these HP system problems but requires a means for producing the 'supercharging' effect. It is envisaged that this could be provided in a civil transport engine whose design, as in the case of some current engines, includes a number of core 'booster' stages mounted on the LP rotor behind the fan. By designing this booster system to be lightly loaded during normal engine operation, and incorporating variable geometry, a significant increase in work capability might be achievable for the 'high thrust mode'. The desired variation of pressure ratio between the two modes of engine operation would be aided by matching the booster stages well below their surge line for 'normal' working. Although such matching would probably result in these stages operating below their maximum efficiency, the consequent penalty in overall cycle efficiency would be small because the work done by the booster stages represents only a small proportion of the total compression process in the engine.

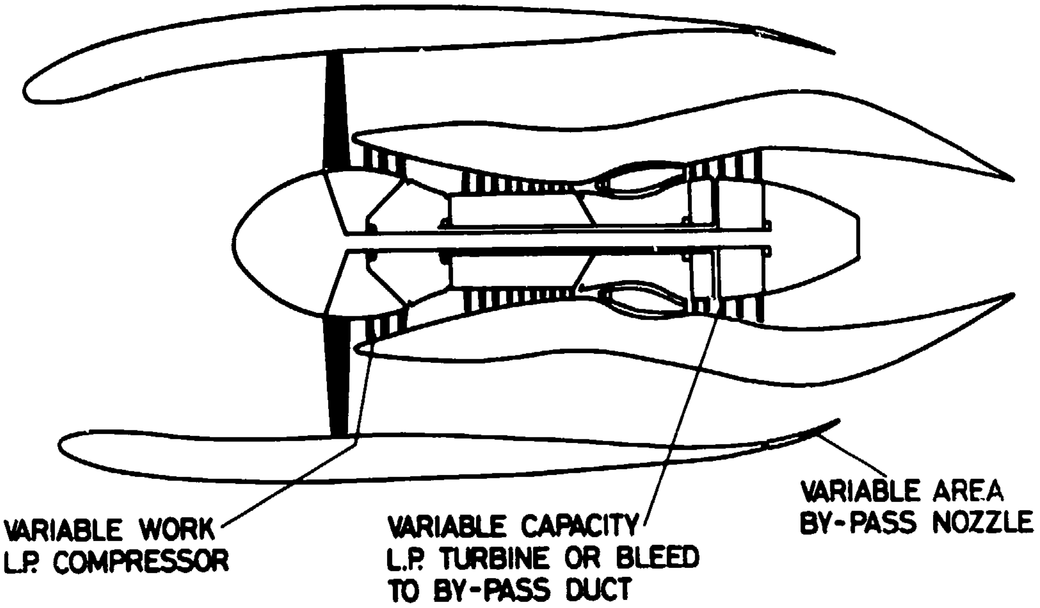

FIG.1 BY-PASS ENGINE FOR CIVIL AIRCRAFT

Figure 1 is a schematic illustration of a civil turbo-fan engine of the two-shaft type with booster compressor stages on the LP rotor. Variable cycle features to provide increased climb thrust are indicated. The LP booster compressor variables are operated in conjunction with a reduction in by-pass propelling nozzle area, forcing the cycle to rematch with reduced by-pass ratio at a given fan speed. In regard to the HP system operation, three possibilities then appear:-

(a) The whole HP system may be run at an unchanged non-dimensional operating condition, resulting in increased rotational speed and turbine entry temperature due to the increase in HP compressor entry temperature.

(b) The speed and turbine temperature increases of (a) can be avoided if the HP system is operated further down its normal working line. While this results in some loss of the HP flow increase relative to (a), there remains a net increase, and hence a thrust gain, relative to the 'normal' cycle.

(c) The HP system non-dimensional flow can be maintained at its full value, without a rise of turbine temperature, by an increase in effective LP turbine flow capacity. This effect, which lowers the HP compressor working line, can be produced either by an actual increase of turbine nozzle throat area or by a bleed of core gas flow upstream of the turbine.

Of these options for HP system operation, it is fortunate that the latter two cases, which meet the objective of achieving greater thrust without increasing turbine entry temperature, do not require core nozzle area variability - at least for the example cycle considered in this study.

Figures 2, 3 and 4 show the results of performance estimates for a two-shaft engine having the following major cycle parameters at its aerodynamic design point at Mach 0.8, 30,000 ft, ISA:-

by-pass ratio	5.5
turbine entry temperature	1500K
overall pressure ratio	27
fan by-pass section pressure ratio	1.7
fan core section plus LP booster pressure ratio	2.25

The gains in thrust available from supercharging, in conjunction with the various ways of operating the HP system discussed above, were estimated for the climb condition. These, and the associated rises in sfc are shown in Figure 2, plotted against the pressure ratio generated by the fan root section plus LP booster compressor stages. On the same basis, Figure 3 indicates the deployment required in other engine variables and Figure 4 shows some important cycle conditions. Consistent notation and styles of line are used for all three Figures.

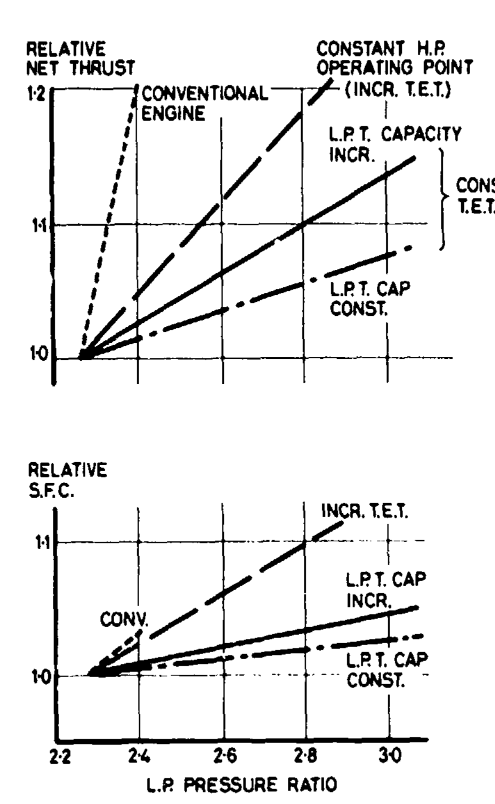

FIG.2 CIVIL ENGINE PERFORMANCE (CLIMB RATING - M 0·8, 30,000 FT)

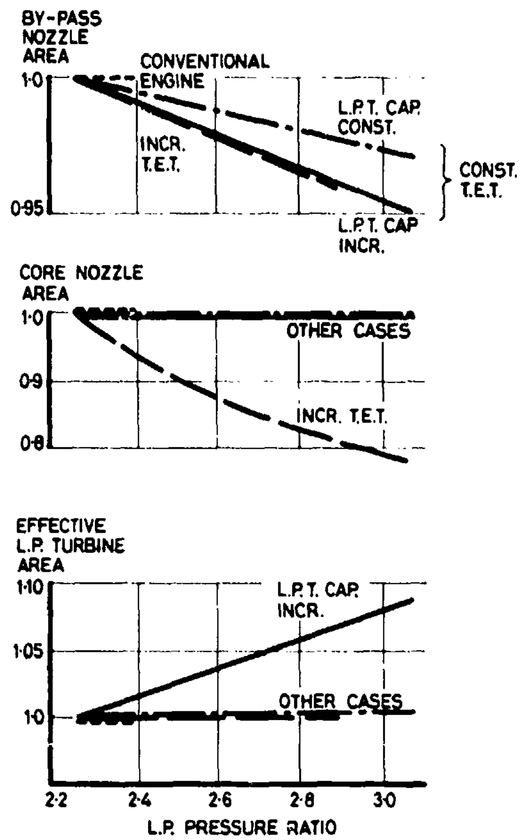

FIG.3 CIVIL ENGINE VARIABLES

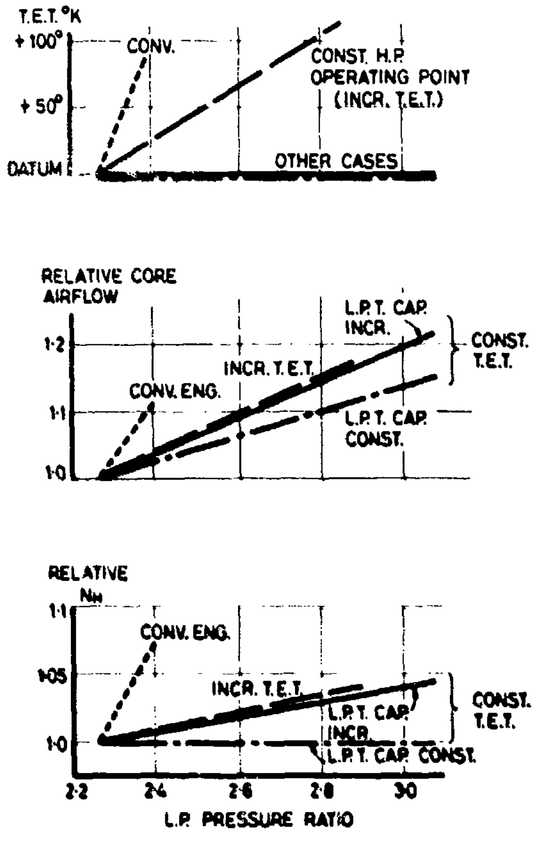

FIG.4 CIVIL ENGINE CYCLE CONDITIONS

The broken lines labelled 'conventional engine' refer to the datum case where speed is increased in a fixed geometry engine, resulting in increased flow and turbine temperature. Only a small increase in LP pressure ratio occurs, arising from the LP system moving further up its working line. By contrast, for the variable cycle cases the fan speed, total airflow and by-pass section pressure ratio remain constant. The cases labelled 'constant HP operating point', 'LPT cap const', and 'LPT cap incr' refer respectively to the alternative methods (a), (b) and (c) of operating the HP system as outlined already.

For the latter two cases, turbine entry temperature remains constant. As might be expected, increasing effective LP turbine flow capacity to maintain HP compressor non-dimensional airflow gives a significant advantage in terms of thrust for a given LP pressure ratio. Furthermore, estimates using reasonable assumptions showed little difference in thrust between a direct turbine capacity increase and bleeding an equivalent gas flow for exhaust via separate nozzles. From a cycle change involving increases in LP pressure ratio from about 2.3 to 3.1, and in effective LP turbine capacity of about 8 per cent, together with a reduction in by-pass nozzle area of about 5 per cent, a thrust gain of nearly 15 per cent is achievable with no change of turbine entry temperature. For a conventional engine, a similar thrust increase would require turbine entry temperature to rise by about 80°C. For both cases HP rotor speed would rise by around 5 per cent, while for the conventional engine there would also be some increase in fan speed. Specific fuel consumption in the high thrust mode would be slightly higher for the variable cycle case than for the conventional engine.

It may be noted that the 'constant HP operating point' alternative offers no advantage over the conventional engine except that it does not call for any increase in fan speed. For a given thrust increase, turbine entry temperatures and HP rotor speeds are very similar, and sfc is higher for the variable cycle case.

4.1.2 Military combat aircraft

The wide flight envelope of high performance military aircraft dictates the use of engines of much higher specific thrust than the turbo-fans used for modern subsonic civil transports. Although the by-pass principle is commonly used, by-pass ratios are generally around unity or lower. The by-pass and core engine flows are generally remixed in the jet pipe to feed a single propelling nozzle. Although jet pipe reheat is used to provide the large thrust boost needed for supersonic flight, there is also considerable emphasis, at some flight conditions, on achieving the greatest possible thrust with reheat unlit. A case of especial importance is that of the very low altitude 'penetration' to the target on a strike mission, requiring flight at a high speed with good fuel economy - ie with reheat unlit. The maximum dry thrust available from the powerplant tends to be the factor which limits flight speed, the problem being accentuated by high aircraft drag due to the carrying of external stores. In looking at the question of whether thrust can be increased by cycle variation, attention is again concentrated here on the technique of supercharging the core engine.

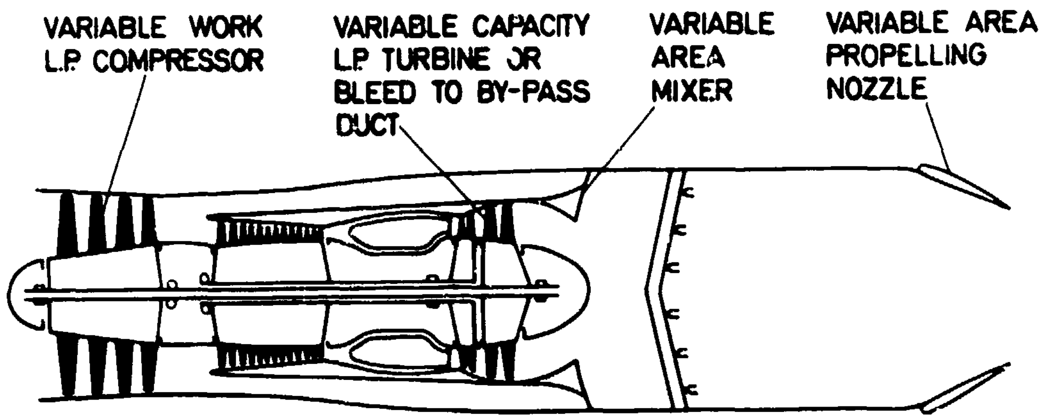

FIG. 5 BY-PASS ENGINE FOR MILITARY AIRCRAFT

Figure 5 illustrates a military two-shaft low by-pass ratio powerplant of the type just described. Variable cycle features for thrust boosting are indicated. The requirement for a common static pressure as the two streams rejoin in the jet pipe exerts a significant constraint on engine matching which tends to limit the scope available for cycle variation. Also, with a low by-pass ratio multi-stage fan it becomes unrealistic to envisage a variation of the core section pressure ratio independently of the by-pass section, particularly if, as is often the case, no LP core booster stages are used. For this study then, the increase of LP pressure ratio for the 'high thrust mode' is considered to apply to the fan as a whole, thus raising by-pass pressure as well as core entry pressure.

A representative flight condition for the phase of the mission in question is Mach 0.9, Sea-level. Figures 6, 7 and 8 show the results of performance estimates at this condition for an engine having the following cycle parameters at its sea-level, static design point:-

by-pass ratio	1.0
turbine entry temperature	1600K
overall pressure ratio	21
fan pressure ratio	2.9

The same three alternatives for operation of the HP system as for the civil engine have been studied, and a similar style adopted for the presentation of results. Again the 'conventional engine' case has been included as a datum.

Considering the same range of LP pressure ratio (though this now applies to the by-pass as well as to the core stream) the results for this military example are broadly similar to those for the civil engine. The thrust increase for the increased effective LP turbine capacity case is almost the same as for the civil engine, as also is the rise of turbine entry temperature associated with this thrust increase for the datum conventional engine. The thrust gain available without increasing effective

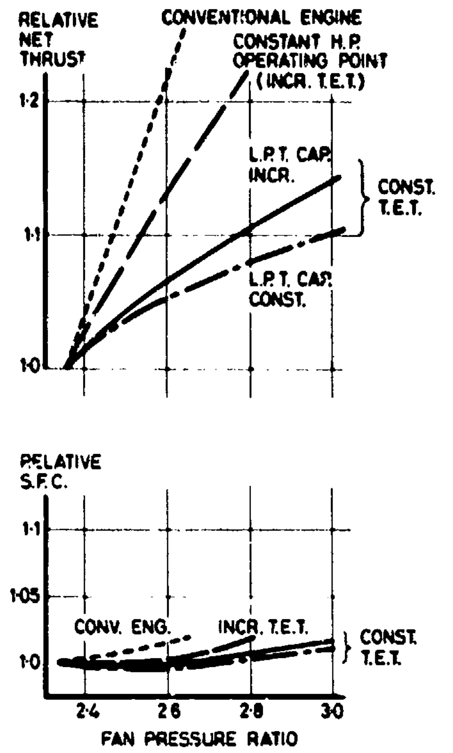

FIG.6 MILITARY ENGINE PERFORMANCE (MACH 0·9, SEA LEVEL)

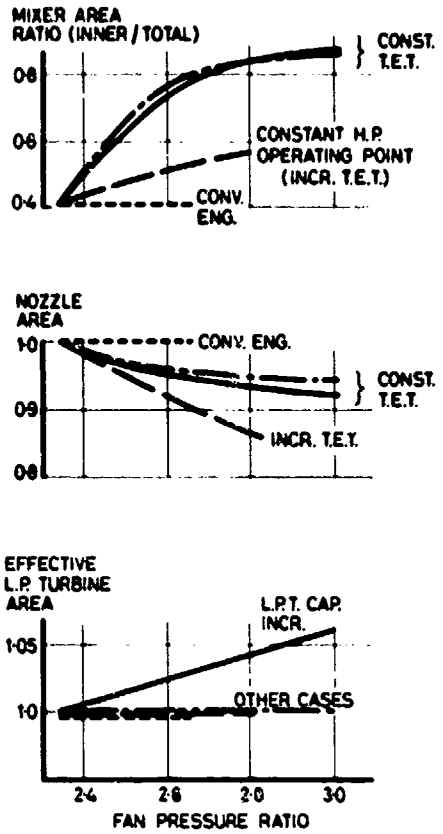

FIG.7 MILITARY ENGINE VARIABLES

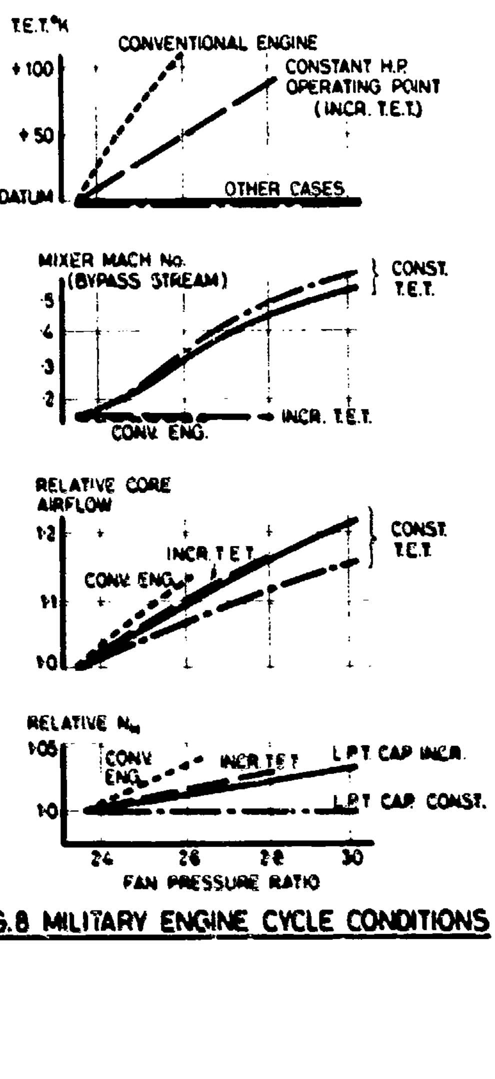

FIG.8 MILITARY ENGINE CYCLE CONDITIONS

turbine capacity is rather better than for the civil engine. Again, as for the civil engine, the 'constant HP operating point' case appears to offer little attraction relative to the conventional engine.

Mention has already been made of the added constraint associated with exhaust flow mixing in the military engine. For the constant turbine entry temperature cases, matching at the mixing plane becomes progressively more difficult with increasing thrust boost, despite the use of a mixer having variable relative flow areas. The change in core section area as a proportion of the constant total area is shown on Figure 7. At the higher thrust conditions the Mach number of the outer stream at the mixing plane is approaching 0.6 (see Figure 8), while the inner stream Mach number has fallen to about 0.16. This marked disparity would have been still greater if the design point fan pressure ratio had not been selected so as to give a Mach number for the outer stream lower than for the inner, specifically to allow for these trends. The problem would of course be alleviated if a degree of independent control of the by-pass and core section LP pressure ratios were possible, as assumed for the civil engine.

The above examples have explored the application of the core supercharging principle to increase the thrust of two very different designs of by-pass type engine, without increasing turbine entry temperature and with a relatively small rise in HP rotor speed. This basic principle is of course already used, particularly for civil turbo-fans, in the conventional development of uprated fixed geometry engines. The use of cycle variability to obtain a 'high thrust mode' is only attractive where the use of a generally uprated engine would lead to serious penalties elsewhere - for example, in cruising fuel consumption. The variable cycle engine would be significantly more complex than a conventional engine. The provision of the LP pressure ratio increase required for supercharging is clearly a key feature of the concept. As mentioned in discussion of the civil engine, this capability might be sought from a combination of compressor variable geometry - hopefully confined to the stator blading - and matching the stages in question low on their operating characteristics for 'normal' operation. In regard to the increase in effective LP turbine flow capacity which is also needed if the maximum potential from core supercharging is to be realised, it is noteworthy that these studies suggest that a bleed upstream of the LP turbine may be virtually as effective in thrust terms, and possibly considerably simpler, than varying the capacity of the turbine itself.

Finally, mention should be made of the possibility of increasing the achievable thrust boost by combining the core supercharging concept with an increase in total engine airflow. In principle, the variable features discussed above could be used to rematch the cycle at higher LP speed, as well as increased pressure ratio, to give a greater fan airflow while still maintaining turbine entry temperature constant. The increased speed capability would in general require some increase of LP rotor weight, and its effectiveness in increasing flow might be limited at high non-dimensional speeds by fan choking. Because of this latter consideration the more fruitful application would probably be to the military engine case at Mach 0.9, sea-level, where non-dimensional speed is reduced below its take-off value due to the elevated intake total temperature.

4.2 Improving fuel economy

The combat type aircraft discussed in the previous Section also provides a good example of a flight condition where the prospects for improving fuel economy by cycle variation invite study. This is a return cruise to base at low altitude. The aircraft is now without external stores and at light weight. Flying at a modest speed in the region of Mach 0.6, drag is low and the powerplant is operating at only a fraction of its maximum dry thrust. The specific fuel consumption of the engine is therefore well above its minimum value, on a curve which rises rapidly as thrust is reduced. Thus, the lower the aircraft drag level, the higher will be the sfc. For instance, if the aircraft has variable-sweep wings, the advantage of using minimum sweep at this flight condition to reduce lift-dependent drag will be partially offset by an increase in engine sfc. A further factor which accentuates the problem is that powerplant installation drag is high at low thrust, due to the low engine airflow demand.

Figure 9 shows engine performance at Mach 0.6, sea-level. The thrust requirement might be in the region of 30 per cent of the maximum dry value. At this thrust level, the sfc of the basic engine is about 15 per cent above its minimum value. The reasons for this situation are clear from the other curves on the Figure. Although propulsive efficiency improves with thrust reduction, the benefits of this are outweighed at low thrust by the progressive decrease of thermal efficiency resulting from degradation in the major cycle parameters of overall pressure ratio and turbine entry temperature. For example, at 30 per cent thrust the pressure ratio, which has a powerful influence on cycle efficiency, has fallen to around half its full-power value. The powerplant is clearly not well matched to the propulsion requirement at this flight condition, and it would appear that there might be scope for improvement by cycle variation.

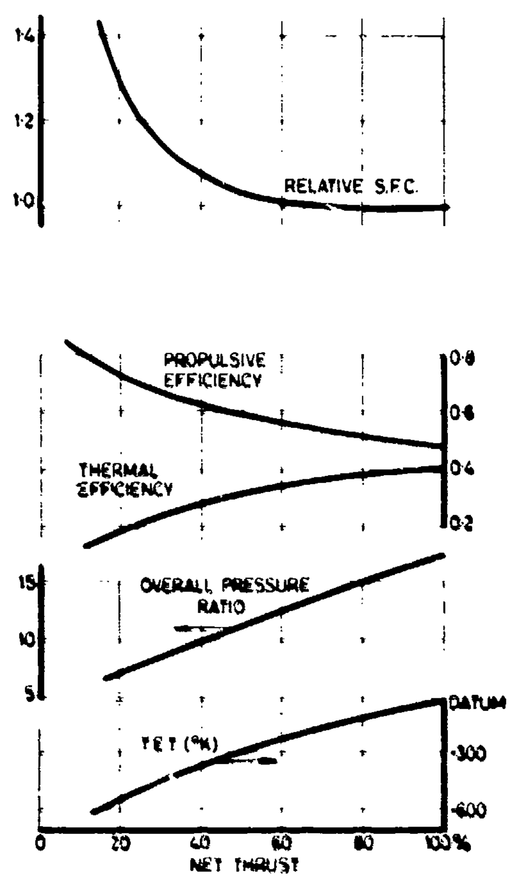

FIG.9 MILITARY ENGINE PERFORMANCE (M·0·6, SEA LEVEL)

Figure 10 provides an indication of the results of a preliminary exploration of such prospects. The sfc versus thrust curve for the basic conventional engine is shown again, as the datum. The broken curve which branches from that for the basic engine at about 45 per cent thrust shows the order of improvement in sfc which is obtainable from a simple form of variable geometry, namely closure of the propelling nozzle until its throat area is about 70 per cent of the reheat unlit design point value. The effect of this is to increase specific thrust and raise thermal efficiency to a greater extent than the associated fall of propulsive efficiency. At the thrust level of interest, engine sfc is improved by about 5 per cent. Unfortunately, this gain will be partially offset by an increase of installation drag due to the reduced engine airflow. While the magnitude of this drag penalty is very dependent on the aircraft intake and afterbody design, it could cut the benefit in fuel economy to about half the basic engine sfc improvement. The gains in fuel economy from this technique are therefore small, although quite readily available by arranging for the variable propelling nozzle to be capable of closure below its design point reheat unlit area.

The variable nozzle may also offer scope for fuel economy gains at some flight conditions which require higher cruise thrust levels, when opening the nozzle may be profitable. An advantage of such cases is that installation drag is now reduced, allowing a double benefit if engine sfc also improves. It should be emphasised that improvements in fuel economy by nozzle variation in dry engine

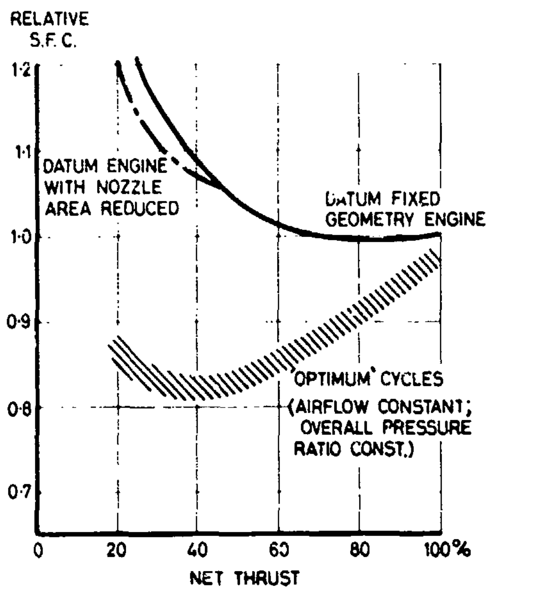

FIG.10 SCOPE FOR S.F.C. REDUCTION
(MILITARY ENGINE M=0·6, SEA LEVEL)

operation are sensitive to thrust level, basic engine cycle and the detailed performance characteristics of the engine components - for example, the distribution of efficiency contours on the fan characteristic. Detailed study of individual cases is thus particularly necessary. The attraction here is that although the available gains are generally small, they can be had essentially by arranging for the more refined control of an existing powerplant component, namely the variable nozzle which is necessary for reheat operation.

Moving to the other extreme in terms of complexity, the lower, shaded band gives a broad indication of what might be obtainable within the constraints of the same carcase size as the basic engine, and the same overall pressure ratio, but given otherwise complete freedom in varying the cycle. Thus the variables for these estimates were by-pass ratio, turbine entry temperature, LP and HP compressor pressure ratios (with the proviso of a constant value for their product), and all swallowing capacities except that of the LP compressor, which was maintained constant at the design point value for the datum engine. Component efficiency values were also assumed equal to the design point values for the datum engine.

For the low thrust case, very large sfc improvements are theoretically available, amounting to more than 30 per cent before taking any credit for reduced installation drag. However, implied in such improvements are drastic cycle changes - for example, an increase of by-pass ratio to more than 10! This seems well beyond the bounds of feasibility at reasonable levels of complexity and cost. To attempt to approach such a degree of cycle variation would result in the whole engine design being dominated by the variable cycle requirement, with enormous technical risks. More realistically, the challenge here is to devise practicable designs which yield a worthwhile performance level intermediate between this extreme example and the small gains available from the nozzle area trimming discussed above. Preliminary studies suggest, however, that the difficulty of finding a cost-effective solution in this area may be considerable, unless the additional component variability required could also be used to good effect in other parts of the aircraft flight envelope.

4.3 Noise reduction

It has been noted earlier that cycle variation aimed at reducing noise can take a variety of directions, depending on the balance of the engine noise sources. For engines having a high jet velocity, a valuable technique for noise reduction at reduced thrust conditions is to simply open the propelling nozzle. This results in a reduction of turbine entry temperature and jet velocity, the required thrust then being achieved at higher LP rotor speed and airflow, and lower turbine entry temperature and jet velocity. A notable application is to the Olympus 593 engine of Concorde, for noise-abatement climb and approach to landing. At the latter condition, an increase in propelling nozzle throat area of about 30 per cent gives a noise reduction of about 6 PNdB (Reference 5).

For high by-pass ratio engines where fan or LP turbine noise may be the significant problem at low thrust conditions, some improvement might be achievable by reducing the core nozzle area. This has the effect of reducing the LP rotor speed and increasing the core jet velocity; the balance of noise sources is therefore changed. This technique was discussed by Wilde and Pickerell[6] in connection with design studies of three-shaft civil turbo-fan engines, a possible noise reduction at approach of about 3 PNdB being estimated.

Far more radical schemes for cycle variation are envisaged by some project engineers as a potential solution to the noise problem of future supersonic transport aircraft (SST's). The design requirement for long range supersonic cruise leads to strong emphasis on minimising powerplant installation drag and weight. Furthermore, the high aircraft velocity at supersonic cruise necessitates an engine of essentially high jet velocity. Current SST's (Concorde and TU144) therefore make more noise in the vicinity of airports than modern subsonic aircraft fitted with high by-pass ratio turbofans. However, future SST designs will need to show significantly improved noise characteristics. Unless a new and highly effective technique of jet noise suppression is established, this will entail a reduction in jet velocity at the take off condition.

Figure 11 gives a simplified indication of the variation of jet noise with velocity. Due to the shape of this curve, a considerable change of engine design will be necessary to gain a substantial noise reduction. Allowing for some improvement in aircraft climb characteristics, and taking into account the important subjective effects of noise spectral content and duration, it has been estimated that the achievement of noise levels appropriate to a future SST will require at least a 50 per cent airflow increase at a given thrust. To achieve this by simply fitting larger engines and their associated

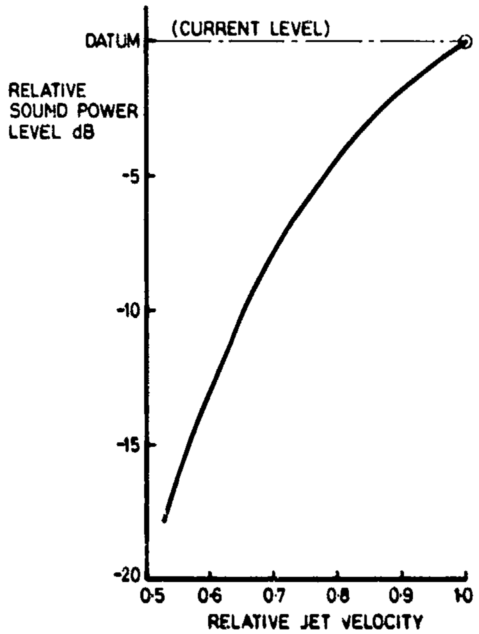

FIG.11 VARIATION OF JET NOISE WITH VELOCITY

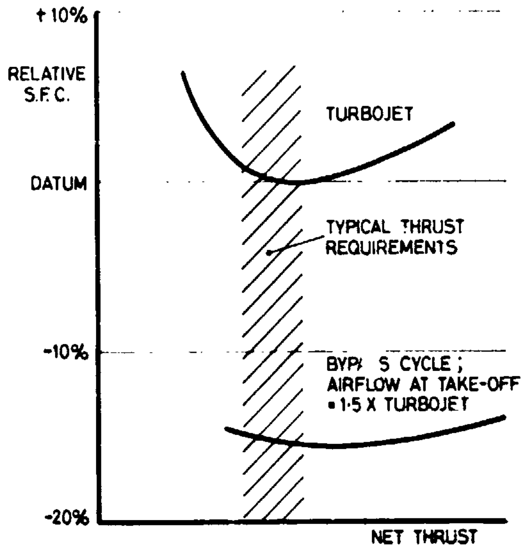

FIG.12 S.S.T. SUBSONIC ENGINE PERFORMANCE

intake systems could give rise to a substantial penalty in terms of weight and supersonic cruise drag. Variable cycle schemes involving large increases of powerplant airflow at take-off and landing conditions have therefore been widely studied. While such schemes would add considerably to powerplant cost and weight, scope does exist for compensating for these disadvantages by virtue of the better fuel economy characteristics potentially available from a higher flow powerplant during subsonic phases of the flight.

An SST needs to carry a very substantial fuel load for subsonic flight, covering departure from the airport area, climb to height and possible subsonic cruise over densely populated areas, as well as diversion and holding conditions. This can amount to approximately double the passenger payload. Therefore an improvement in subsonic fuel economy can be very significant. Figure 12 compares subsonic sfc levels of a turbo-jet sized for supersonic cruise requirements, and a by-pass cycle having 50 per cent higher airflow than the turbo-jet at the take-off condition, thrust levels being related to the needs of an aircraft of slender delta form. If most of the potential improvement shown could be obtained in practice during the 'high flow' mode of operation of a variable cycle powerplant, the saving in subsonic fuel would be an important factor to set against the added weight of the powerplant. Thus it might be possible to produce an SST having a payload fraction similar or superior to current aircraft of this class but with much better noise characteristics. This having been said, the problems of implementing a radical variable cycle powerplant concept for an SST should also be emphasized. Development to meet the stringent civil safety requirements at minimum weight, together with high performance and good in-service maintenance characteristics, would be a large task indeed.

5. IN CONCLUSION

The control of the powerplant cycle with the object of gaining performance improvements has been a subject of study by propulsion engineers over many years. More recently, reduction of noise has entered the scene as an additional objective. However, although some significant examples of cycle variation have appeared in production engines, these have generally been of a relatively straightforward nature, requiring little in the way of additional complexity or novelty. Indeed, most aeronautical applications have consisted of the exploitation of a variable propelling nozzle. Schemes involving substantial increase of development effort and risk have not so far been implemented.

What, then, are the prospects for the future? The launching costs of new aeronautical projects have now reached high levels in relation to available resources, even for collaborative ventures between major partners. Proposals for novel design features which imply increased risk and development cost will therefore inevitably attract highly critical scrutiny. Valuable benefits relative to alternative solutions will need to be in prospect, together with a good level of confidence in the success of the proposed development.

Such confidence must depend largely upon the availability of a basis of practical experience of the design features in question. This was acquired quickly for variable propelling nozzles and variable compressor stators, and led to their widespread application. As mentioned earlier, experience of an encouraging nature is now accumulating on turbines with variable nozzle guide vanes - much of it in the non-aeronautical gas turbine field. Some experience is also being gained, on both sides of the Atlantic, in the use of variable rotor blading for single-stage fans. Bleed valve systems are already in quite common use in the lower temperature parts of engines, both for compressor surge margin improvement and for supplying substantial quantities of compressed air for aircraft services such as flap blowing. This again forms a useful fund of experience, though the more extreme variable cycle concepts would require far-reaching further development. Regarding the control of variable cycle powerplants, the sophisticated capability required could be provided by digital systems which are now finding their way into the aircraft propulsion world.

For the future, variable cycle schemes will continue to face strong competition from alternative solutions. The more radical schemes will not easily find acceptance, especially where their development would require a significant diversion of resources from the mainstream of powerplant evolution. At the same time, we may expect the growth of component experience, discussed above, to produce a climate where variable cycle principles will be used more widely in well-chosen applications. Certainly there will continue to be scope for ingenuity in thinking about the design of powerplants to meet the varied propulsion requirements posed by modern aviation.

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ACKNOWLEDGMENT

The authors wish to record their appreciation of the assistance given during the preparation of this paper by colleagues at NGTE.

DISCUSSION

N.F.Rekos

You did raise a question at the very end. You expressed a great deal of concern regarding, let us call them the "radical concepts" and let's say, general reluctance of getting involved into any such development. These radical concepts will probably be discussed later. Do you expect to have or do you expect to enlarge your capabilities in your work at NGTE to include some of these so-called radical cycles?

Author's Reply

Several project studies aimed at establishing practicable variable cycle powerplant configurations have been undertaken at NGTE over the last decade or so, and the Establishment will naturally continue with such studies. However, regarding the more radical variable cycle concepts, a very careful evaluation of the benefits and penalties, in the light of particular project requirements, must precede any decision to embark on major experimental programmes which divert resources away from the mainstream of powerplant evolution.

My own view is that we will tend to adopt a more evolutionary approach than is currently advocated in some quarters, and that we shall concentrate on developing the techniques which enable us to progressively introduce greater degrees of variability in well chosen applications.

Mr Swan has proposed some dramatic changes to the power plant which would necessarily involve tremendous technical risks. However, there is much in his paper with which I would agree. In his discussion of the 'High Throttle Ratio' concept he has shown that substantial benefits can result from careful selection of the engine design condition. I believe that, by adopting a highly integrated approach to the design of engine/airframe combinations for particular mission requirements, it will be possible to avoid the complexity, and inherent risks, associated with some of the more radical variable cycle concepts.

N.F.Rekos

I was wondering where some of your work could be directed to.

I know we are in the United States concentrating a great deal on variable area turbines variable geometry nozzle valoring arrangements for military and civil applications. I think later on you are going to hear a paper on propellers. You will find from this paper this has applicability from both military and civil applications. You also will hear a paper relating to cycles to optimize fuel consumption. We are highly concerned of saving our fuel resources. The feeling is that one must consider these things even though it is a great rise. The alternative is performing limited operations with conventional vehicles and possibly using too much fuel.

Author's Reply

I have been expressing some personal reservations concerning the prospects for complex multi-mode variable-cycle powerplants. It was not my intention to give the impression that I consider work along the lines you have just mentioned to be unrealistic. To the contrary, much of our effort is directed along similar paths.

Regarding the variable pitch propeller I agree that this deserves very serious consideration.

E.Willis

Your presentation seems to emphasize an evolutionary or incremental approach to VCE development. Do you believe that *major* progress will result from this approach? Do you not feel that efforts should also be applied to some of the more "radical" ideas?

Author's Reply

Clearly the case for radical VCE schemes needs to be investigated, and I am not suggesting that such work should not be undertaken, indeed it should. What I am doing is expressing some scepticism about the liklihood of such schemes being adopted as a result of a step change in technology. My belief is that a more evolutionary approach will probably be pursued for the reasons discussed in the paper.

E.Willis

It's kind of a circular process though because without the examination and demonstration of the critical parts the scepticism is always going to remain.

Author's Reply

Yes, but in mounting a major exercise at this stage one would risk wasting a large portion of the available research and development resources.

R.J.Latimer

Have you used any change in turbine efficiency in your cycle calculations?

Author's Reply

Not for the studies reported in the paper. A number of relatively lightly loaded uncooled variable turbines have been run with, it has been claimed, very small penalties in efficiency. Indeed, we have some experience at NGTE with such turbines. However, on cooled turbines for aeronautical applications the problems of maintaining efficiency are very great, but we have assumed constant efficiencies in our calculations, at levels typical of fixed geometry machines.

PARAMETRES D'OPTIMISATION DES MOTEURS EN FONCTION DE LA MISSION

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RESUME

On examine les conditions dans lesquelles se détermine le choix du cycle pour un moteur militaire . Des méthodes élaborées d'optimisation existent mais les auteurs essaient d'éclaircir le problème par une réflexion directe. Dans le cas des avions de combat, les contraintes imposées au choix sont passées en revue et l'intérêt de la géométrie variable apprécié .

L'influence des hypothèses définissant la mission est très grande . Si dans certains cas l. .hoix des caractéristiques fait l'objet d'un compromis assez facile, dans d'autres cas des contradictions apparaissent.

L'étude et la mise au point de nouveaux moteurs doivent être abordées en tenant compte d'un vaste ensemble de paramètres dont certains ne sont ni techniques ni quantifiables.

INTRODUCTION

La caractéristique des êtres vivants est d'avoir atteint un degré de perfectionnement admirable qui leur permet de se reproduire en s'adaptant toujours mieux aux conditions de leur vie. Sans devenir capable de maitriser l'alchimie des chromosomes l'homme a su sélectionner patiemment des variétés optimisées pour ses besoins et disposer de vaches qui ne donnent que du lait, comme de boeufs qui ne donnent que du filet, c'est à dire exactement ce qu'on leur demande.

Quelle surprise de constater que lorsqu'il s'agissait de fabriquer de toutes pièces des mécanismes, les ingénieurs, qui n'ont rien à rendre aux éleveurs, ne soient pas parvenus à définir des modèles qui recueillent l'unanimité des faveurs ! Nul n'ignore cependant que la naissance d'un nouveau moteur d'avion est désormais entourée de nombreux cercles de têtes pensantes, qu'elle est le fruit d'un long processus de décision, dans lequel apparaît bien souvent le maître-mot "optimisation".

Essayer de comprendre à quelles difficultés peut se heurter cette démarche a priori puissante et efficace , comment l'effort d'optimisation peut être mis en défaut, quels espoirs peut faire naître la nouvelle race des moteurs à géométrie variable, tel est ici notre propos, limité au cas particulier des avions de combat, sans toutefois exclure la référence à d'autres types d'utilisation.

Un programme de moteur d'avion d'armes ou de transport commercial est devenu une affaire d'Etat, qui engage des sommes très importantes, des efforts de longue durée, l'avenir de nombreuses personnes . C'est pourquoi toute décision doit s'appuyer sur une analyse approfondie pour laquelle on requiert les moyens les plus modernes.

Malheureusement il y a eu des cas où on a découvert trop tard que les ordinateurs ne pouvaient restituer que ce qu'on leur avait fourni, et des avions qu'on croyait parfaitement optimisés n'ont eu aucun succès. Sans doute parce que les critères d'optimisation retenus n'étaient pas bons. Nous pensons que l'avion (avec son moteur) optimal est celui qui remplit bien la mission qui lui est confiée . Mais cette mission est bien difficile à définir et à exprimer en termes quantifiables assimilables par une machine. On pourrait dire par exemple que la mission d'un avion commercial est de rapporter le gain maximal à son opérateur . Ce serait oublier qu'il doit d'abord se faire accepter par les habitants de cette planète en pleine crise de conscience technologique. De même la mission d'un avion .ilitaire serait de donner la victoire si l'on oubliait la recherche sincère et répandue d'une paix véritable.

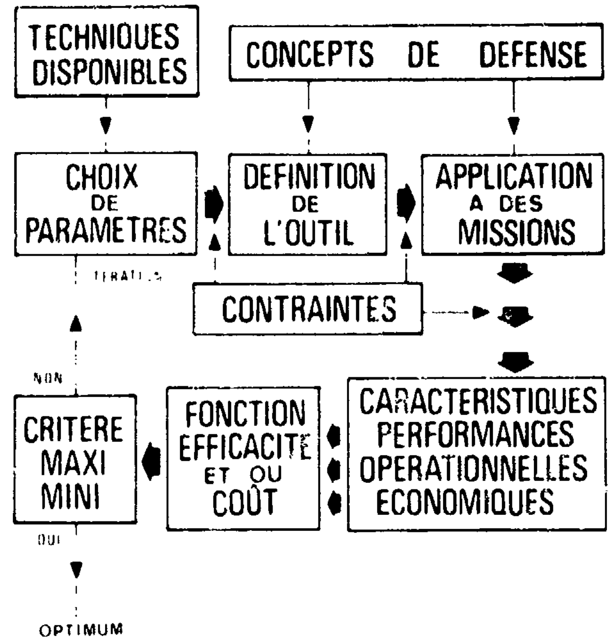
Il y a donc peu d'espoir que la réunion dans un vaste programme d'ordinateur des équations de la mécanique, de la thermodynamique, de l'aérodynamique, des meilleures méthodes de calcul des structures et des écoulements, des corrélations les plus récentes sur les tendances technologiques, des statistiques sur les coûts et délais,etc....puisse donner un résultat valable si quelques"coups de pouce" ne sont pas donnés au bon moment.

Notre exposé ne prétend pas résoudre le problème ni même sans doute le poser complètement. Dans un premier temps, nous allons essayer de voir dans quelles conditions peut se dérouler une optimisation formelle faisant appel à des données quantitatives et techniques. Ensuite nous examinerons d'autres éléments qui, pour être moins techniques et surtout moins quantifiables n'en sont pas moins importants.

OPTIMISATION FORMELLE

Le processus d'optimisation comprend normalement les phases suivantes : (voir schéma ci-contre)

- définition de la mission à accomplir par l'outil à réaliser
- établissement de critères quantifiés mesurant la valeur du travail effectivement accompli,
- recensement des paramètres régissant les caractéristiques de l'outil produit et des contraintes subies ,
- calcul déterminant la combinaison des paramètres donnant la meilleure valeur au travail accompli au cours d'une mission.



Prenons un exemple :

Mission	:	virage demi-tour d'un avion de combat connu,
Outil	:	trajectoire suivie,
Critère	:	temps pour tourner (à altitude maintenue),
Paramètres	:	angles d'incidences et d'inclinaison , poussée ,
Contraintes	:	limites de la structure, du pilote, du moteur, de l'aérodynamique, etc...
Calcul	:	voir par exemple la référence 1.

Le calcul explicite est possible parce que :

- la mission est clairement et simplement définie,
- le critère est bien calculable et appréciable,
- l'influence de chaque paramètre est déterminée.

Essayons d'appliquer ces notions aux avions militaires et à leurs moteurs .

Issue de réflexions qui prennent leur source dans la politique de défense, la configuration générale de l'avion impose certaines contraintes au moteur et la mission à accomplir est grossièrement définie. C'est ce que schématise le tableau de la figure 1. Ce n'est que très lentement, à mesure des découvertes de la recherche et des réussites technologiques, que les concepteurs imaginent des systèmes ayant des configurations très nouvelles.

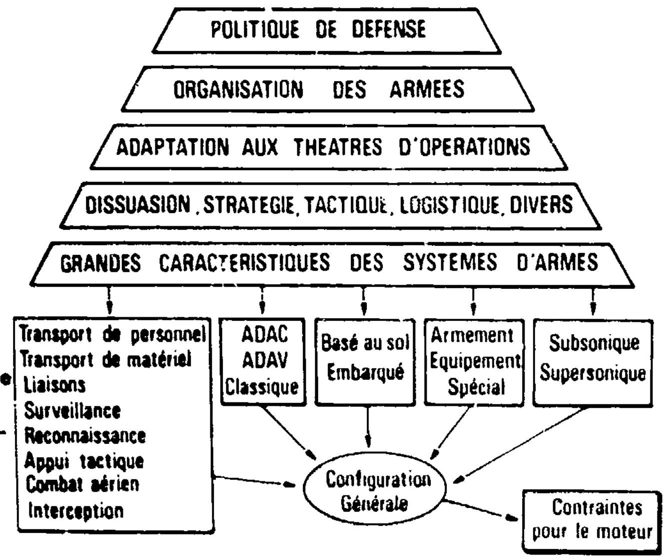


Fig. 1 - Définition générale de l'avion

Au stade actuel et comme le montre le diagramme de la figure 2 , l'analyse des missions conduit à la conception de systèmes plus ou moins spécialisés. S'ils sont très spécialisés, il est plus facile de les définir et les missiles concurrencent facilement les avions. Si la mission comprend une grande part de souplesse et donc d'intelligence active, la présence de l'homme devient indispensable : il s'agit d'un avion peu spécialisé mais bien difficile à définir. Avec plus ou moins de conventions et dans des conditions que nous examinerons plus loin, on arrive généralement au concept d'un avion ou de plusieurs avions remplissant les fonctions nécessaires et qu'il convient de réaliser suivant un certain calendrier et dans un certain contexte industriel.

Un processus d'optimisation peut s'engager alors qui devrait attribuer une valeur, positive ou "efficacité", négative ou"coût,"à chacune des caractéristiques résultant de la conception du moteur définie par un choix de paramètres dans le cadre de la technologie disponible. Mais cette valeur ne peut se calculer que dans le cadre d'une mission précisée. Il s'agit en effet de pouvoir formuler une "fonction d'efficacité globale" dont on essaiera d'atteindre le maximum.

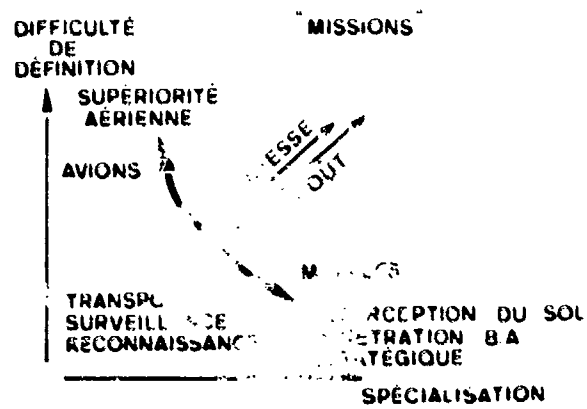


Fig. 2 - Première analyse des missions

Les principaux paramètres disponibles sont rappelés en figure 3 . Nous nous bornerons ici à examiner le cas des turbomachines faisant appel à des cycles de type connu . Ces cycles sont finalement définis par un nombre assez restreint de caractéristiques :

Le nombre de flux est limité par l'ingéniosité et rencontre évidemment des problèmes de complexité et donc de poids. Leurs importances relatives peuvent, sous les mêmes réserves, varier largement, entre la valeur très grande attribuable à l'hélice et la nullité des plus simples turboréacteurs. Les rapports de pression caractérisant les cycles des divers flux peuvent être choisis dans une large gamme mais la marge est étroite entre les températures acceptables par le thermodynamicien et celles supportées par les matériaux. Enfin la réchauffe après turbine est un vieil exemple de cycle variable pratiquement par tout ou rien.

Des travaux constants de recherches et de mise au point, dans les domaines théoriques et technologiques ont permis depuis les premiers modèles à géométrie fixe de se libérer progressivement de multiples contraintes, et l'avenir peut apporter beaucoup plus. Le caractère de simplicité attaché à la géométrie fixe a certes été nécessaire pour les premières réalisations, mais on le retrouve toujours grâce aux qualités inhérentes en matière de coût, de facilité d'entretien , de robustesse, qu'il s'agisse d'avions-écoles ou d'appareils consommables, lorsque les contraintes rappelées ci-dessous ne sont pas insupportables.

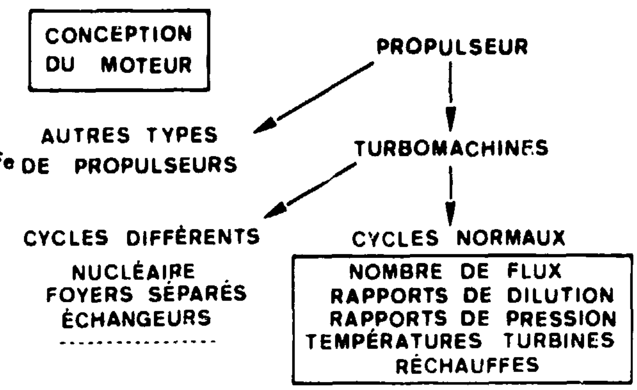


Fig. 3 - Principaux paramètres disponibles

Contraintes en géométrie complètement fixe

- Nombre de flux toujours le même,
- Compresseur et turbine : caractéristiques dépendant du nombre d'étages qui varie de façon discrète ,
- Compresseur et turbine tournent à la même vitesse, en permanence,
- Le rapport de dilution varie naturellement en fonction du régime de vol,
- Le rendement de l'entrée d'air varie, la distorsion peut être trop forte,
- Pertes incontrôlées par jeux, refroidissements, décharges,
- Régime de combustion se désadaptant ,
- Pas de réchauffe possible.

Il est bien clair que ce stade est dépassé et que les avions élaborés qui assurent le transport des passagers dans des conditions remarquables, ou qui figurent dans les systèmes de défense les plus puissants tirent parti de véritables progrès dont les principaux sont rappelés ici :

Au stade actuel des améliorations

Portes de décharges interflux :	Rapport de dilution et nombre de flux varient d'une façon limitée
Stators de compresseurs à calage variable :	Déformation du champ du compresseur (transitoires, démarrage ...)
Entrée d'air à géométrie variable :	Le rendement varie peu, la distorsion est réduite
Tuyères à section variable :	Le point de fonctionnement est mieux contrôlé ; les transitoires plus rapides ; la réchauffe est possible
Soufflante à calage variable :	Adaptation meilleure à la vitesse , transitoires améliorés

Quant à l'avenir, pour autant qu'il ne soit pas bouleversé par des crises difficilement prévisibles, on peut penser qu'il verra se développer des perfectionnements ou des nouveautés dont les plus attrayantes paraissent être les suivantes :

Dans l'avenir

Vannes de direction des flux :	- cycles serie-parallèle - inversion jet chaud -jet froid (poussée basse vitesse , bruit)
Section variable des distributeurs de turbine :	Ligne de fonctionnement optimisée , débit constant
Calage variable :	Rendement amélioré
-Cambrure des aubes, rotors à calage variable : -Transmission à vitesse variable -Étages débrayables	Optimisation permanente : marge au pompage , rendement du compresseur, rapport de pression , débit du moteur

Certains des dispositifs de géométrie variable ont été utilisés en fait depuis assez longtemps sur les turboréacteurs . Par exemple :

- Les stators de compresseurs à calage variable

  Initialement il s'agissait d'améliorer la marge au pompage du compresseur aux très bas régimes, mais, grâce au développement des systèmes de régulation perfectionnés, ils peuvent être maintenant utilisés pour améliorer le rendement ou le débit du compresseur dans tout le domaine, ou pour combattre en toutes circonstances les effets de la distorsion de l'écoulement dans l'entrée d'air .

- Les vannes de décharge étaient à l'origine de simples échappements d'air perdu destinés à permettre les évolutions aux faibles vitesses de rotation ; désormais il s'agit de larges orifices permettant une redistribution des flux dans un assez large domaine de fonctionnement.

- Les tuyères à section variable ont été indispensables pour permettre l'accroissement de débit réduit produit par la réchauffe des gaz. Cette réchauffe est d'ailleurs un véritable changement de cycle thermodynamique accompagné d'un changement de la géométrie. Mais leur emploi s'étend désormais à la modification de l'adaptation des compresseurs sur le moteur sans réchauffe ; on peut ainsi diminuer certaines pertes d'interaction.

Cet examen rapide et non exhaustif permet néanmoins de classer les dispositifs en trois catégories :

Premier niveau : éléments améliorant localement l'efficacité ou le fonctionnement d'un composant du moteur sans répercussion notable sur le cycle : ces éléments aujourd'hui pratiquement limités aux aubages directeurs de compresseurs devraient apparaître prochainement sur les turbines et sur la chambre de combustion. Leur action sera de maintenir optimal le rendement du composant concerné sur la plus grande plage possible d'utilisation.

C'est ainsi qu'on peut penser s'attaquer au problème des performances à régime réduit. Un dispositif particulièrement intéressant serait celui qui maintiendrait à tout instant à leur niveau minimal les écoulements parasites ou secondaires générateurs de pertes comme ceux dus aux jeux en bout d'aubes ou aux circuits de refroidissement.

Deuxième niveau : il s'agit de faire varier l'adaptation des divers composants du moteur entre eux afin d'optimiser la configuration pour chaque cas de vol . Les variations de cycles qui en résultent restent limitées puisqu'il n'y a pas de changement de configuration. Le seul élément aujourd'hui utilisé, la tuyère à section variable, ne suffit pas pour réaliser cette optimisation : il convient de rendre variables les autres sections critiques du moteur. Le développement de cette géométrie variable passe donc par la mise au point du distributeur de turbine à section variable.

Troisième niveau : c'est celle qui provoque une modification complète du cycle par un changement de configuration. La réchauffe ou post-combustion entre dans cette catégorie. Parmi les systèmes en étude la diversion des flux parait promise à un bon avenir grâce aux possibilités qu'elle offre de résoudre les difficultés du transport civil supersonique . Qu'en est-il pour les moteurs militaires ?

Intérêt pour les moteurs militaires

Le premier niveau de géométrie variable s'inscrit directement dans le cadre de l'amélioration des composants du turboréacteur. Elle intéresse donc tous les types de moteurs et ne peut donc être considérée comme d'un intérêt spécifique au domaine militaire.

Le troisième niveau peut paraître séduisant pour la propulsion d'un avion militaire polyvalent. On sait en effet que le meilleur cycle pour les missions de pénétration, de convoyage ou comportant une part d'attente importante correspond à un taux de dilution élevé pour un rapport de pression élevé ; pour des missions de combat tournoyant, un taux de dilution moyen et un rapport de pression élevé associés à une post-combustion pilotable conviennent mieux ; enfin,pour les missions d'interception en haut supersonique, le meilleur cycle correspond à un taux de dilution et un rapport de pression faibles avec post-combustion.

Un moteur capable de réaliser tous ces cycles pourrait paraître idéal, mais outre qu'il paraît douteux que l'on parvienne jamais à une telle "souplesse", un tel moteur serait très certainement lourd et surtout encombrant, ce qui se traduirait pour un avion fortement motorisé par des pénalités et traînées qui contrebalanceraient les bénéfices de la formule.

Le niveau intermédiaire semble plus prometteur pour les moteurs militaires. Si l'on examine le domaine de vol d'un moteur supersonique (figure 4) on s'aperçoit que la performance maximale du moteur est limitée par des facteurs mécaniques ou thermiques différents selon les cas de vol. Il s'ensuit que la poussée du moteur ne correspond à l'optimum de la formule que dans une petite partie du domaine de vol . Une adaptation variable devrait permettre de déplacer ces limitations de façon à autoriser le fonctionnement à l'optimum du paramètre intéressant (pressions pour la poussée, rendements pour la consommation , etc.....).

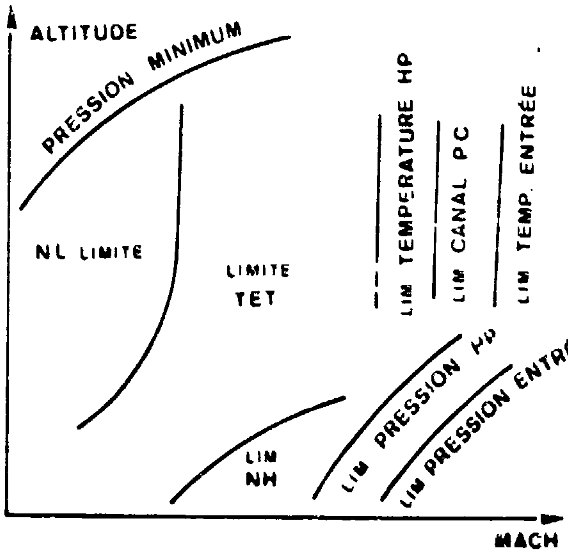


Fig.4- Domaine et limitations du moteur

LE MOTEUR ET L'AVION

Dans le processus d'optimisation dont nous suivons le déroulement se place maintenant une étape décisive , celle de l'intégration du moteur à l'avion. En effet le moteur n'est qu'une partie du système et même du sous-système qu'est encore l'avion . L'optimisation doit donc porter sur la totalité du système. Mais, pour l'instant, considérons seulement l'avion et ses qualités intrinsèques . En associant une formule de moteur à une formule de cellule, il est possible, au prix de calculs d'avant-projet dont les références 2 et 3 publiées par l'AGARD, donnent les grands principes, de faire apparaître les qualités caractéristiques rassemblées dans le tableau de la figure 5.

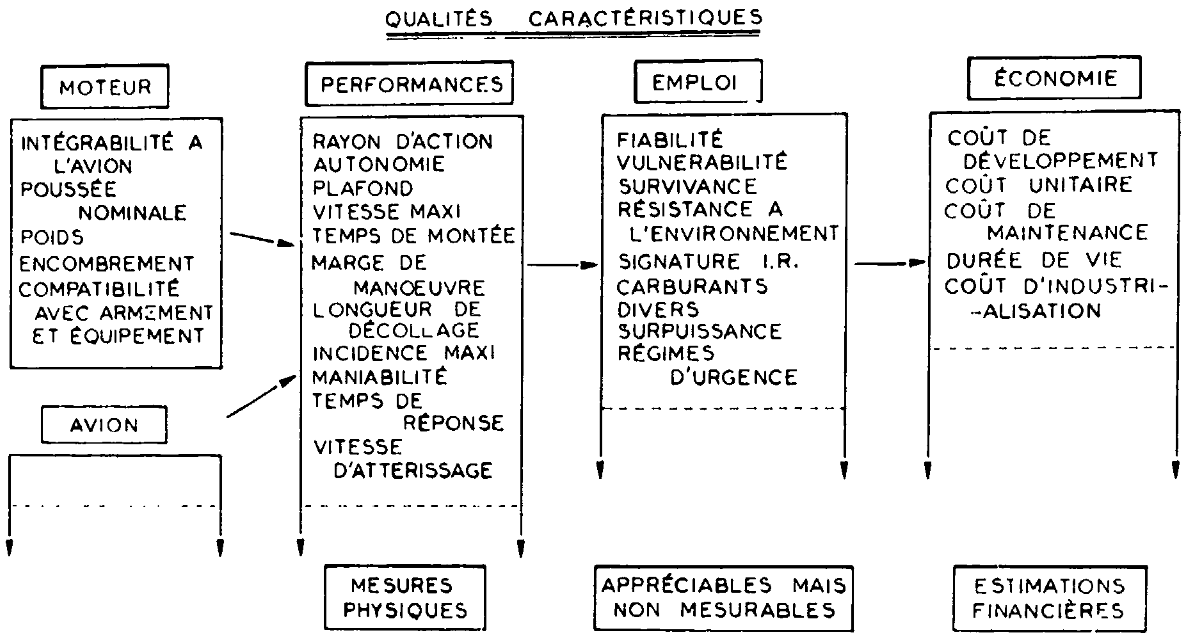


Fig. 5 - Principales qualités caractéristiques

Nous avons classé ces caractéristiques en trois groupes :

Les performances sont physiquement mesurables et s'expriment en kilomètres, en secondes, en nœuds ou en "g".

Les caractéristiques économiques s'expriment en unités monétaires bien définies mais ce ne sont que des estimations, la mesure ne pouvant se faire que bien plus tard, trop tard.

Enfin les qualités d'emploi ne se mesurent pas vraiment ; elles peuvent s'apprécier ou prendre un caractère statistique.

Dès lors le problème qui va se poser est celui de l'amalgame de ces qualités, ramenées si possible à une seule unité de mesure, et pondérées pour former la fonction à optimiser. Nous allons voir que suivant le type d'avion envisagé, ce problème est plus ou moins difficile, voire réaliste. En effet, l'opérateur principal qu'il faut appliquer aux caractéristiques de performances pour les mesurer est la mission exacte que devra tenter d'accomplir l'avion . Dans le cadre du présent exposé il ne sera pas possible d'entrer dans les détails et d'ailleurs notre propos doit s'orienter vers l'apport de la géométrie variable.

Voyons sur un exemple l'application des concepts exposés précédemment.

Soit le problème de la surveillance des côtes ; l'énoncé devient :

- Mission : Surveillance des côtes
- Outil : Avion classique subsonique
- Critère : Taux de présence en tout point
- Paramètres : Taille et cycle du moteur
- Contraintes : Emport du matériel spécialisé
  Budget alloué
- Calcul : Pour divers avions de transport existants on peut déterminer le taux de présence correspondant à la flotte permise par le budget

Nous voyons ici qu'intervient très fortement l'une des qualités, qu'on peut représenter par l'autonomie. Recherchant ainsi les points importants dans un certain nombre de cas, on peut dresser le tableau de la figure 6 dans lequel la taille des rectangles indique approximativement l'importance attachée à chaque caractéristique pour la mission correspondante.

Examinons quelques cas plus en détail .

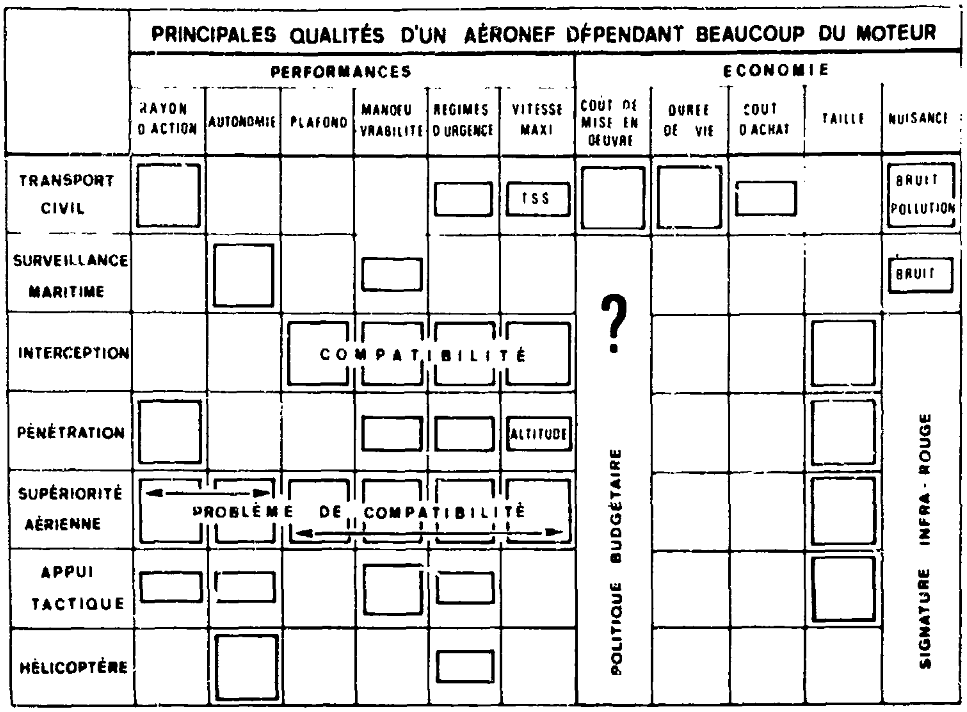


Fig. 6 - Importance des qualités pour les missions

AVIONS DE TRANSPORT CIVILS SUBSONIQUES

Les profils de missions d'un avion de transport civil étant assez peu variés et les critères de choix étant des critères économiques, le problème de l'optimisation d'un projet d'avion est assez facile à formuler. On peut le résumer de la façon suivante :

- Minimiser le coût du transport (coût du siège x kilomètre ou de la tonne x kilomètre) ;
- Respecter des contraintes imposées par la règlementation, les infrastructures aéroportuaires, le réseau des compagnies aériennes, les goûts des passagers, telles que
  . le niveau de bruit
  . la longueur de piste nécessaire au décollage et à l'atterrissage
  . la charge marchande
  . la longueur d'étape
  . la vitesse de croisière
  . l'altitude de croisière
  . le plafond avec un moteur en panne

Dans la pratique l'optimisation peut être découplée en deux optimisations séparées du moteur et de la cellule. Ceci est évidemment schématique, en réalité une collaboration entre avionneurs et motoristes est nécessaire, lorsqu'on lance un projet de moteur il est évidemment préférable de savoir à quel type d'avion on le destine. Cette collaboration est en particulier indispensable pour choisir un point de croisière et pour choisir la taille du moteur. Une fois ce choix fait, il s'agira pour le motoriste de :

— Réduire la consommation spécifique au point de croisière retenu . Ceci est imposé par le poids du coût du carburant dans le coût du transport aérien .

  Si la longueur d'étape était imposée,on pourrait dans la recherche d'un cycle optimum tenir compte des phases de montée et de descente . En fait une telle optimisation serait de peu d'intérêt car le poids respectif des phases de montée, de croisière et de descente dépend énormément de la distance à parcourir. Il serait économiquement absurde de concevoir des moteurs différents pour long-courriers et court-courriers. C'est pourquoi on peut dire qu'il existe un point de calcul (typiquement autour de Mach 0,8 30 000 ft) où il s'agit pour le motoriste de minimiser la consommation spécifique.

  Un optimum économique entre coût du moteur et consommation spécifique existe sans doute. En effet la diminution de la consommation des moteurs a pour conséquence une sophistication et donc un coût accru de ceux-ci . L'augmentation du coût du carburant déplace l'optimum vers des moteurs moins gourmands mais plus chers à l'achat et d'entretien. Il en est ainsi dans les pays industrialisés (qui fabriquent les moteurs et les avions) mais on pourrait concevoir que dans des régions du monde où le carburant est bon marché et le personnel qualifié est rare et cher, l'optimum soit différent et que des matériels peu performants mais rustiques et bon marché soient préférables.

— Respecter le niveau de bruit imposé par la règlementation, pour autant qu'il existe une population susceptible d'être gênée.

— Veiller au rapport poussée/poids du moteur car la masse du moteur se répercute (avec un coefficient multiplicateur) sur la masse de l'avion et en aéronautique le poids est l'ennemi n° 1 des performances et des coûts.

Les deux contraintes que nous venons d'évoquer n'entrent pas nécessairement en conflit avec la recherche d'un optimum de consommation comme le montre l'évolution des cycles des moteurs d'avions civils. Les moteurs très dilués, à fort taux de compression et à température devant turbine élevée permettent de satisfaire les trois exigences citées plus haut (JT9 , RB 211 , CF 6 , JT 10 , CFM 56).

Les constructeurs de cellules procèdent ensuite à une optimisation de leurs projets (flèche, épaisseur de voilure, taille de voilure, nombre de moteurs etc....) compte tenu des moteurs disponibles, de l'état de l'art en matière d'aérodynamique, de structures, de systèmes et des contraintes évoquées plus haut.

La décomposition de l'optimisation des projets d'avions civils en vue d'une optimisation séparée des moteurs et des cellules traduit assez bien la réalité mais elle la schématise. Nous avons parlé de l'intervention des avionneurs au moment du choix du point de croisière et de la taille du moteur mais elle nous semble souhaitable également au moment du choix du cycle. En effet, la recherche du taux de dilution, si elle est favorable au bruit et à la consommation, nuit au rapport poussée/poids à grande vitesse , à la traînée des nacelles, à l'encombrement des moteurs, ce qui a des répercussions sur les cellules.

AVIONS MILITAIRES

Le problème est de même assez facile à poser au motoriste lorsque les missions sont simples et bien définies. Ce sera le cas des avions de transport ou de surveillance maritime dont le cas diffère peu de celui des avions civils. Il est significatif que ces avions emploient les mêmes moteurs que les avions civils, le choix du turbopropulseur ou du réacteur double flux résultant du choix du point de croisière.

Pénétration , appui tactique

Ce sera également le cas des avions de reconnaissance, de pénétration ou d'appui tactique. Pour ces avions il s'agira toujours de maximiser le rayon d'action à charge militaire donnée ou de maximiser la charge militaire à rayon d'action donné, ce qui se traduit pour le motoriste par la recherche d'une consommation spécifique minimale au point de croisière fixé, l'avionneur se chargeant de produire une cellule de masse et de traînée minimale. Pour ces avions le taux de motorisation est secondaire, il est moins important que le rayon d'action ,il faut simplement assurer à l'avion des capacités d'auto-défense suffisantes et conserver des longueurs de piste au décollage raisonnables. On pourrait schématiquement poser le problème de la façon suivante :

- Minimiser la consommation spécifique en croisière à taux de motorisation en combat et au décollage imposés. Ces deux contraintes interviennent non seulement dans le choix du cycles mais aussi dans le dimensionnement du moteur.

La seule part d'arbitraire dans la formulation précédente provient du choix du taux de motorisation . Le respect d'un taux de motorisation minimum va à l'encontre de la consommation la plus faible. Le meilleur avion du point de vue rayon d'action serait sans doute très peu motorisé (à cause de la forme des courbes Cs (F)) mais très peu apte au combat et demanderait beaucoup de piste pour décoller. Un problème analogue se pose d'ailleurs au constructeur de cellule lorsqu'il doit choisir une surface de voilure. Le meilleur avion de pénétration au point de vue rayon d'action aurait un moteur et une voilure dimensionnés pour la croisière basse altitude, c'est le missile de croisière.

Intercepteurs

La caractéristique essentielle demandée à l'intercepteur est la capacité d'accélération et la vitesse ascensionnelle qui dépendent du bilan poussée-traînée :

$$\frac{T - X}{M} = \frac{dv}{dt} + \frac{g}{V}\ \frac{dH}{dt} = \frac{g}{V}\ \frac{dHt}{dt}$$

avec T = poussée

X = traînée

V = vitesse

H = altitude

$Ht = H + \frac{V^2}{2g}$ = altitude totale

M = masse de l'avion

L'intervalle de temps nécessaire pour passer de l'altitude totale Ht1 à l'altitude totale Ht2 est donnée par la formule :

$$t = \int_{Ht1}^{Ht2} \frac{g}{V}\ \frac{M}{T - X}\ dHt$$

L'intercepteur doit donc avoir un rapport poussée/poids le plus élevé possible, le rapport poussée/poids du moteur doit donc être le plus élevé possible.

La poussée variant avec le Mach et l'altitude de façons différentes suivant les cycles, le cycle le meilleur est celui qui maximise la probabilité d'interception des avions hostiles. En toute rigueur cet optimum dépend des performances des radars et des missiles (portée et capacité d'accélération ou de denivelée) , plus les missiles sont performants et moins il est nécessaire que les avions intercepteurs soient capables de très grandes vitesses. Ceci n'est pas sans conséquences sur le choix des cycles des moteurs. Par conséquent si on peut dire que l'augmentation des rapports poussée/poids est un paramètre essentiel d'optimisation, le choix du meilleur cycle doit résulter d'une optimisation des performances du système avion-missile, compte-tenu des hostiles à intercepter.

La consommation de carburant n'est pas à négliger car l'intercepteur doit posséder une certaine capacité de poursuite à grand Mach et grande altitude pour contrer d'éventuelles évasives de l'avion à intercepter. Or à consommation spécifique donnée le débit de carburant est proportionnel à la poussée.

Cependant l'augmentation de la poussée à Cs constante conduit en général à diminuer la quantité de carburant nécessaire pour atteindre un niveau d'énergie donné. Pour mesurer l'efficacité d'un avion en croisière on définit la consommation kilométrique :

$$C\,km = \frac{C_s\,T\,\Delta t}{V\,\Delta t} = \frac{\text{débit de carburant}}{\text{vitesse}}$$

De même pour mesurer l'efficacité d'un intercepteur sur le plan de la consommation de carburant, on peut rapporter celle-ci à l'accroissement d'altitude totale :

$$CHt = \frac{\text{carburant}}{\text{altitude totale}} = \frac{C_s\,T\,dt}{dHt}$$

$$CHt = \frac{C_s\,Mg}{V}\;\frac{T}{T-X}$$

En différenciant :

$$\frac{d\,CHt}{CHt} = \frac{dC_s}{C_s} + \frac{dT}{T} - \frac{dT}{T-X} = \frac{dC_s}{C_s} + \frac{dT}{T}\left(\frac{-X}{T-X}\right)$$

et à Cs constante :

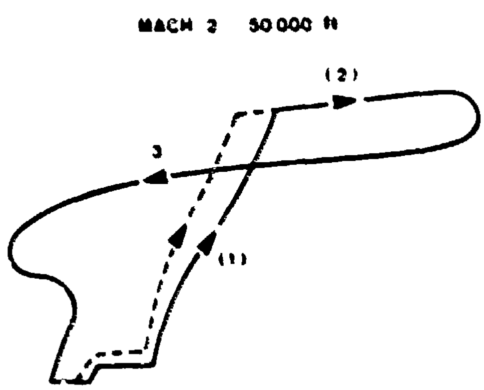
$$\frac{d\,CHt}{CHt} \Big/ \frac{dT}{T} = \frac{-X}{T-X} < 0$$

La recherche de rapports poussée/poids élevés conduit donc à diminuer les temps de montée et à améliorer les temps de poursuite . On peut donc augmenter la poussée en dégradant la Cs (par exemple par augmentation du taux de post-combustion) sans pour autant augmenter CHt , du moment que :

$$\frac{dC_s}{C_s} - \frac{X}{T-X}\;\frac{dT}{T} \leq 0$$

La figure 7 montre l'influence d'une augmentation de 10% de la poussée dans tout le domaine de vol, sur le déroulement d'une mission d'interception. L'avion le plus motorisé monte plus vite en parcourant une distance plus faible, il peut effectuer une poursuite plus longue mais notons qu'en rayon d'action les deux avions sont équivalents.

MISSION D'INTERCEPTION



INFLUENCE D'UNE AUGMENTATION DE POUSSÉE +10% A CS CONSTANTE

	Carburant consommé %	Référence (——)	Poussée Augmentée (- - -)	Gains
1	Roul. Décollage. Montée Taxi T.O. Climb	66	57	-9
2	Poursuite. P.C. Partielle	14	23	+9
3	Retour. Desc. Réserves	20	20	=
		Temps de montée		-25%
		Temps de poursuite		+60%
		Rayon d'action Radius		=

EFFECT. OF 10% MORE THRUST AT SAME S.F.C

Fig. 7 - Exemple de calcul d'influence

Supériorité aérienne

Le problème devient plus difficile à poser lorsque le moteur doit equiper un avion dit de supériorité aérienne. En effet ce terme est assez vague et il recouvre des missions extrêmement variées.

La figure 8 donne des exemples typiques de missions de supériorité aérienne . On peut constater que l'avion doit posséder de nombreuses qualités :

. Etre apte au combat tournoyant donc avoir une grande manoeuvrabilité,

. Avoir de bonnes capacités d'accélération supersonique ,

. Etre économe en carburant en croisière subsonique vers Mach 0,6 - 0,8 et 20 000 - 30 000 ft,

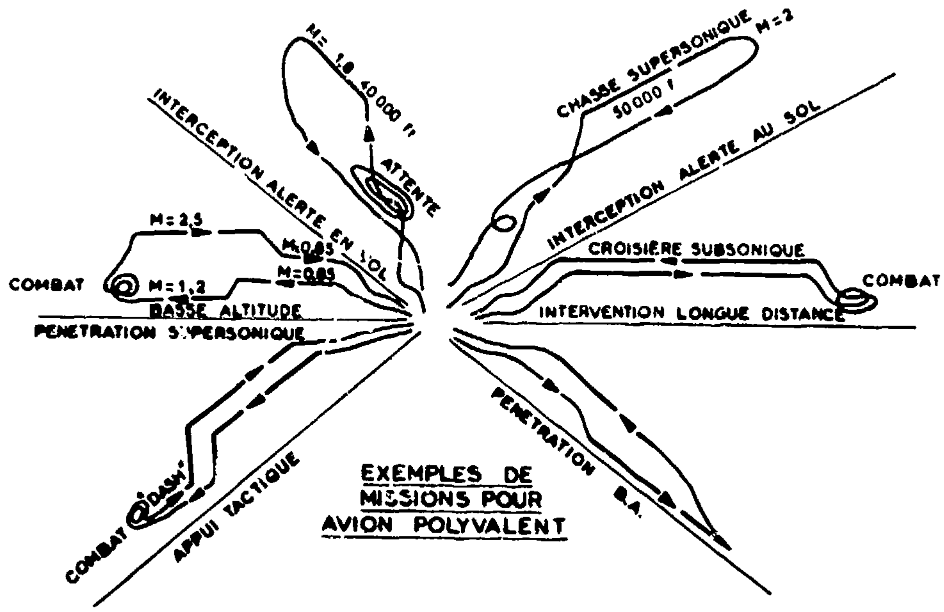


Fig. 8 - Description de missions de supériorité

De plus au cours de sa carrière opérationnelle un tel avion peut se voir confier des missions de pénétration basse altitude ou d'appui tactique. Il faudra donc rechercher un bon rayon d'action à basse altitude.

En bref, l'avion conçu pour la supériorité aérienne est un avion polyvalent du fait de la variété des missions que recouvre le terme supériorité aérienne. Il est donc difficile de parler d'optimisation dans ce cas car on attache autant d'importance à des exigences contradictoires, comme nous le verrons plus bas. Il serait plus exact de dire que l'avion et son moteur sont les résultats de compromis plutôt que d'une optimisation. L'exigence de forte motorisation est contradictoire avec celle d'une faible consommation kilométrique à basse altitude ou d'une grande autonomie en combat à la poussée maximale. Mais il faut noter que le taux de motorisation à une vitesse et une altitude donnée dépend autant du choix de la taille du moteur que du choix du cycle.

GEOMETRIE VARIABLE DES AVIONS ET DES MOTEURS

On demande à un avion polyvalent typique :

- d'avoir un rayon d'action élevé en pénétration à basse altitude ,
- d'être capable d'intercepter des avions volant à grande vitesse et haute altitude,
- de posséder des caractéristiques à basse vitesse satisfaisantes (vitesses d'atterrissage et de décollage).

De même que la forte motorisation (pour le combat et l'interception) va à l'encontre de la faible consommation à poussée réduite (pour la pénétration à basse altitude) , de même les deux premières exigences conduisent à des surfaces de voilure faibles et les deux dernières à des surfaces de voilure grandes. De plus les performances en supersonique demandent des ailes minces en flèche au détriment des performances à basse vitesse. C'est dans le but de résoudre ces contradictions qu'ont été construits les prototypes à géométrie variable . La flèche variable permet en effet de concilier les performances à basse vitesse et les performances en supersonique au prix bien entendu d'une complexité et d'un coût accrus.

Ici nous pouvons tenter de faire un parallèle entre la géométrie variable appliquée aux cellules et la géométrie variable appliquée aux moteurs.

Ce qu'on a coutume d'appeler avions à géométrie variable sont les avions à flèche variable. En réalité les avions à flèche fixe équipés de becs et de volets mobiles, d'entrées d'air à parties mobiles sont déjà des avions à géométrie variable.

Quant aux moteurs, quant on parle de géométrie variable on pense souvent cycle variable et en particulier taux de dilution variable. Il ne faut pas oublier que les moteurs équipés de post-combustion sont déjà des moteurs à cycle variable et que les portes de décharge, les aubes à calage variable sont de la géométrie variable.

Il y aurait donc pour les cellules comme pour les moteurs une "grande" géométrie variable et une "mini" géométrie variable.

	Cellules	Moteurs
- Grande G.V. Niveau III	Flèche variable	Taux de dilution variable
- Mini G.V. Niveau II	Hypersustentateurs Cambrure variable	Aubes à calage variable Portes de décharges (?)
Niveau I	Entrées d'air Tuyères	

Essayons de justifier ce parallèle plus en détail :

- Le choix d'une flèche et d'une épaisseur relative de voilure est un choix du même type que le choix d'un taux de dilution. Le constructeur de cellules doit faire un compromis entre des performances basse vitesse et des performances grande vitesse, le constructeur de cellules doit faire un compromis entre la consommation spécifique en subsonique et la consommation et la poussée spécifiques à grand nombre de Mach.
- Le choix de la surface de voilure "ressemble" à celui d'une taille de moteur. Une grande surface permet de diminuer les vitesses d'approche et d'améliorer la manoeuvrabilité mais augmente la trainée lorsque le vol se fait à grande vitesse et faible incidence (pénétration basse altitude à grande vitesse ou interception) . Un taux de motorisation élevé donne des performances brillantes en combat et en accélération mais est inutile et même nuisible en pénétration et en appui tactique.
- Les dispositifs hypersustentateurs (becs de bord d'attaque et volets de bords de fuite) permettent une meilleure adaptation de la voilure à son point de fonctionnement (vitesse et incidence). Ils procurent un domaine de bon rendement plus large qu'une voilure de cambrure et de vrillages fixes, à partir d'une forme en plan et d'une loi d'épaisseur données. Il y a donc une certaine analogie entre cette mini G.V. et les niveaux I et II de GV sur les moteurs.

L'expérience acquise dans la géométrie variable sur avion peut alors nous éclairer sur l'intérêt de la géométrie variable appliquée aux moteurs.

Les essais en vol et les études ont montré que dans certains cas particuliers la flèche variable ne s'imposait pas. Les progrès de l'hypersustentation du point de vue aérodynamique et réalisation mécanique permettent de trouver une combinaison de surface de voilure, de flèche et d'épaisseur relative qui réalise un bon compromis entre diverses exigences contradictoires et qui réponde aux besoins (en particulier en matière de vitesse d'atterrissage).

Cette conclusion n'a pas de portée générale quant à l'intérêt de la flèche variable. Il est bien certain que si l'on voulait réduire la vitesse d'approche sans réduire les autres demandes, le recours à la flèche variable serait inévitable. Mais dans certains cas on peut estimer qu'une réduction supplémentaire de la vitesse d'approche ne vaut pas la complexité et le coût de la flèche variable. Il y a bien sûr une part d'arbitraire et certains utilisateurs qui ont des besoins opérationnels particuliers ont pu faire des choix différents (F 14 , MRCA).

Mais comme nous le disions plus haut, le résultat d'un processus d'optimisation dépend des contraintes imposées a priori au produit final et il est très important de connaître le coût de ces contraintes ; il est évident que plus on est exigeant et plus le produit final sera cher.

Dans le cas des avions, les progrès de la géométrie variable au niveau II permettent de se passer dans une certaine mesure de la géométrie variable du niveau III.

CONCLUSION PARTIELLE

Au terme de l'examen que nous venons de faire de quelques cas particuliers nous pouvons tirer une première conclusion. Certes cet examen est très incomplet. Nous n'avons pas essayé d'expliciter toutes les missions militaires, cela nous aurait conduit beaucoup trop loin . Nous n'avons même pas fait appel à tous les paramètres significatifs définissant une mission. Par exemple nous nous sommes limités aux grandes distances parcourues par l'avion ; les petits mouvements sont aussi importants, c'est à dire les manœuvres indispensables pour le tir des armes et missiles, les combats tournoyants etc.... Le pilotage classique de l'avion dans les manoeuvres opérationnelles actuelles conduit à exiger du moteur la stabilité dans un domaine d'incidence-dérapage étendu, comme le montre la figure 9 .

Malgré la simplification faite, il est clair que le problème des choix techniques a pu être raisonnablement formulé en termes d'optimisation pour des avions (et des moteurs) spécialisés , par exemple maximiser le rayon d'action basse altitude à Mach 0,8 , à masse au décollage et taux de motorisation imposés, ou bien inversement minimiser la masse au décollage à rayon d'action imposé. Le problème est alors bien posé et les choix peuvent être éclairés par des études paramétriques significatives.

Par contre lorsqu'on veut réaliser un avion ou un moteur polyvalent et qu'on accorde une importance égale au rayon d'action, aux capacités d'accélération, à la manœuvrabilité, à la vitesse d'atterrissage, il ne s'agit plus de faire une optimisation mais des compromis. Supposons néanmoins que l'on veuille utiliser une fonction "coût" à minimiser qui fasse inter-venir le rayon d'action, le taux de motorisation, la vitesse d'approche etc..., le taux d'échange entre ces diverses performances serait totalement arbitraire. Quelle augmentation de poussée à Mach 2 faut-il pour compenser une augmentation de 1% de Cs à Mach 0,9 ? Toute réponse contient une bonne part d'arbitraire et la fonction "efficacité globale" d'un avion de combat reste encore à trouver. La recherche d'un meilleur coût-efficacité est un état d'esprit plus qu'un problème mathématique, surtout pour des avions militaires pour lesquels le problème de la rentabilité économique ne se pose pas.

C'est pour cela nous semble-t-il que l'optimisation formelle d'un moteur (ou d'un avion) en fonction de sa mission au sens classique est un problème bien difficile à poser. Toute tentative de faire apparaître les concepts "mission", "outil","critère", "paramètres" et "contraintes" est un échec dès lors qu'il ne s'agit pas d'un cas très spécialisé. Est-ce à dire que des solutions raisonnables n'existent pas ? En fait il y a longtemps que des solutions approximatives assez satisfaisantes ont été trouvées (tableau extrait de AGARD-L.S.72) :

<u>Moteurs adaptés à divers types d'avions</u>

	Rapport de pression	T chambre	Dilution	T.E.T.	Rendement du compresseur	Compression soufflante
Sustentation	faible	fort	nulle	maxi	pas très important	-
Intercepteur courte portée grand Mach	faible	fort	nulle	maxi	pas très important	-
Chasseur courte portée	moyen	fort	nulle	maxi	non dominant	-
Chasseur grande portée	fort	moyen	< 1	maxi	non dominant	forte
Bombardier supersonique	fort	fort	≈ 1	maxi	non dominant	forte
Bombardier subsonique	fort	moyen	moyenne	limitée	important	moyenne
Appui tactique	fort	moyen	moyenne	limitée	important	moyenne
Surveillance	fort	faible	forte	limitée	important	faible
Transport longue distance	fort	faible	forte	limitée	extrêmement important	faible

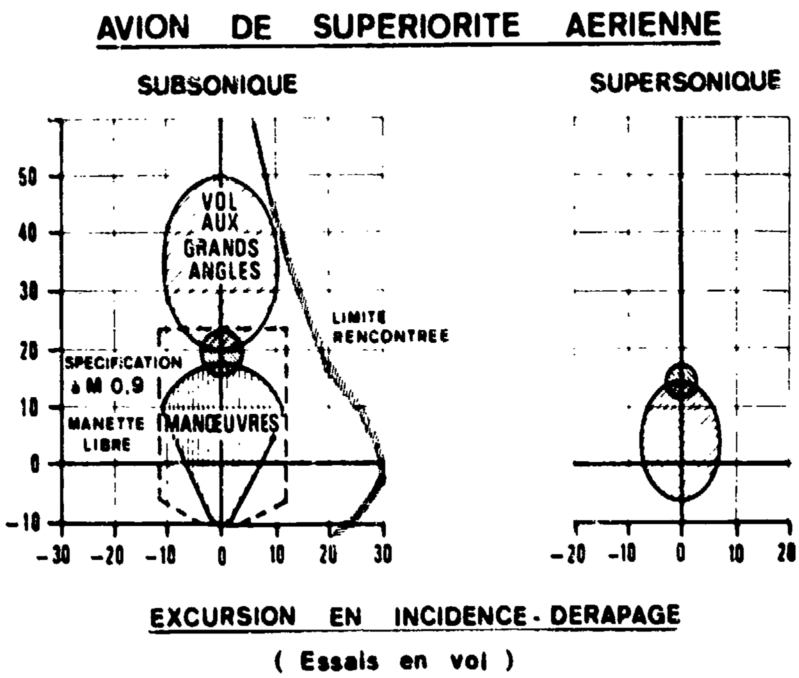


Fig. 9 - Domaine de vol en incidence - Dérapage

LA MISSION AU SENS LARGE

Nous avons vu que l'optimisation formelle se heurtait à des difficultés principalement parce que la mission effective de certains avions était trop complexe. Néanmoins nous savons assez bien quel modèle il convient de choisir parmi les types déjà existants et la géométrie variable ne modifierait pas fondamentalement le problème tout en le compliquant par la présence de choix supplémentaires. Quest-ce qui va donc guider la décision de fixer tel paramètre (par exemple T.E.T maxi) à telle ou telle autre valeur ? Tout d'abord des considérations sur la mission, au sens large cette fois, du moteur .

Un premier point est de savoir si l'utilisation du moteur doit être pensée en fonction d'un contexte de paix ou de conflit. En temps de paix l'un des objectifs de tout système de défense sera de se procurer des armes d'un haut niveau technologique sinon du plus haut niveau accessible. En cas de conflit il s'agira de produire un maximum d'appareils éprouvés . Le tableau ci-après reprend les différences caractéristiques entre ces deux situations . Il est clair qu'à la lumière de telles considérations des choix très différents peuvent être faits suivant le point de vue adopté.

On notera en particulier que le choix d'un avion bi-moteur plutôt que mono-moteur est fortement influencé par l'aspect sécurité des vols pendant les périodes d'entraînement. Il en résulte que la taille du moteur et son cycle sont déterminés très différemment. La géométrie variable apportera-t-elle une solution nouvelle en offrant des cycles et régimes d'urgence augmentant la fiabilité d'ensemble ?

Paix		Conflit
	Objectif	
Niveau technologique		Equipement massif
	Efficacité	
Taux d'accidents faible		Grand nombre de sorties avant indisponibilité
Nombre élevé d'années exploitables		Taux de survicance élevé après bataille
Economie d'emploi		Performances de pointe
	Conditions d'emploi	
Entretien régulier		Environnement hostile
Apprentissage		Service intensif
Entrainement		Logistique allégée
Service programmé		Conduite brutale
Bruit-pollution à éviter comme nuisibles		Nuisances non considérées
	Conditions de production	
Production étalée		Production accélérée
Modifications possibles		Approvisionnement difficile ou impossible
Produits spéciaux disponibles		
	Emploi effectif (exemple)	
Photographier		Détruire

Un point important est aussi de savoir si le budget alloué à la mise au point et à la mise en oeuvre de l'avion (et du moteur) sera effectivement suffisant. Le coût des systèmes de défense étant généralement considéré par beaucoup de personnes plus ou moins influentes comme toujours trop élevé, on peut dire que la mission d'un nouveau système comporte une certaine réduction des coûts . Au niveau de l'achat et de l'entretien on peut se demander si la géométrie variable permettrait de ne réaliser qu'un seul moteur pour plusieurs types d'avions ce qui aurait des conséquences industrielles peut-être favorables grâce à des séries allongées et des pièces de rechange plus standardisées. On objectera cependant qu'il s'agirait vraisemblablement d'un appareil plus compliqué, moins facilement modulaire, d'une mise au point plus délicate et par conséquent peut être plus cher.

Au niveau global on cherchera à se procurer un matériel capable d'une vie prolongée . Il est possible que la géométrie variable ait la capacité d'assurer, au moins en partie, une compensation de la dégradation due à l'usure, et prolonge la vie utile des moteurs. On peut surtout imaginer que la géométrie variable offre deux modes de travail, grâce à une programmation de la régulation : un mode "routine" permettrait des performances honnêtes tout en garantissant une faible consommation du potentiel, un mode "opérationnel" permettrait des performances supérieures en cas de nécessité.

Enfin, pour des raisons industrielles ou politiques, il peut être intéressant d'exporter le plus possible de matériel Le modèle retenu pour un projet doit donc présenter le maximum d'attrait pour divers utilisateurs étrangers. Si la géométrie variable offre peut être l'occasion de répondre mieux, avec un même moteur, aux différentes missions auxquelles les acheteurs éventuels porteront plus d'intérêt, et de mieux s'adapter aux climats parfois très particuliers correspondants, il convient de penser aux problèmes que peut poser le niveau technologique pour la maintenance au moins.

LES CONDITIONS DE L'OPTIMUM

Les chapitres précédents ont montré l'impossibilité de formuler mathématiquement le problème de l'optimisation et que le choix optimal devra être arrêté sur la base d'un compromis. Quelles sont les conditions à réunir pour que ce compromis aboutisse à un résultat honorable sinon brillant ?

Les niveaux de réalisation de ces conditions nous paraissent être les véritables paramètres de l'optimisation.

Au premier chef il s'agira d'assigner à l'avion une mission correctement définie même si elle est imprécise. Autrement dit, il faut que les spécifications du système d'arme ne présentent pas d'exigences maladroites. Par exemple il serait vain de spécifier le décollage court sur terrain non aménagé d'un avion par ailleurs très coûteux et vulnérable car on ne le risquera pas dans de telles conditions. Inversement il ne faut pas oublier d'incorporer dans les spécifications tout ce qui se montrera utile pendant la vie opérationnelle. Une certaine dose d'imagination est sûrement nécessaire pour penser à l'avance à tout ce qu'on pourra demander à un même avion. D'autant plus que la mise en service elle-même est une étape encore lointaine au moment de décider le lancement du projet.

Cette projection dans le temps est le propre des études prospectives. C'est sur leur qualité et leur exactitude que repose la définition des missions ou plus simplement de la façon dont on se servira de l'avion (et de son moteur) quelques 10 ans plus tard. Sur un tel intervalle de temps la politique peut évoluer largement et les pilotes peuvent inventer de nouvelles manœuvres. On voit que la prospective doit plonger ses racines très loin, à tous les niveaux de responsabilité et ne pas se contenter d'un formalisme théorique.

D'une façon plus directe, on pourra dire que le moteur optimal est celui qui satisfait son utilisateur. Pour autant que celui-ci ne se montrera pas déraisonnable, il sera très souvent possible de le satisfaire effectivement, à la condition essentielle de savoir exactement ce qu'il désire. Une information réciproque sur les exigences et les possibilités doit s'établir, c'est à dire un dialogue, coopératif, entre des interlocuteurs "valables". Or les problèmes de communication des idées ne sont pas toujours faciles à résoudre ; les mêmes mots n'ont pas partout le même sens. Des efforts particuliers doivent être faits pour s'assurer que toute l'information est parvenue, intacte, à tous les endroits nécessaires et qu'elle est convenablement exploitée.

La réussite dans ce domaine repose en partie sur une organisation systématique des services concernés (d'Etat ou Industriels) avec des sections et des responsables chargés de veiller à l'élaboration, à la transmission et à l'exploitation de l'information. Une méthode précieuse est celle des réunions largement ouvertes. A ce titre il est frappant de voir le nombre de réunions organisées par les Sociétés savantes américaines, qui provoquent un flot de communications grâce auxquelles on peut prendre connaissance des travaux effectués dans tout le pays. La publication d'index signalétiques très complets y aide encore.

C'est ainsi qu'il est très facile d'avoir les noms de plusieurs personnes qui travaillent sur un sujet auquel on s'intéresse et de les consulter. Il ne nous semble pas qu'une telle facilité se retrouve ailleurs. Les réunions de tout type organisées par l'AGARD entrent dans le cadre d'un tel effort et on ne peut qu'espérer pour elles une audience toujours plus large.

Un autre facteur est celui des motivations qui animent tous les échelons concernés. On peut penser par exemple que dans un contexte de concurrence et d'émulation de meilleurs efforts seront faits pour tendre vers une réussite commerciale ou nationale. Faut-il rappeler ici les progrès très rapides de la technologie en temps de guerre?

La conception, la mise au point, la production d'un moteur représentent une grosse affaire industrielle qui relève donc des méthodes modernes de gestion. Celles-ci comprennent une fois de plus une organisation adaptée avec en particulier des maîtres-d'oeuvres munis de moyens d'action puissants et efficaces. C'est toute la structure des services concernés qui est en cause. Au minimum on exigera la continuité de l'action et la définition des responsabilités.

La pierre d'achoppement des grands projets est le dépassement des prévisions financières. L'une des conditions pour que l'optimum recherché puisse être atteint est donc que les prévisions initiales soient réalistes sinon exactes. Pour les établir comme pour les respecter il sera bon de faire appel aux méthodes plus ou moins connues et appliquées telles que "design-to-cost", " life-cycle-cost", "approche-système", "analyse de la valeur", etc....

L'organisation et les méthodes qualifient la qualité de l'effort, mais son intensité est aussi un paramètre important Cette intensité est évidemment dépendante de la puissance industrielle mise à contribution. Il est clair que si un Etat est capable de mettre au point simultanément ou presque plusieurs modèles d'avions différents, chacun d'eux pourra être spécialisé et donc, comme nous l'avons vu, bien optimisé. C'est la raison pour laquelle des projets sont abordés en coopération internationale. Les assises industrielles sont alors plus larges et plus puissantes, mais il en résulte bien souvent qu'on veut donner au produit un caractère plus polyvalent pour satisfaire le nombre accru de parties prenantes et cela a pour effet de s'écarter plus ou moins nettement de l'optimum.

Il faut prendre garde que l'intensité de l'effort accompli s'apprécie sur plusieurs plans :

- par le niveau du financement octroyé,
- par le nombre et la qualité des cerveaux employés,
- par les délais accordés pour l'achèvement.

On ne saurait trop insister sur la qualité du travail humain. Il ne s'agit pas seulement des trouvailles plus ou moins ponctuelles grâce auxquelles il serait possible de produire avec peu de moyens, en peu de temps et pour un faible coût un moteur merveilleux. Cela ne mène pas très loin. Il s'agit plutôt d'une formation de base et d'une politique d'emploi qui conduisent au travail méthodique et appliqué des personnels compétents. Ceci est évidemment un problème plus général.

De même que la qualité par la formation est une affaire permanente et de longue haleine, le niveau technologique disponible résulte de la politique menée sur plusieurs décennies. Or il est indispensable de mettre à la disposition des concepteurs les meilleures solutions techniques et technologiques . C'est à dire qu'on ne saurait pas réaliser un moteur optimisé, même si on peut le concevoir, si pendant de nombreuses années des efforts de recherches et de développement exploratoire n'ont pas été menés systématiquement, même sans bien savoir à quelle mission particulière ils pourraient être utiles. Un effort parallèle d'équipement est nécessaire, pour assurer une mise au point rapide des prototypes; nous pensons en particulier aux moyens d'essais , qu'il s'agisse d'investigations au niveau des matériaux ou des grandes souffleries capables de traiter un ensemble propulsif complet.

Deux exemples particuliers, parmi des sujets récemment discutés dans les Panels de l'AGARD, illustrent la nécessité absolue d'un effort de recherches et d'études, souvent arides et parfois difficiles à faire apprécier.

a) Il est clair que le moteur satisfaisant devra présenter un rendement élevé, ou du moins conforme aux prévisions si le niveau a été délibérément choisi en fonction d'autres critères. L'une des conditions est que les calculs d'écoulement internes dans les turbomachines que le constructeur est capable de faire pour dessiner les aubages soient exacts ; moyennant quoi il lui sera possible d'affiner sa conception et la réalité sera à la hauteur de ses espoirs. Mais le chemin est long, des équations de la mécanique à un programme opérationnel !

b) Faisant appel aussi à des considérations parfois très théoriques et scientifiques, mais encore plus dépendante de considérations pratiques est la technique (presqu'un art) de l'intégration d'un moteur à un avion . Or il s'agit là d'un élément essentiel de la réussite d'un projet . " An exceptional aircraft requires a superior engine with superior integration into a superior airframe - A poor engine in a good airframe yields an inferior aircraft - A good engine in a poor airframe yields a poor aircraft - A good engin improperly installed in a good airframe yields an inferior aircraft "(A.E. Fuhs ref 3).

CONCLUSION

Le tour d'horizon que nous avons fait est encore bien incomplet . De nombreux types d'avions militaires (ADAV, avions embarqués ...) resteraient à examiner en détail. Nénamoins, les conclusions quant à l'optimisation des moteurs restent les mêmes.

Dans la plupart des cas une optimisation formelle n'est pas possible et un compromis est déterminé par les facteurs suivants :

- vision claire de la situation présente et future et des objectifs poursuivis,
- moyens disponibles permettant ou non la réalisation de plusieurs modèles différents,
- niveau technologique accessible résultant des recherches et études,
- organisation capable de tirer le meilleur parti de l'état de fait.

Quant à la géométrie variable c'est d'abord une élévation au niveau technique qu'il conviendrait de maîtriser avant de prétendre l'exploiter. Le diagramme de la figure 10 résume les considérations développées :

- des moteurs spécialisés atteignent un haut niveau de rendement,
- un moteur "tous usages" classique reste à un niveau plus modeste (ce qui n'empêche d'ailleurs pas de faire de très bons avions),
- un moteur doté d'une "petite" géométrie variable (niveau 1 et 2) permettrait de faire mieux et sans doute assez facilement ,
- un moteur doté de la "grande" géométrie variable (niveau 3) devrait permettre un rendement moyen et finalement avantageux dans plusieurs cas d'utilisation très différents, mais il s'agit d'une perspective lointaine.

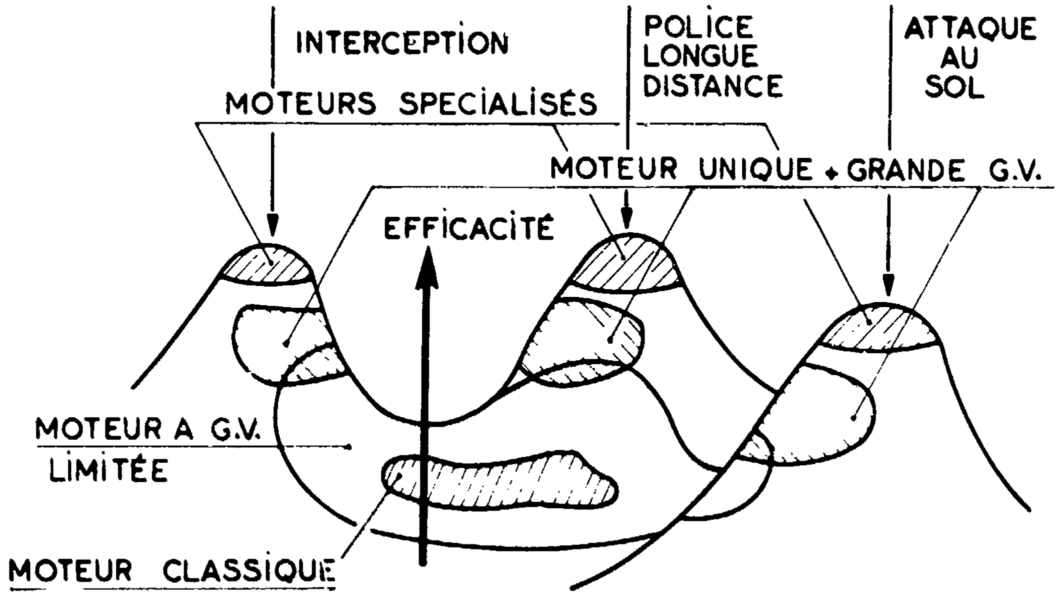


Fig. 10 - Relation efficacité - Spécialisation

Revenant au rapprochement avec les éleveurs que nous avons osé au début, on pourrait dire approximativement que si la technique actuelle nous permet de faire des moteurs qui ne sont pas des "veaux", un certain niveau de géométrie variable permettrait de faire des "pur-sangs", mais que nous ne sommes pas près de savoir produire des "moutons à cinq pattes".

En conclusion finale, nous recommanderons que s'instaurent, là où ils n'existent pas encore, des travaux effectifs théoriques et expérimentaux sur :

- l'évaluation, au sens des fonctions d'optimisation, des caractéristiques de performances, d'emploi et économiques des systèmes d'armes ,
- la conception et la mise au point de composants de moteurs à géométrie variable.

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DISCUSSION

N.F.Rekos

(1) You mentioned reconnaissance and long range subsonic patrols (ASW) as mission requirements using turbofan engines. Have you given much thought to high speed turboprop engines for the missions?

(2) You mentioned the need for modern test facilities. What are you doing about obtaining the types of facilities needed to test the variable geometry engines discussed at this meeting (SST, VTOL, Prop Fan).

Author's Reply

Les avions de surveillance maritime sont un cas où le problème d'optimisation est soluble car il s'agit de maximiser le taux de présence ou la durée d'observation d'un point déterminé, si vous voulez, et les contraintes sont d'emporter le matériel nécessaire à l'observation, et à partir de ce moment-là, on peut, en prenant des modèles de moteurs et d'avions existants calculer très facilement le taux de présence pour un budget global donné.

Il y a toujours une solution si on veut faire un avion qui ait une très grande autonomie, il y a toujours une solution avec n'importe quel moteur; mais le coût de l'avion peut devenir énorme parce que l'avion est très gros. Je crois que c'est un cas où la caractéristique dominante c'est l'économie de fonctionnement, ou très faible consommation spécifique, et c'est tellement dominant que l'on peut optimiser sur ce paramètre-là. Je n'entre évidemment pas dans les considérations techniques sur la méthode exacte pour obtenir une très faible consommation. Je ne dis pas qu'il faut faire une turbine variable, ou prendre une hélice. Je dis qu'il faut prendre le moteur qui donne la plus faible consommation spécifique et un nombre d'avions suffisant dans le cadre du budget alloué.

En ce qui concerne les moyens d'essais je pense qu'il y a une chose qui a déjà été dite ce matin, c'est l'intégration totale du problème du moteur avec celui de l'avion – et il faut absolument être en mesure de faire des essais incorporant les systèmes propulsifs complets. L'entrée d'air, le moteur, la tuyère, et peut-être dans certains cas, un morceau d'aile ou un morceau de fuselage suivant l'intégration du moteur et de la cellule. Alors ceci conduit à de très grandes installations d'essais. Il y a aussi certainement besoin de certains types d'installations d'essais qui correspondent mieux aux turbines à géométrie variable, ou aux compresseurs avec des rotors à calage variable.

M.Boudigues

Vous avez mentionné les dépassements budgétaires ou le manque de prévisions financières, ce qui laisserait penser que les constructeurs, en particulier français, ne savent pas faire ces prévisions. Je vous signale que nous en avons toujours faites et respectées.

D'autre part, je regrette que vous n'ayez pas parlé de la difficulté qu'il y a pour le motoriste à obtenir à la fois un rendement thermique et un rendement propulsif satisfaisants.

Author's Reply

Je dois dire que, ce n'est pas parce que je m'exprime en français que j'ai pris tous mes exemples parmi les seuls constructeurs français. Au contraire, il me semble assez facile de trouver des cas parmi nos amis étrangers où les prévision n'ont pas toujours peut-être été bien respectées. C'est ce que j'appelle s'écarter de l'optimum, parce que l'optimum c'est de rester dans ses propres prévisions. Je ne pense pas que mon exposé ait eu l'intention de se baser uniquement sur les expériences françaises que je suis censé mieux connaître que les autres, ce qui n'est pas prouvé. Deuxième point en ce qui concerne le rendement je crois que dans le papier précédent il y avait justement une figure où l'on voyait qu'à régime réduit le rendement propulsif croît et le rendement thermique décroît et il en résulte une décroissance de performance globale. Donc c'est exactement le problème que vous venez de soulever et nous avons essayé de penser à ça en disant il y a le premier niveau de géométrie variable qui s'addresse justement au rendement donc à l'aspect thermique et le deuxième niveau qui s'adresse à l'aspect adaptation au régime de vol, donc à l'aspect propulsif. C'est une meilleure expression peut-être de ce que nous avons voulu dire.

ADVANCED ENGINE DESIGN CONCEPTS AND THEIR INFLUENCE ON THE PERFORMANCE OF MULTI-ROLE COMBAT AIRCRAFT*

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Summary:

The effectiveness of a multi-role weapon system places new demands not only on the aircraft but also on the powerplant. When these demands are contrasted with the typical performance characteristics of current engines, it is logical to ask for a variable cycle engine.

In this context, therefore, such an engine concept is being considered which combines the advantages of the reheated turbojet, i.e.

- favourable ratio of dry thrust to thrust with reheat
- favourable fuel consumption with reheat
- excellent handling characteristics

with the important advantage of the reheated turbofan

- favourable dry fuel consumption.

It is shown to what extent this aim can be achieved by a suitable design, including variable geometry in several components. The performance data of this engine are compared with the data of current engines. The penalty in engine weight and complexity compared with current engines as well as the development problems expected are discussed.

The results indicate that the performance of multi-role combat aircraft could be improved by the installation of engines of the described concept provided the unavoidable weight penalty of such an engine remains within acceptable limits.

Introduction

Development work on advanced reheated engines for combat aircraft has not only led to enormous technical progress in the last two decades but also to considerable specialisation. Even if such engines have comparable thrust/weight ratios, turbine entry temperatures and overall pressure ratios, this specialisation is characterized by the wide range of bypass ratios, which now stretches from 0.25 to about 2.0. The specialisation results from the matching of the engine to various aircraft and missions, for which engine performance data and operating characteristics with due regard to installation effects serve as the criteria.

The soaring development and production costs involved in future weapon systems entail a trend to a diminishing number of available units. It also becomes more compelling to manage with as few complementary aircraft types as possible, employing them flexibly with high effectiveness. When modern combat aircraft are used on flexible missions, their effectiveness has to be compared with that of aircraft designed specially for specific missions. This comparison has to be made both airframe-wise and engine-wise. Engine-wise, it seems logical in this context to aim at combining the advantages of engines with low and high bypass ratios. This raises at once the question of engines with a variable thermodynamic cycle, i.e., variable geometry of the components.

Even if, in future, aircraft tailored for special missions are given preference, the requirement for engines with greater flexibility than at present remains relevant. This is understandable, as substantial demands have to be made on the flexibility of engines for combat aircraft, which operate in the subsonic to the supersonic range without and/or with reheat.

An experimental programme [1, 2]+ being carried out on behalf of the U.S. Air Force is partly focused on the increase in engine flexibility, as indicated above, through the introduction of variable-geometry components.

It should also be mentioned that considerable efforts are being made in the U.S.A., with respect to future supersonic commercial aircraft as well, in order to comply with requirements on the powerplant regarding economy in the supersonic and subsonic range and low ecological impact. Within the framework of the programme [3, 4, 5] being carried out by NASA together with the U.S. aircraft industry for this purpose, engine concepts with partly novel flow paths and layouts of the components, inter alia, are being investigated [6, 7, 8, 9, 10].

* The investigation was sponsored by the Ministry of Defence of the Federal Republic of Germany, ZTL No. MTU 1.15-1 and MTU 4.13-1

+ Numbers in square brackets refer to references at the end of the text

Against this background, an investigation was made, within the framework of the present study, of the extent to which the called-for flexibility in performance and operating characteristics of engines for modern combat aircraft can be implemented in a single engine concept. From the point of view of performance, an endeavour was made to combine the advantages of the reheated turbojet

- good specific fuel consumption with reheat
- high dry thrust relative to thrust with reheat

with the important advantage of the reheated turbofan

- good dry specific fuel consumption.

Furthermore, an effort was made to implement the well known advantage of the reheated turbojet compared with the reheated turbofan

- superior operating stability and handling qualities in connection with the afterburner.

Attention was also paid to

- favourable mass flow characteristic with regard to the intake
- good compatibility with inlet distortions
- optimum flow conditions around the aircraft afterbody to reduce afterbody drag.

Lastly, the aim was to meet the requirements while keeping the engine to

- acceptable weight and
- manageable complexity, especially of the control system.

Variable cycle engine concepts

The engine concepts initially considered are shown schematically in Fig. 1 . In each case - as an arbitrary example - the turbomachines are sketched as a three-shaft arrangement. With this, however, the shaft arrangement to be chosen later on for a specific case is not anticipated.

Concept I is a mixed flow turbofan in line with the current engine generation, the characteristics of which, as a function of the bypass ratio, are known. Owing to the mixing of the two flows, there is, depending on the flight condition, a specific relation between mass flow and thrust which can be influenced only slightly by changing the setting of the nozzle. The compressor stability can be increased by opening the bleed off-take downstream of the IP-compressor, which may be desirable in critical flight conditions.

This concept was used as a point of departure and basis of comparison for the other concepts to be discussed.

Concept II differs from I only in the variable LP-turbine. This allows the bypass ratio to be varied to a limited extent and consequently a degree of flexibility in the dry and reheated performance to be obtained [11] . Apart from the increase in the stability of the IP-compressor possible through the bleed off-take as in concept I, in this case the stability of the IP-compressor can be further improved by changing the setting of the LP-turbine. Reference is made also to this point in [12] . The three-shaft arrangement has an advantage over the two-shaft arrangement in this case, in that the speed of the HP-system is virtually unaffected when the setting of the LP-turbine is changed (Cf. [11, 12, 13]).

Due to the variation of the bypass ratio, problems could arise, on the one hand, with the flow between fan exit and IP-compressor entry. It remains an open question whether, on the other hand, a variable splitter is sufficiently justified, in view of the limited range in which the bypass ratio can be varied.

Concept III allows, by the use of separate nozzles for the primary and duct flows, a considerable variation of the bypass ratio when the settings of the IP- and LP-turbines and of the primary and duct nozzles are changed. At the same time, much more freedom is obtained in arranging a desirable relation between thrust and engine mass flow than in the case of concept II. With a view to the stability of the fan and the IP-compressor behind it, the wide range in variation of the bypass ratio makes a splitter with variable geometry seem indispensable, at least for the time being.

The separation of the primary and duct flows with no provision for duct burning eliminates, on the one hand, the problem of fan/afterburner interaction. This problem is increasingly evident in mixed flow turbofans with higher bypass ratio. On the other hand, it requires special routing of the afterburner cooling air. This portion of the mass flow, which is a considerable percentage in modern engines, is taken off substantially throttled, i.e., in such a manner that any immediate afterburner reaction on the compressor section is definitely prevented. Without reheat, the cooling air off-take is shut down. This requires an appropriate change in the settings of the IP- and LP-turbines and of the primary and duct nozzles. What is important is that the HP-system, in particular the HP-turbine, can be designed with fixed geometry.

When there are inlet distortions, the duct nozzle can be opened to increase the stability of the fan without having unfavourable repercussions on the turbomachines, as would be the case with concepts I and II.

As the primary nozzle is enveloped by the duct flow, the afterbody flow separation which is virtually inevitable in dry operation in the case of engines based on concepts I and II can be avoided.

Concept IV makes it possible, through simultaneous changing of the settings of the turbines and of the nozzle, to change the thrust by varying the turbine entry temperature, while maintaining high engine mass flow and high overall pressure ratio. As a result, at dry partload, specific fuel consumptions can be obtained which come close to those of turbofans. In this case, too, the afterburner cooling air must be taken off and throttled as in concept III. This concept is extremely advantageous for matching the engine mass flow to the intake capacity.

The configuration of the aircraft afterbody is just the same as for concepts I and II. At dry partload, however, the nozzle area is larger than for the turbofan, which reduces the tendency to afterbody flow separation at subsonic cruise flight.

This concept is familiar in principle from publications [14, 15].

Selection of engines considered

To enable a fair assessment of various engine concepts which are in existence or conceivable for the future, it is intended, within the framework of this study, to compare them on the basis of the currently available status of component technology. Logically, therefore, the per se attractive concept IV was not included, as the variable HP-turbine presupposes a very advanced standard of turbine technology, which surely must be considered as unavailable for some time to come. Though concept II promises an increase in performance flexibility, it was also not included in the present study, because it offers no decisive progress in operating stability and handling qualities.

Thus, the study was restricted to a comparison of concept I, representing the current engine generation, with the concept III variable cycle engine*. Two versions with bypass ratios of 0.25 and 1.3 (current engines C 0.25 and C 1.3) were considered in the process for concept I. The bypass ratio 0.25 corresponds to the minimum duct flow required for afterburner and nozzle cooling. Fig. 2 gives the main design parameters of the engines compared. Consistent assumptions were made about the component efficiencies and pressure losses, etc., taking into consideration the variable geometry of the turbines.

In the case of the conventional engines, as they are mixed flow engines, the fan pressure ratio is fixed by selection of turbine entry temperature, overall pressure ratio and bypass ratio. For the C 0.25 engine, which has a high fan pressure ratio and subsequently a moderate core pressure ratio, a two-shaft arrangement was chosen. For the C 1.3 engine with its lower fan pressure ratio, a three-shaft arrangement was selected.

In the case of the VCE, which is an unmixed flow engine, there is a degree of freedom about the fan pressure ratio. Variation of the fan pressure ratio in the range FPR = 1.8 - 3.6 showed that a relatively low pressure ratio FPR = 1.95, which can be managed with a two stage fan, gives favourable performance both in dry (especially dry partload) and reheated operation. The substantial throttling of the afterburner cooling air, introduced to avoid fan/afterburner interaction, leads to a loss in thrust of about 1.5 % in operation with reheat. In dry operation, dropping the mixing of the primary and duct flows worsens the specific thrust and specific fuel consumption in the same order of magnitude. In view of the low fan pressure ratio, a three-shaft arrangement was preferred. At the same time, this arrangement leads to an afterburner cooling air off-take at suitable pressure level situated between IP- and HP-compressor.

Fig. 3 shows the general arrangements of the VCE and the C 1.3 engine of equivalent thrust at design point. Whereas the C 1.3 engine requires only 4 control variables, the VCE has 8 control variables. The VCL is considerably heavier than the C 1.3 engine. The excess engine weight of the VCE is partly the result of the concept, i.e., a certain inevitable extra weight derives from the additional outer engine casing, the variable splitter with its actuation system, the actuation system for the variable IP- and LP-turbines, the heavier nozzle system and the substantially more complicated control system. A further cause for the excess weight of the VCE is given by the chosen cycle. As can be seen from Fig. 4, the specific thrust with reheat of the VCE at the selected bypass ratio of 0.34 is lower than in the case of the C 0.25 and C 1.3 engines, because reheat is restricted to the primary flow. A higher specific thrust comparable with the specific thrusts of the C 0.25 and C 1.3 engines can be obtained from the VCE, if the bypass ratio with reheat is brought down near to zero. In this way, leading dimensions and weights of the turbo-machinery and the afterburner comparable with those of the C 0.25 engine are obtained for the VCE. From other considerations it can be expected that, due to the flexibility of the VCE concept, the worsening of the cruise specific fuel consumption originally feared, will be only slight in the case of this modified design.

In spite of this possibility of improvement, the VCE was compared with the C 0.25 and C 1.3 engines in the present study on the basis of the design parameters in Fig. 2.

*Later called VCE for the sake of simplicity

Operating characteristics and performance

The same maximum turbine entry temperature and reheat temperature schedules were taken for all engines, as a function of flight altitude and Mach number, in line with the state of the art currently feasible with regard to cooling and stressing. For the same reasons, in spite of the given flexibility, the fan speed and the overall pressure ratio characteristics of the VCE were limited to those figures which automatically result from the turbine entry temperature schedule in the case of the C 0.25 and C 1.3 engines.

On this basis, for maximum dry and reheat thrust the other available control variables of the VCE (mass flow capacities of the IP- and LP-turbines and of the primary and duct nozzles) were set so that all three compressors are working under all flight conditions on their optimum running lines. At dry partload, in the case of the VCE, the cycle parameters turbine entry temperature, overall pressure ratio, fan pressure ratio and bypass ratio were fixed by means of appropriate setting of the control variables just mentioned, so that with optimum working lines in the three compressors minimum specific fuel consumptions are obtained.

Cycle and component data

Fig. 5a shows, for dry partload, turbine entry temperature and overall pressure ratio as a function of thrust, the behaviour of the compared engines being practically the same. Fig. 5b shows the associated characteristics for engine mass flow, bypass ratio and nozzle pressure related to engine entry pressure. The relatively high engine mass flow maintained by the VCE at partload, i.e., particularly under cruise conditions, is favourable with respect to engine/intake matching. The bypass-ratio plot shows that the VCE comes closer to the C 0.25 engine in the range of high thrust but is more like the C 1.3 engine at low thrust. This is roughly what was expected. The pressure in the duct nozzle reflects the fan pressure ratio and thus the fan mass flow characteristic, as the fan operates on a specified working line. However, there is a relatively big pressure drop in the primary nozzle as thrust decreases, as the fan draws off more and more specific work from the primary flow due to the increasing bypass ratio.

Owing to the high pressure level in the afterburner at maximum thrust, a similarly favourable afterburner blow-out ceiling can be reckoned with for the VCE as in the case of the C 0.25 engine.

The optimisation of the cycle parameters shown at dry partload was done first for the uninstalled engine for a variety of reasons. The selected control characteristic for the LP-turbine is anticipated to have a big influence on the optimum engine mass flow characteristic. When the installed engine behaviour is optimized, the shifting of the control characteristic of the LP-turbine makes it possible to obtain an engine mass flow characteristic which corresponds to an optimum engine/intake matching in terms of minimum installed specific fuel consumption.

The mass flow capacities of the IP- and LP-turbines required at max. dry and max. reheat thrust are shown in Fig. 6 as a function of the flight altitude and flight Mach number. The required capacities of both turbines for the important cruise condition SL/MN = 0.6 are also shown on this figure. The mass flow capacity characteristics for max. dry and max. reheat thrust can be reduced basically to a pure function of the ratio of turbine entry temperature to engine entry temperature. However, the changing of the settings of the turbines for dry partload is more complicated. Initially the turbine flow capacities tend to increase with decreasing thrust, reaching a maximum and then falling below the original level when thrust is decreased still further. Admittedly, the settings of both turbines can presumably be changed affinely. In the case of the LP-turbine, the required range of mass flow capacity is considerable.

In Fig. 7 the required ranges of mass flow capacity for IP- and LP-turbines under various running conditions are contrasted with the changes in the nozzle guide vane throat areas required and the efficiency characteristic to be expected according to available experimental data (Cf. [13, 16, 17]). Tangible losses in efficiency are inevitable in the case of the LP-turbine at max. reheat thrust, if good LP-turbine efficiency is aimed at for dry partload operation to obtain favourable cruise specific fuel consumption. It remains to be mentioned that, owing to the required wide setting range of the LP-turbine, the change in the flow direction relative to the axial direction at the LP-turbine exit will be in the region of $\pm 10^{\circ}$.

Analogously to the mass flow capacities of the turbines, Fig. 8 shows the ranges of primary and duct nozzle throat areas. The areas of both nozzles must be variable roughly in a 1:3 ratio. Similarly to the case of the turbines, the changing of the settings of the nozzles for max. dry and max. reheat for all flight conditions and engine running conditions can be managed as a pure function of the ratio of turbine entry temperature to engine entry temperature. Also at dry partload the situation is similar to that for the turbines, i.e., as thrust drops the nozzle areas first increase and then decrease as thrust drops still further. The analysis of these nozzle control characteristics shows that the relation between the areas of the two nozzles can be represented, with good approximation, by a unique function.

The setting of the splitter between fan and IP-compressor - see Fig. 3 - depends on whatever bypass ratio is obtained from the other control variables. As the splitter nose is in the innermost position at dry partload due to the high bypass ratio, a diffusor is established in this case between the fan and the IP-compressor. The area ratio of the diffusor seems acceptable, especially as the IP-compressor counters tendencies to flow separation at the hub.

On the basis of the control characteristics for the turbines and nozzles shown in Fig. 6 to 8 , the working points obtained under all engine running and flight conditions for the three compressors, as shown in Fig. 9 , are in the region of optimum efficiencies and desirable surge margins. It should be noted that the three compressors do not reach their aerodynamic design speeds at the same time.

- At SLS reheat operation, the fan and IP-compressor reach their design speeds together, while the HP-compressor is running at reduced speed. This is necessary in order to match the mass flows to be swallowed by the IP- and HP-compressor with the bleed off-take for the afterburner cooling open and at the same time to keep the overall pressure ratio within acceptable limits.

- At max. dry operation, the speed of the IP-compressor is lower and that of the HP-compressor high, in order to match the mass flows to be swallowed by the two compressors with the afterburner cooling air shut down.

At this point, it is emphasized that specifically in the case of the VCE, i.e., going beyond the possibilities of the C 0.25 and/or C 1.3 engines, the following control measures to improve the compatibility of the compressor section with inlet distortions are possible:

- The working line of the fan can be shifted away from the surge line without unfavourable repercussions on the turbomachines by opening the duct nozzle;

- the working line of the IP-compressor can be shifted away from the surge line by closing the LP-turbine.

In addition, it can be expected that good acceleration times can be obtained from the VCE (Cf. [18]).

Performance data

Fig. 10 shows the max. reheat thrusts and the associated specific fuel consumptions. Both the VCE and the C 0.25 engine have a flatter thrust characteristic with reheat than the C 1.3 engine, due to their low bypass ratio. In addition, at high Mach numbers the VCE turned out to have a progressively falling-off thrust characteristic, which does not appear attractive. In principle, this disadvantage could be largely eliminated by duct burning with a moderate temperature increase to 1000 K. Leaving aside the considerable complication involved, however, the deliberately introduced physical separation of afterburner and compressor section would be lost. For this reason, owing to the demands made on handling qualities of the engine, this version does not seem attractive.

The design modification of the VCE substantiated earlier to obtain a very low bypass ratio in reheat operation - which is also attractive from the point of view of engine weight and leading dimensions - allows a quite satisfactory thrust characteristic similar to that of the C 0.25 engine to be obtained.

As expected, the specific fuel consumptions of the VCE and of the C 0.25 engine with reheat are much better than that of the C 1.3 engine due to their low bypass ratios.

Fig. 11 shows the max. dry thrusts and associated specific fuel consumptions, from which the familiar relation between performance data and bypass ratio can be seen. As expected, the VCE comes close to the C 0.25 engine.

Two examples of engine performance at dry and reheated partload are shown in Fig. 12 . The transition accomplished by the VCE from the C 0.25 engine characteristic at reheat and max. dry operation to the C 1.3 engine characteristic at dry partload is evident.

Development problems

In line with the assumption that available component technology should be the starting point wherever possible, no outstanding development problems with the VCE going beyond those of current engines are to be expected, as far as the basic engine is concerned. However, the variable nozzle guide vanes of the cooled IP-turbine can certainly be considered the critical element, whereas the conditions are much better in the case of the variable nozzle guide vanes for the LP-turbine. Furthermore, the development problems to be expected on the control system must not be underestimated, particularly as 8 control variables have to be mastered. A control system managing the called-for flexibility of the engine in the presence of inlet distortions is also likely to require special efforts.

Installation influences

As shown in Fig. 5b , the VCE will swallow up to 80 % of the maximum mass flow under typical cruise conditions at sea level, whereas only 60 - 70 % can be achieved with the C 0.25 and C 1.3 engines. A substantially smaller intake spillage drag is therefore to be expected from the VCE under these conditions. The resultant increase in specific fuel consumption due to installation effects is 1 - 2 % lower for the VCE than for the C 0.25 and C 1.3 engines (Cf. [16, 19, 20]).

Regarding extreme power and bleed off-take from the HP-system, for instance, in an emergency case with one engine failed, high power can be obtained in the HP-system of the remaining engine by setting the configuration of the VCE - i.e., the geometry of the variable components - for max. dry thrust. In this way, power and bleed off-take can be expected to have minimum effects on the engine performance.

Owing to the primary nozzle of the VCE being enveloped by the duct flow, the after-body flow conditions are favourable, as shown in Fig. 13 . As a result, the generally inevitable flow separation and corresponding increased drag in subsonic and transsonic flight without reheat associated with afterbodies of conventional configuration can be avoided. However, a degree of extra friction drag is generated by the duct flow. Admittedly, the expected drop in the resulting afterbody drag can only be verified experimentally (Cf. [19, 21, 22,]).

Engine Evaluation

A first rough comparative evaluation of the three engines under consideration was made on the basis of an imaginary multi-role combat aircraft for the missions "Interception", "Air Superiority" and "Battlefield Interdiction/Close Air Support". The mission breakdowns are given in Fig. 14. For the three alternative engines compared, the weight of engine + fuel and thus the take-off gross weight were taken to be the same, each mission being a special case. As a basis of comparison, which is in principle arbitrary, it was also assumed that the mission phases marked with arrows in Fig. 14 can be achieved by engine C 1.3 with the called-for ranges.

Due to the different thrust characteristics of the three alternative engines, their required sizes are fixed by different mission phases (Cf. Fig. 14). Thus, the available reheat thrusts of these engines at the design point SL/MN = 0,8/ISA differ somewhat as shown in Fig. 15 . In order to emphasize the decisive role of the excess weight of the VCE, the thrust/weight ratio of this engine was introduced parametrically. The C 0.25 and C 1.3 engines were assumed to be specifically equally heavy in line with the current state of the art.

The lower afterbody drag expected from the VCE was not taken into consideration, i.e., the afterbody drag was taken to be the same for all three alternative engines installed. Even for the two engines C 0.25 and C 1.3, this is certainly quite a simplification.

Fig. 15 shows the ranges, as defined differently for each of the three missions, achievable with the alternative engines. If the VCE could be built to have the same specific weight as the conventional C 0.25 and C 1.3 engines, it would be at least as good as the two latter, or superior to them, depending on the mission. However, the equality or superiority of the VCE in all missions is questionable, as soon as it has a slight extra weight of 10 %.

Naturally, Fig. 15 covers only part of the evaluation which has to be carried out, especially as important evaluation criteria such as "turn rate" and "specific excessive power" must also be included. Last not least, aspects relating to operating stability and handling qualities should not be underestimated.

Conclusions

The present data are certainly still not adequate for a binding evaluation of the VCE concept described. Nevertheless, for a start the following provisional conclusions can be drawn:

- The aimed-at combination of the performance advantages of the turbojet and of the turbofan with and without reheat seems feasible.

- The improvement concerning operating stability and handling qualities in connection with the afterburner compared with current reheated turbofans was only established qualitatively but is potentially feasible.

- As a side effect, progress in compatibility with inlet distortions can be expected compared with current engines.

- The complexity to be expected in the VCE control system is considerable, but is likely to be manageable.

- The VCE described does not yet meet weight requirements. An improvement can be expected, if the engine is modified to obtain a minimum bypass ratio with reheat. A positive clarification of the weight problem is of decisive importance.

Acknowledgements

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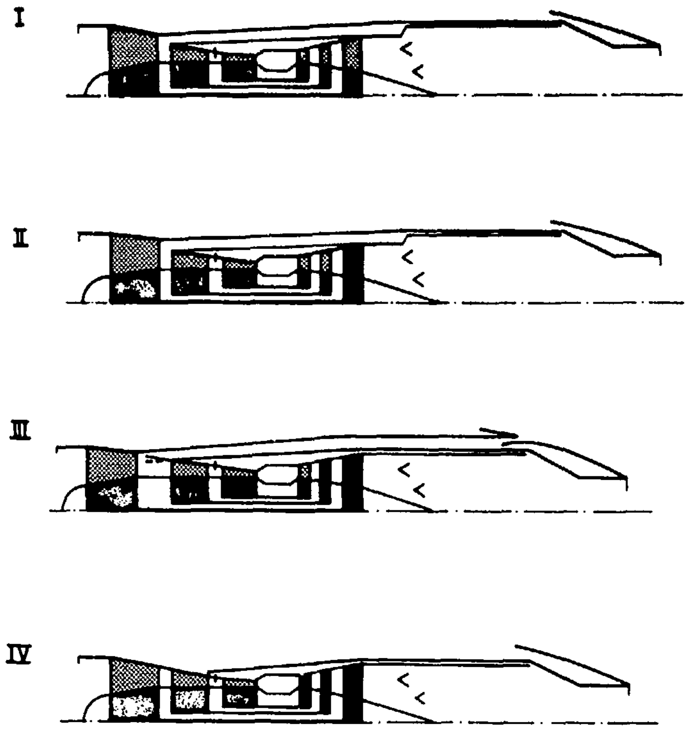


I — Fixed Cycle Mixed Flow Turbofan
- Fixed Compressors and Turbines
- IPC Bleed Control
- Variable Nozzle
- Compr. Section and AB Connected

II — Variable Cycle Mixed Flow Turbofan
- As (I) but
- Variable LPT

III — Variable Cycle Unmixed Flow Turbofan
- Variable Splitter, IPT, LPT
- Variable Primary and Duct Nozzle
- AB-Cooling Air Throttled / Shut Down
- Compr. Section and AB Separated

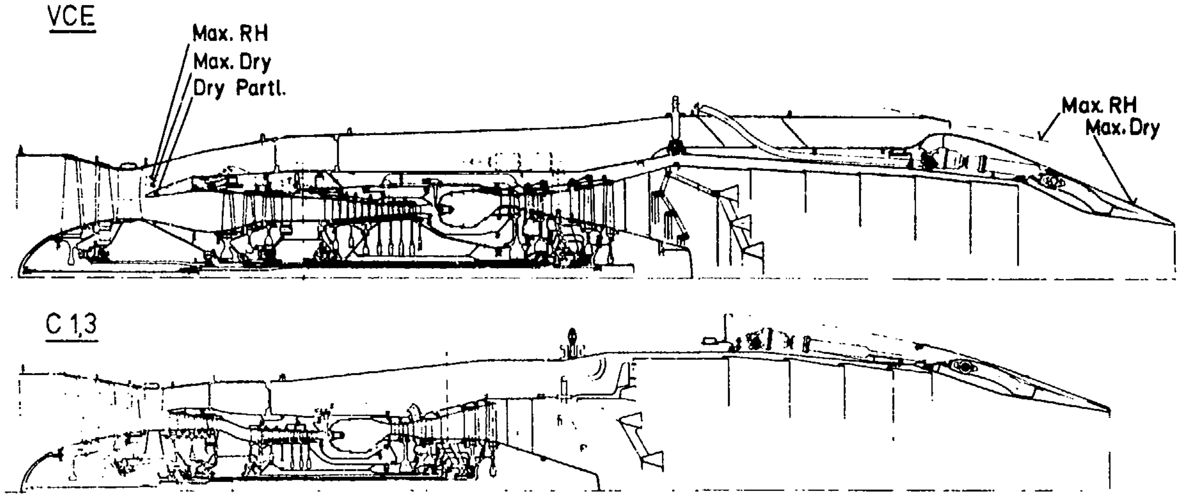
IV — Variable Cycle Turbo Jet
- Variable HPT, IPT, LPT
- AB-Cooling Air Throttled / Shut Down
- Variable Nozzle
- Compr. Section and AB Separated

Fig. 1 Derivation of VCE-Concepts
(3-Shaft Arrangements)

			Variable Cycle Engine	Current Engines (Mixed Flow)	
SL/MN = 0,8/ISA/Uninstalled			VCE	C 0,25	C 1,3
Max. RH (Max. Dry)	F	kN	80 (43,6)	80 (42,7)	80 (31,8)
Fan Massflow	M	kg/s	102	81	94
Number of Shafts			3	2	3
Turbine Entry Temperature	TET	K		1600	
Reheat Temperature	RHT	K		2050	
Bypass Ratio M_D/M_{HPC} =	BPR		0,34 (0,65)	0,25*	1,3
AB-Cooling Air Ratio	M_C/M_{HPC}		0,25	-	-
Overall Pressure Ratio	OPR			20	
Fan Pressure Ratio	FPR		1,9	3,3	2,4
IPC-Pressure Ratio	IPR		2,4 (1,9)	-	2,4
HPC-Pressure Ratio	HPR		4,4 (5,4)	6,1	3,5
Throttle Ratio of AB-Cooling Air	P_{RH}/P_{IPC}		0,57	-	-

* Duct Flow for Cooling of Afterburner only

Fig. 2 Main Design Parameters of Engines Compared



Control Variables:

VCE	C 1,3
1) Main Fuel Flow	1) Main Fuel Flow
2) Reheat Fuel Flow	2) Reheat Fuel Flow
3) IPC-Bleed (AB-Cooling)	3) IPC-Blow Off (Aer. Stability)
4) Primary Nozzle	4) Nozzle
5) Duct Nozzle	
6) IP-Turbine NGV	
7) LP-Turbine NGV	
8) Splitter	

Fig. 3 General Arrangement of VCE and Engine C 1,3

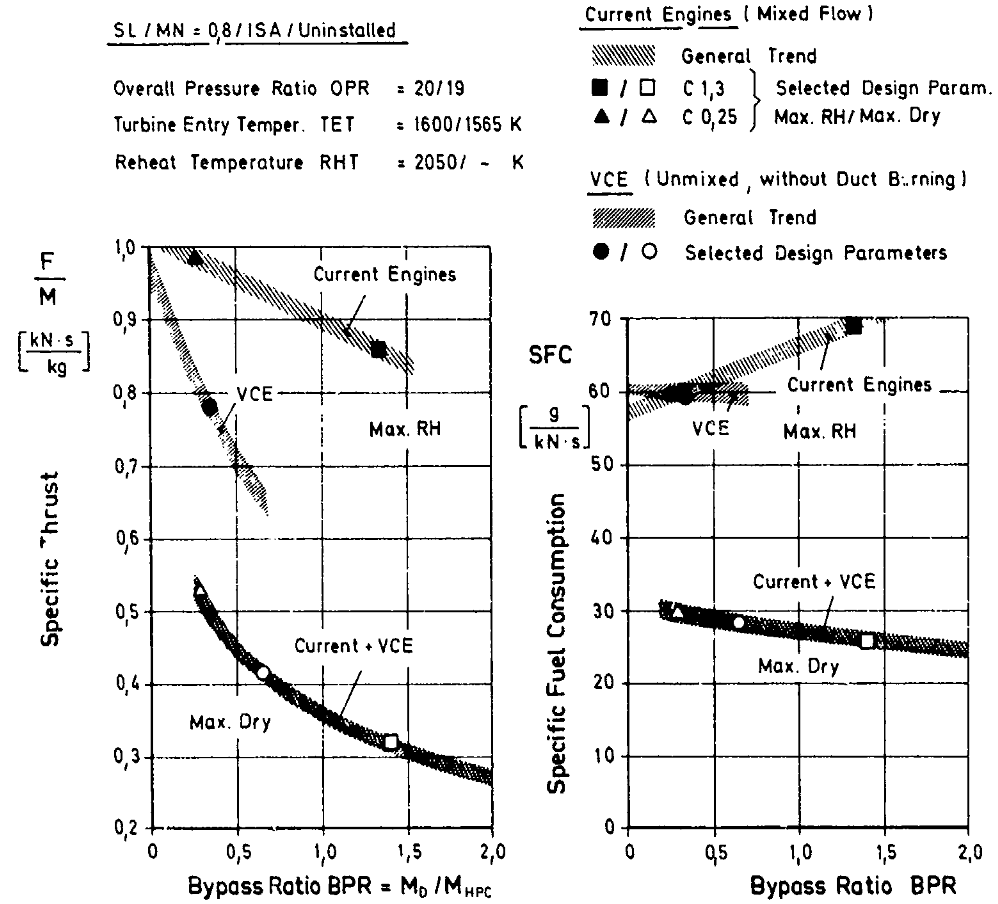


Fig. 4 Selection of Design Cycle Parameters

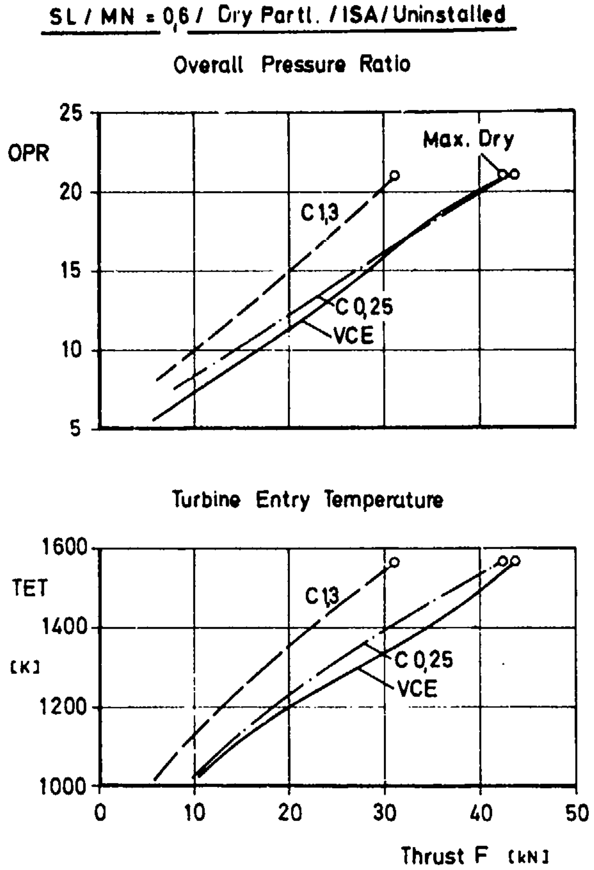


Fig. 5a Dry Partload Cycle Data

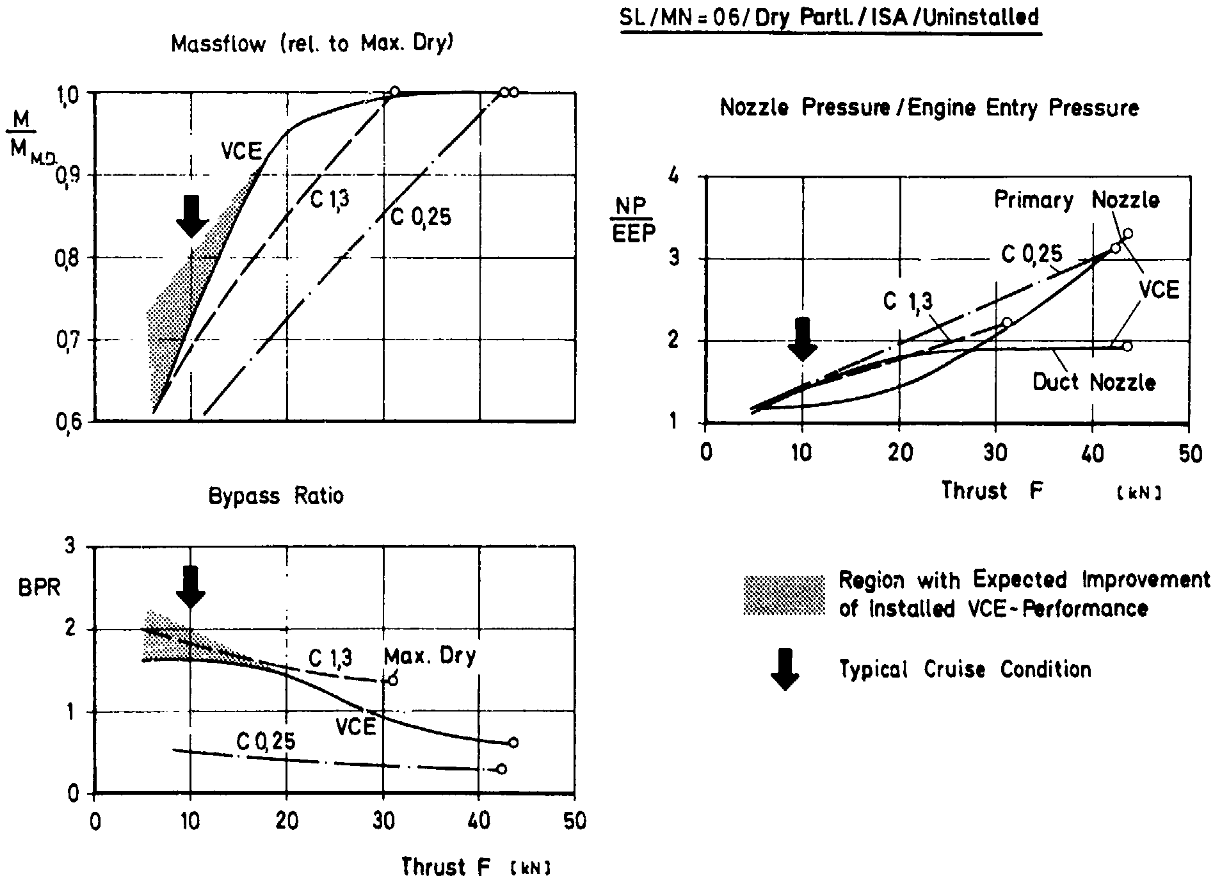


Fig. 5b Dry Partload Cycle Data, Continued

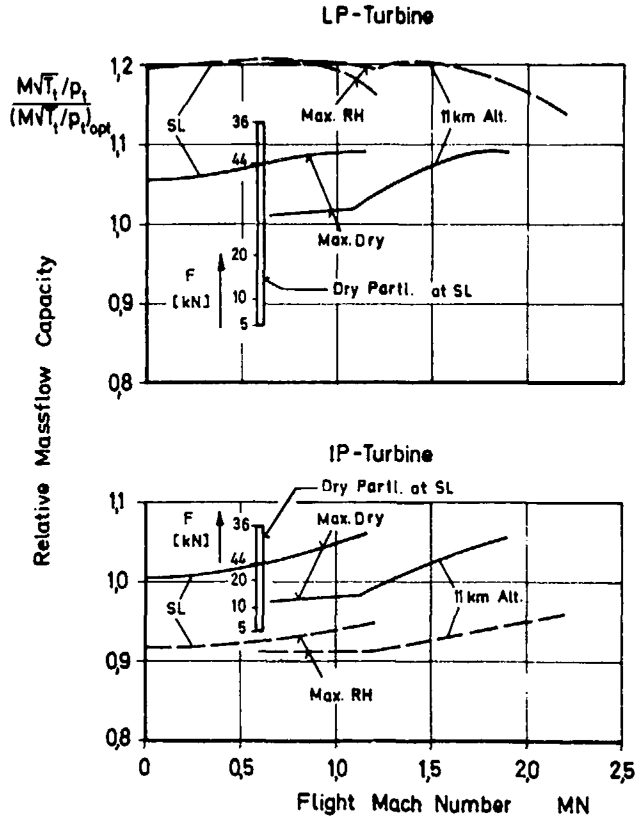


Fig. 6 Relative VCE Turbine Massflow Capacity

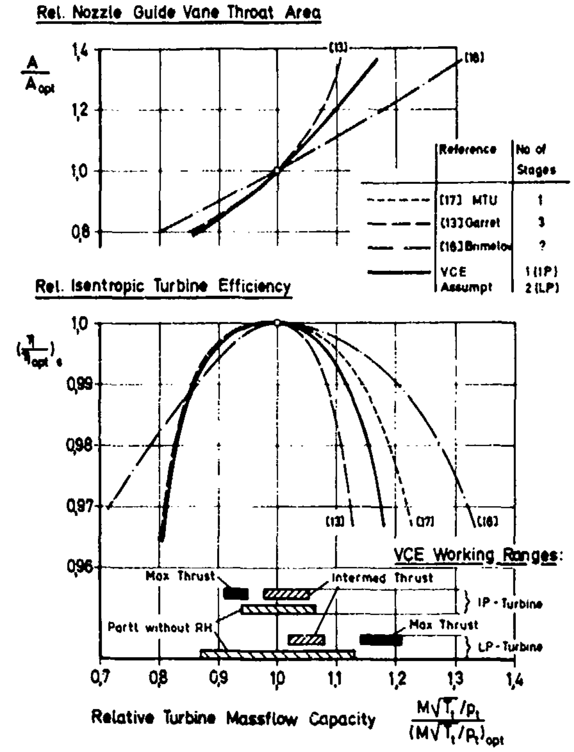


Fig. 7 Turbines with Variable First Nozzle Guide Vanes

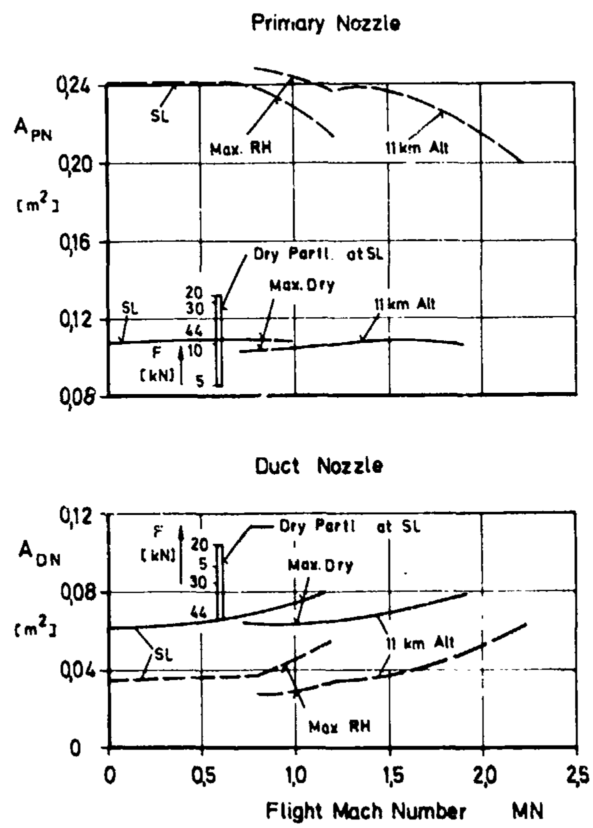


Fig. 8 VCE Nozzle Throat Areas

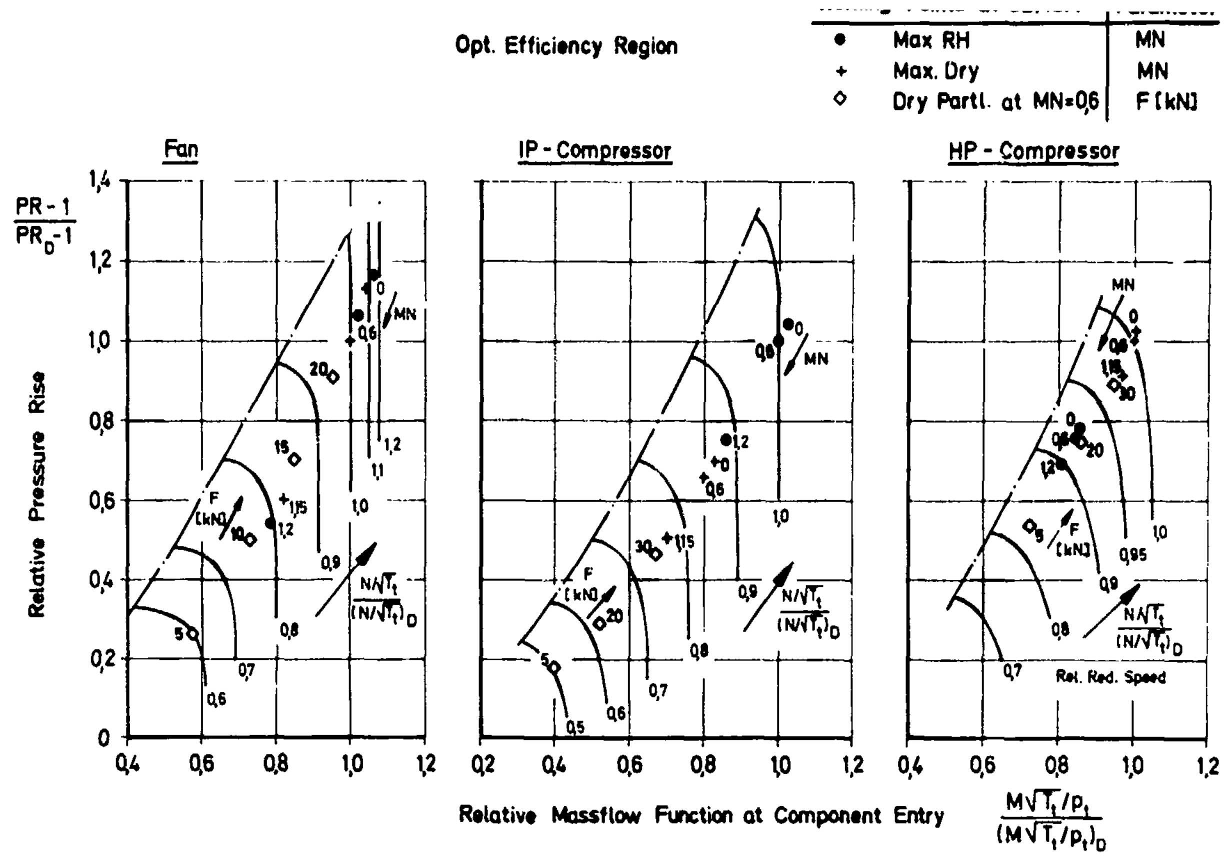


Fig. 9 VCE Compressor Characteristics with Working Points

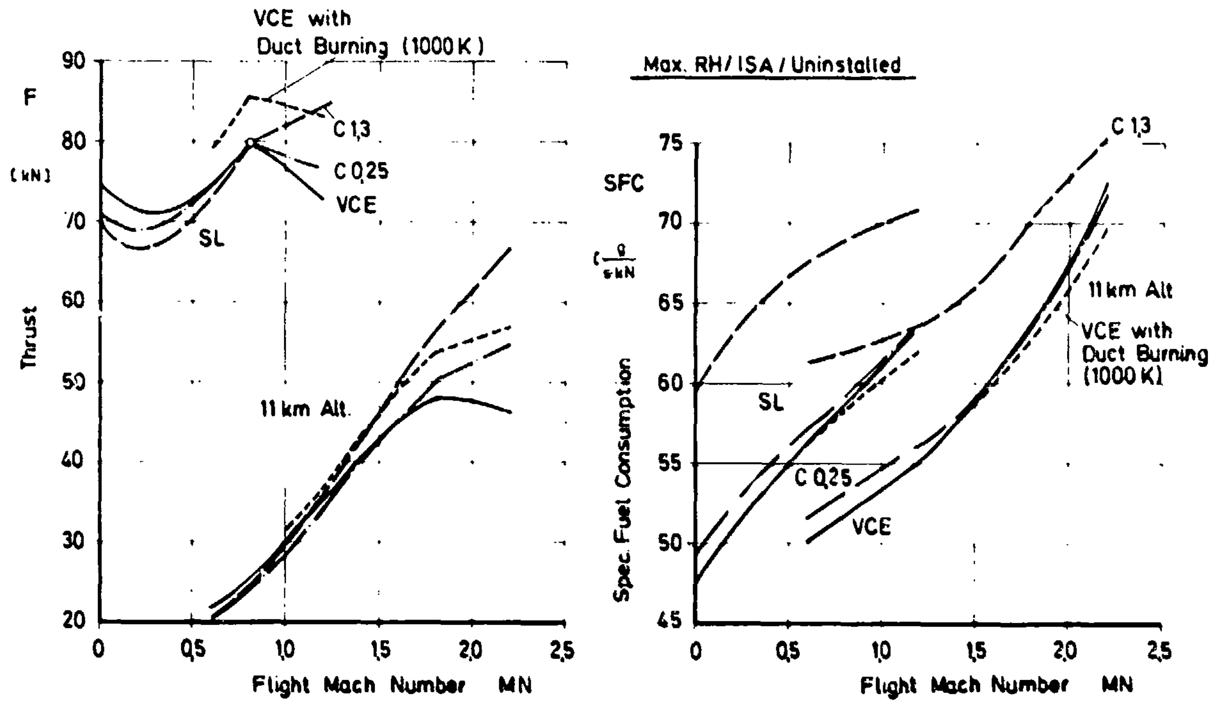


Fig. 10 Comparison of Thrust and SFC at Max. RH Rating

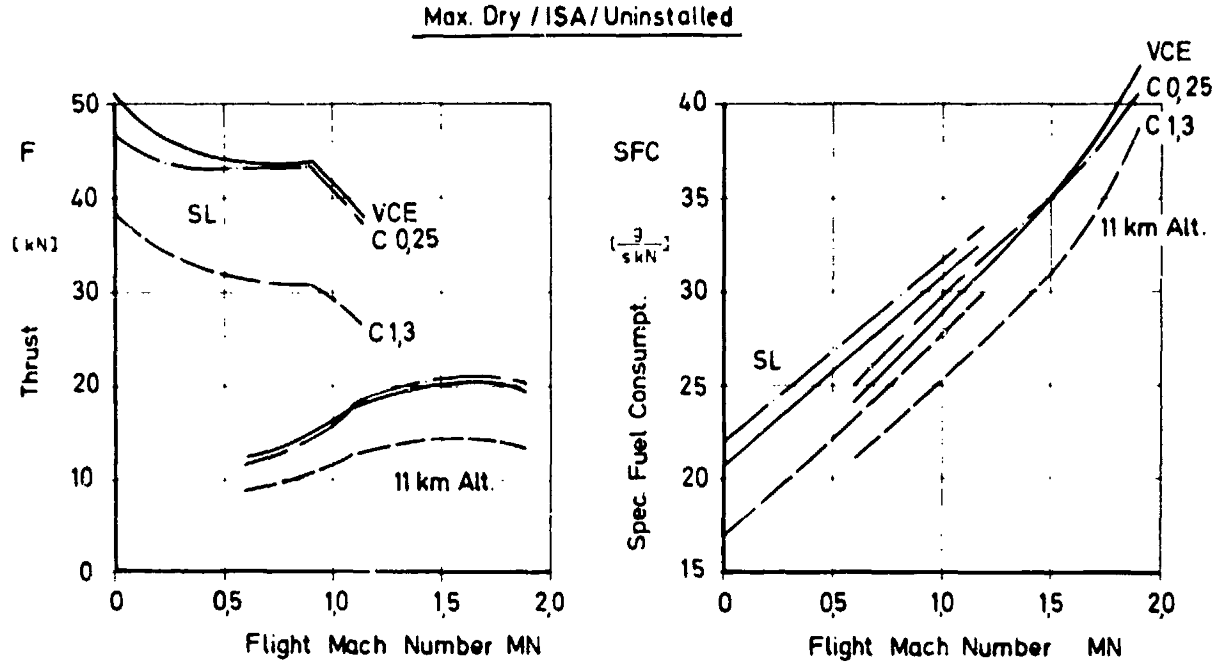


Fig. 11 Comparison of Thrust and SFC at Max. Dry Rating

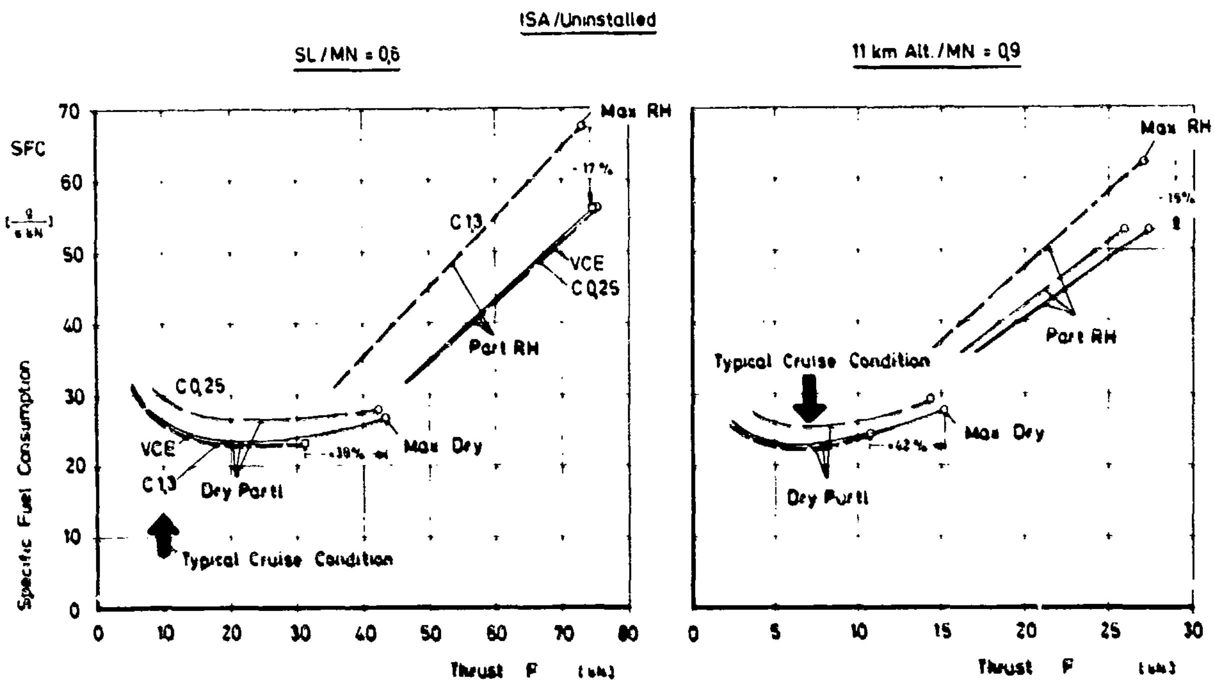


Fig. 12 Comparison of Partload SFC-Characteristics

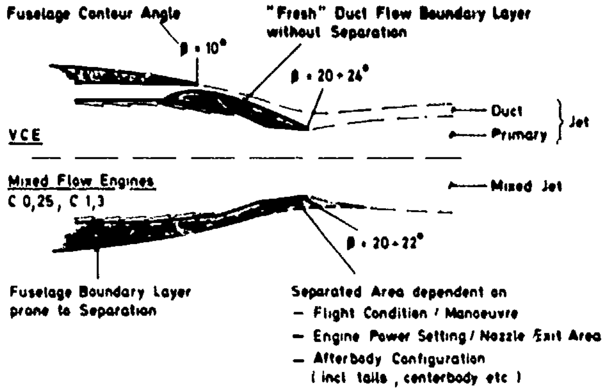


Fig. 13 Subsonic Afterbody Flow Pattern (Dry Rating)
Comparison of Mixed Flow Engine and VCE Installation

Breakdown of Missions as a Basis for Engine Evaluation

Mission	Mission Phase		Engine Power Setting	Alt. km	MN	Range Segm. km	TOGW t
Interception	Take Off		Max. RH	0	0 - 0.2		19
	Acceleration		Max. RH	0	0,2 - 0.9		
	Climb		Max. RH	0-13	0,9-1,6-1.3		
	Combat, 3 min	2) 4)	Max. RH	13	1,3 - 1,6		
	Cruise Back	3)	Dry Partl.	11	0,9	[illegible]	
Air Superiority	Take Off		Max. RH	0	0 - 0.3		18
	Acceleration		Max. Dry	0	0.3 - 0.6		
	Climb Out	1)	Max. Dry	0-7.6	0.6	[illegible]	
	Cruise Out	2)	Dry Partl.	7.6	0.6		
	Combat, 4 min	4)	Max. RH	3.0	0.79		
	Climb Back		Max. Dry	3-7.6	0.6	[illegible]	
	Cruise Back	3)	Dry Partl.	7.6	0.6		
	Loiter		Dry Partl.	0	0.3		
Battlefield Interdiction/ Close Air Support	Take Off		Max. RH	0	0 - 0.3		20
	Acceleration		Max. Dry	0	0.3-0.6		
	Cruise Out, 17 min		Dry Partl.	0.08	0.6		
	Dash Out	2)	Dry Partl.	0.15	0.9	[illegible]	
	Dash Back	2)	Dry Partl.	0.15	0.9	[illegible]	
	Cruise Back, 18 min		Dry Partl.	0.08	0.6		

1) Condition for Sizing Engine C 1.3
2) Condition for Sizing Engines C 0.25 and VCE
3) Range influenced by Alternative Engines Installat.
4) Combat Time with Engine C 1.3

Same Combat Flight Manoeuvres with Alternative Engines

Fig. 14 Breakdown of Missions as a Basis for Engine Evaluation

	Interception	Air Superiority	Battlefield Interdiction Close Air Support
RH - Time / Miss. - Time	34 ÷ 43 / 43 %	6 %	0,6 %
	Cruise Back Range	Mission Range	Dash Range

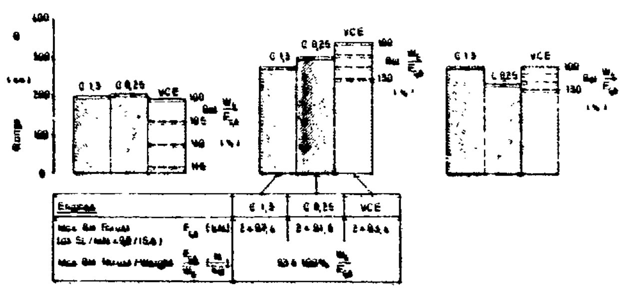


Fig. 15 Multi Role Combat Aircraft Performance
Influence of Alternative Engine Installations on Range

DISCUSSION

J.F.Chevalier

La conférence du Dr Grieb était très précise, et il est probable qu'il faudrait étudier à fond la dossier pour savoir si on arrive aux mêmes conclusions que lui. Mais je pense quand même qu'il peut y avoir des remarques à faire sur sa conférence.

Personnellement, j'ai une petite remarque à faire. Un peu de détails. Vous avez attaché beaucoup d'importance à l'air de refroidissement de la post-combustion et sur les différences de performances que l'on peut attendre suivant la façon dont on traite cet air de refroidissement. Personnellement j'ai été un peu surpris, je ne m'attendais pas à ce que ça intervienne autant dans les différences entre moteurs.

Author's Reply

We found that the influence of the throttling of the afterburner cooling air on performance is very small. But from experience with reheated turbofans it is known that the compressor section is very sensitive to the afterburner light up and blow off. This can be prevented by separation of afterburner and duct flow and throttling of the afterburner cooling air, i.e. that the engine works like a reheated turbojet.

P.A.Kramer

Why is it important, as pointed out in the lecture, to have a separation between afterburner and compressor?

Author's Reply

During afterburner light up and blow off there is no proper combustion in the afterburner and it can well be that for very short time the efficiency of combustion is much better or worse than the scheduled efficiency. That means, during this process, the temperature in the afterburner, i.e. the *real* reduced mass flow function $M\sqrt{T}/P$ can be very different from the *scheduled* one. This means a change of the nozzle throat is necessary, but this is not realized by the control system. A pressure step either upwards or downwards is encountered which feels the compressor. In case of the turbofan the higher the bypass ratio the higher is the influence of such a process on the fan working line.

E.Willis

To what level of detail did you estimate the weight of the VCE? i.e., were allowances made for weight and/or performance penalities due to unique components and features such as the variable splitter? I am not sure whether the parameter variation of VCE weight in Figure 15 represents the expected range of the penalties due to unique features or merely a range of uncertainty applied on top of your first estimate of the penalties?

Author's Reply

You have seen the slide No 3 where both general arrangements are shown. Both engines were designed to the same thrust at sea level, Mach 0.8. Engine C 1.3 has a bypass ratio 1.3 and the VCE has a bypass ratio .34/.65. We designed both engines properly and estimated the weight of all the components and found that the VCE is much heavier than the turbofan engine. We found also that not only the structure but also the thermodynamic cycle of the VCE plays an important role on this high specific weight. We mean that the minimum excess weight by the variable components and more complicated arrangement will be at least 10 to 15%.

Habrard

Vous avez éliminé le moteur No 2 du point de vue du fonctionnement du fan avec réchauffe. Est-ce que le moteur No 3, qui finalement s'avère intéressant avec chauffe du plus froid, ne présente par les mêmes désavantages at finalement, ne rendraite-il pas le moteur deux aussi intéressant?

Author's Reply

I think, if only performance is important, concept 2 and concept 3 should be put into the same box. But if you take into account also stability problems in connection with the afterburner then you have to separate them. We defined our requirements with respect to performance and to stability also and this is the reason why we left engine No 2.

VARIABLE CYCLE ENGINES FOR V/STOL FIGHTERS

by

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SUMMARY

Future airbreathing engines will require lower fuel consumption and greater operational flexibility than obtained with present engines because of low worldwide fuel reserves and the expanded requirements projected for advanced aircraft. Turbofan engines operate efficiently at subsonic Mach numbers and turbojet engines operate efficiently at supersonic Mach numbers, but an aircraft which requires efficient operation in both speed regimes needs the best of both engine types - this is not possible with current production engines. In addition to diverse flight Mach numbers, V/STOL aircraft require a large variation in thrust levels. At takeoff and landing, the engine must produce a very high level of thrust and thus, is oversized for many of the forward flight conditions. Variable Cycle Engines (VCE) offer a potential approach to the solution of these problems. In the Navy Variable Cycle Engine Selection Program, several VCE's were evaluated in a Navy V/STOL fighter aircraft. The results indicate that some VCF concepts offer cycle flexibility which will greatly benefit the Navy in both reduced fuel consumption and lower aircraft take-off-gross-weight.

1. INTRODUCTION

Supersonic V/STOL fighters require propulsion systems which produce thrust well in excess of aircraft weight, for powered lift and control, and which can be integrated into an aerodynamically efficient airframe. The powered lift requirements, plus the flight performance requirements of Navy fighter missions, result in extensive compromises in the cycle and scheduling of fixed cycle engines. These compromises have resulted in high take-off-gross-weight and relatively poor payload and range performance in many V/STOL aircraft designs. Because of their inherent operational flexibility, variable cycle engines are being studied for V/STOL aircraft. Such engines can potentially reduce the compromises necessary with a fixed cycle engine and, therefore, reduce aircraft size and improve performance.

The "Variable Cycle Engine Selection Program", sponsored by the Naval Air Propulsion Test Center (NAPTC), seeks to determine the impact of variable cycle engines on advanced V/STOL multi-mission fighters. This paper presents a program description, a summary of current results, and the conclusions drawn from the initial phase of the program.

2. VCE SELECTION PROGRAM DESCRIPTION

In June 1975, the Navy initiated the "Variable Cycle Engine Selection Program" to assess the potential benefits from variable cycle engines installed in V/STOL fighters. This program is a complementary effort to a United States Air Force VCE selection program which will determine VCE impact on the size, operational flexibility, and cost of supersonic multi-mission CTOL fighters, Reference 1. Figure 1 lists a number of the critical engine requirements for fighters, including, efficient fuel utilization at both supersonic and subsonic conditions, high thrust for combat, and good airframe integration characteristics. Navy V/STOL fighters, however, have additional engine requirements, also listed in Figure 1, which are equally important in achieving an effective design. A number of payoffs may be derived from the use of variable cycle engines in V/STOL fighters. For example, the utilization of variable airflow features to produce the high thrust required for take-off and landing could eliminate the need for direct lift engines or lift fans. As a result, overall propulsion system development and maintainence costs would be sharply reduced.

FIGURE 1
VCE APPLICATION TO V/STOL

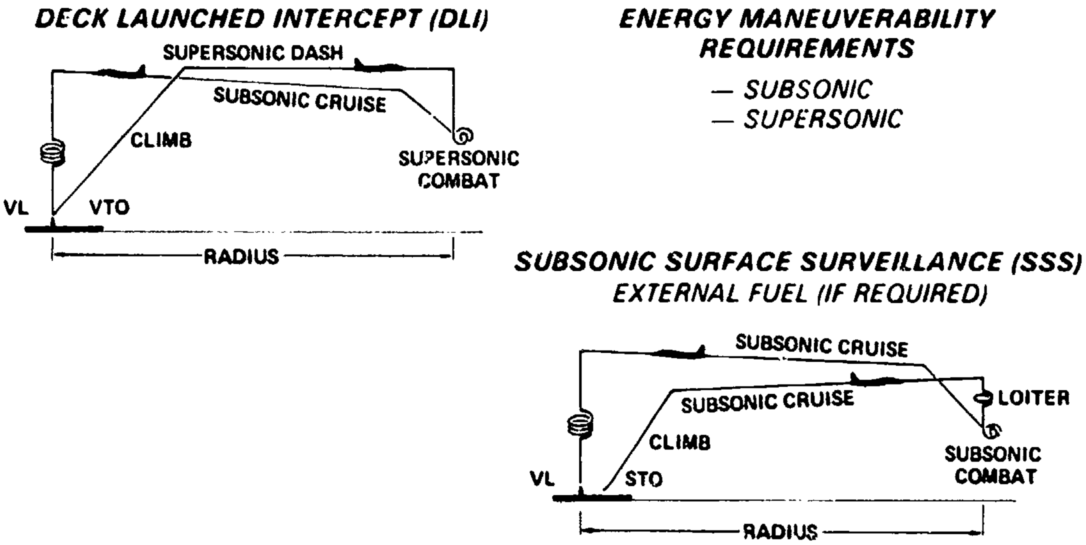
- **MULTIMISSION AIRCRAFT PROPULSION REQUIREMENTS**
  - EFFICIENT SUPERSONIC DASH
  - EFFICIENT SUBSONIC CRUISE
  - EFFICIENT LOITER
  - HIGH COMBAT THRUST
  - IMPROVED INSTALLED PERFORMANCE
- **SUPERSONIC V/STOL**
  - SHORT TIME HIGH V/STOL THRUST
  - RAPID THRUST RESPONSE FOR HOVER
  - HIGH BLEED RATES FOR CONTROL
  - MODERATE EFFLUX

Some VCE designs can also be used to obtain the rapid thrust response required for hover control. Such engines can eliminate the need for thrust spoiling, and thus, reduce the size of the powered lift system. Other VCE concepts can produce high bleed rates for attitude control without oversizing the main propulsion system. The objective of this program is to identify high payoff VCE design concepts and operational characteristics for multi-mission Navy V/STOL fighters. The following paragraphs briefly describe mission requirements used, the evaluation approach and the VCE concepts which have been evaluated.

2.1 Mission Requirements

Meaningful evaluations of engine impact on aircraft size, weight, and performance must be conducted using missions which are representative of future requirements. The two principal missions used for this study are shown in Figure 2. The Deck Launched Intercept (DLI) mission emphasizes high power performance

FIGURE 2
DESIGN MISSIONS



for vertical takeoff, maximum power climb, supersonic dash, and supersonic combat. The Subsonic Surface Surveillance (SSS) mission emphasizes low power fuel utilization efficiency in the long range subsonic cruise out and back, and long loiter on station. The DLI design mission is used to define aircraft internal fuel volume. The SSS mission is accomplished by adding external fuel and performing a short takeoff instead of a vertical takeoff. In both cases, aircraft performance requirements such as acceleration time, maneuverability, specific excess power, and combat ceiling are quite demanding. Superimposing the performance, VTOL, and STOL requirements produces an aircraft with a high thrust to weight ratio. Thus, the aircraft requires an engine with high thrust capability for performance, good high power fuel consumption characteristics for supersonic cruise and combat, and good low power fuel consumption and aircraft installation characteristics for subsonic cruise and loiter conditions.

2.2 Program Approach

The "Variable Cycle Engine Selection Program" is a three-phase effort covering approximately thirty (30) months as shown in Figure 3. Phase I of the program was six months long and was completed in December 1975. Phase I was aimed at identifying VCE payoff potential and selecting complementary aircraft and high payoff VCE concepts for subsequent detailed evaluations in Phases II and III.

Phase II includes development of a computerized V/STOL aircraft sizing and performance procedure, and detailed definitions of the engines and aircraft selected in Phase I. In Phase III, the VCE/airframe design and VCE payoffs will be identified and experimental programs aimed at demonstrating critical VCE components will be defined and recommended.

The Phase I procedures and results are presented in this paper. As illustrated in Figure 4, sensitivities were used in Phase I to estimate the aircraft take-off-gross-weight (TOGW) required to achieve mission and performance objectives for each of six candidate VCE concepts. The sensitivity of TOGW to propulsion system performance for each key mission segment and to propulsion system physical characteristics were included. The performance parameters for which sensitivities were developed included net thrust, fuel flow, inlet drag, and aft-end drag. The physical characteristics included engine weight and engine size.

The approach to be used for Phases II and III is outlined in Figure 5. Phase II involves the modification of the engine/airframe evaluation procedure, developed during the USAF VCE selection program, to permit analysis of V/STOL aircraft and development of a reference aircraft design with advanced technology fixed cycle engines. The reference aircraft will be used in Phase III to determine the benefits obtained by using variable cycle engines in Navy V/STOL fighters. To ensure a valid basis for the evaluations, the reference aircraft design will be sized to the minimum TOGW that can achieve mission and performance requirements.

FIGURE 3
NAVY V/STOL PROGRAM FLOW

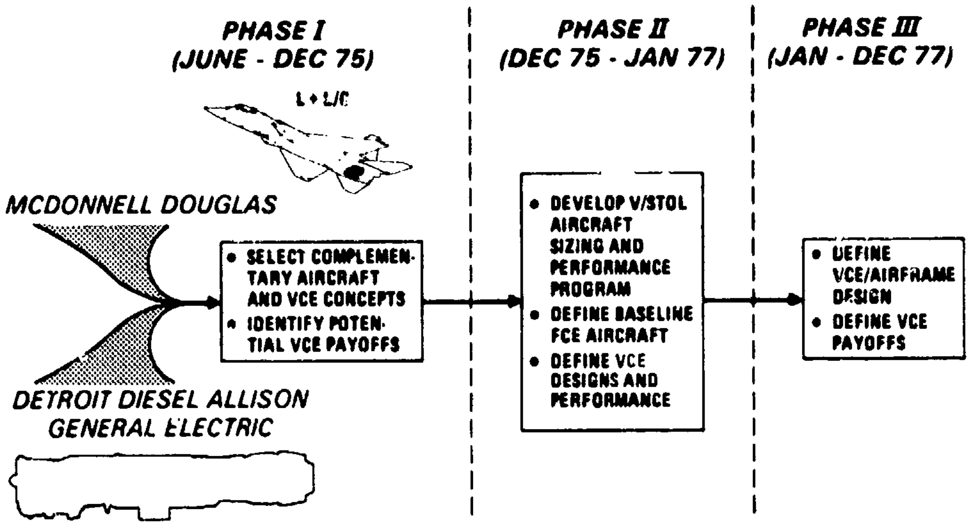


FIGURE 4
PROGRAM APPROACH - PHASE I

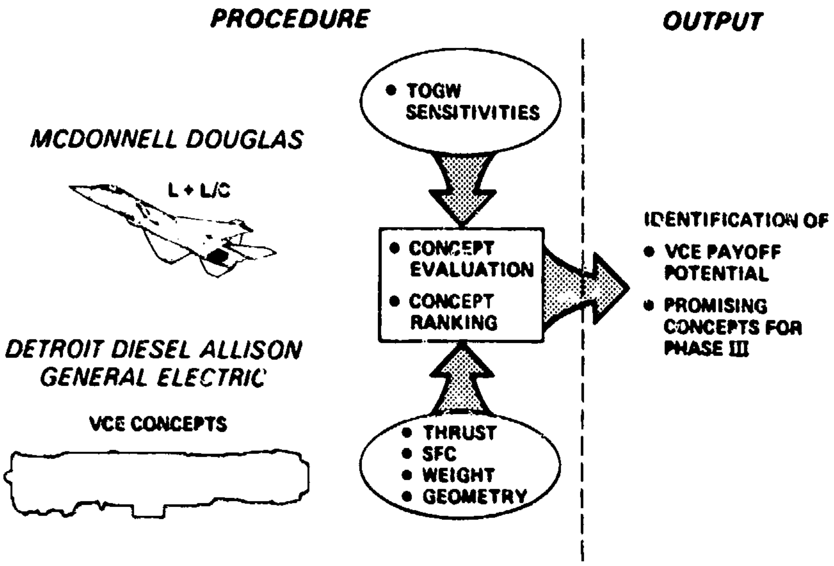
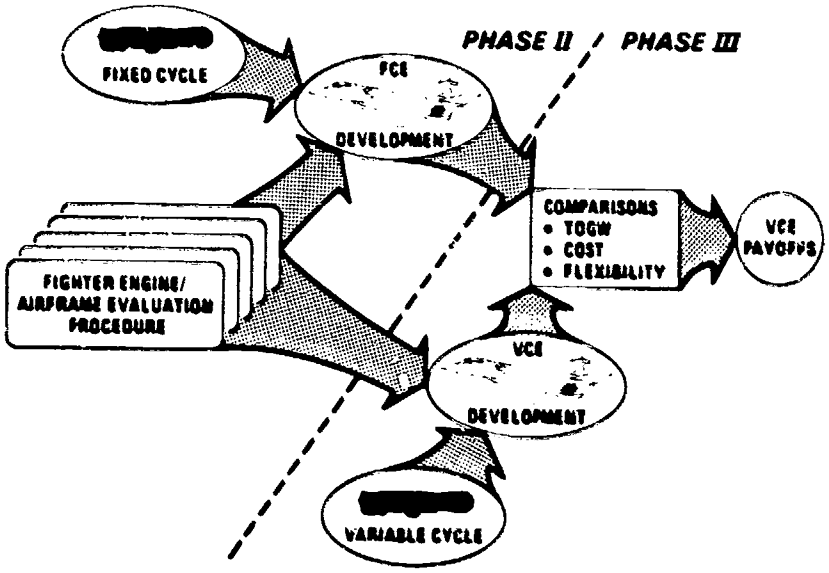


FIGURE 5
PROGRAM APPROACH - PHASES II & III



The engine company selected for Phase II (General Electric) will provide the fixed cycle engine data for the development of the reference aircraft and will investigate the design, operating, performance, and life cycle cost characteristics of the two most promising VCE concepts from Phase I.

In Phase III, McDonnell Douglas will define aircraft for the two selected VCE concepts. Using the engine/airframe evaluation procedure, the aircraft size, weight, and performance relationship to engine and airframe design variables will be defined for each VCE concept. The minimum TOGW aircraft capable of achieving the design mission and performance will then be identified and compared with the reference aircraft of Phase II in terms of TOGW, life cycle cost, alternate mission performance, non-standard d@y performance, and powered lift system operation.

General Electric will define the size, weight, performance, and cost characteristics of the selected VCE concepts. As the vehicle and engine independent design parameter sensitivities become available from McDonnell Douglas, refinement of the VCE concepts may be possible by altering cycle decign points and component operating limits. When the refinement is completed and the capabilities of the VCE's established, the technical feasibility of VCE components will be determined and development recommendations made to the Navy.

2.3 Phase I VCE Concepts

In Phase I, VCE data were provided by Detroit Diesel Allison and General Electric. Figure 6 is a partial listing of the characteristics of the candidate engines. Detroit Diesel Allison's engines were designed with axisymmetric V/STOL exhaust nozzles. General Electric defined engines with both axisymmetric and two-dimensional V/STOL exhaust nozzles. The two-dimensional nozzles were designed to permit augmentation during both forward and vectored thrust operation and are designated as the Augmented Deflector Exhaust Nozzles (ADEN).

FIGURE 6
CANDIDATE VCE CONCEPTS

	AXISYMMETRIC V/STOL NOZZLES						TWO-DIMENSIONAL V/STOL NOZZLES	
	DETROIT DIESEL ALLISON				GENERAL ELECTRIC			
DESIGN CYCLE CHARACTERISTICS	VGT TF	VGT DRV TJ	PARALLEL TURBINE TF	REVERSE PITCH VGT TF	MODULATING BYPASS TF		MODULATING BYPASS TF	MODULATING BYPASS TF AND REMOTE LIFT SYSTEM
FPR	2.9	-	2.9	FWD = 1.4 AFT = 2.3	4.0	3.5	4.0	4.0
OPR	21	11	21	HP = 7.25	20	25	20	20
BPR	1.35	-	1.7	0.4 - 4.5	0.5	0.95	0.5	0.5 (0.9 VTO)

The four VCE designs provided by Detroit Diesel Allison covered a broad range of design complexity and performance potential. The first concept was a variable geometry mixed flow afterburning turbofan. Variable geometry was included in the fan, compressor, and both high and low pressure turbines. The variable geometry features were aimed at improving subsonic fuel consumption characteristics. The Detroit Diesel Allison vertical thrust nozzles were located upstream of the augmentor thereby precluding augmentation during vectored thrust operation.

The second Detroit Diesel Allison concept was a dry variable geometry turbojet. Variable geometry was included in both the compressor and turbine to obtain improved subsonic fuel consumption to complement the turbojet's inherently superior fuel consumption at supersonic conditions.

The third Detroit Diesel Allison concept was the bypass burning parallel turbine turbofan engine. This engine incorporated a bypass duct combustor and a low pressure turbine which extended into the bypass duct. The bypass and primary streams were mixed to simplify thrust vectoring for V/STOL operations. Variable geometry was included in the low pressure turbine and mixer. In this engine, the bypass combustor was operated at high thrust conditions and the bypass duct turbine was used to maintain high airflows. The bypass combustor increased engine specific thrust and provided good supersonic fuel consumption characteristics. For subsonic operation, the bypass burner was turned off, and essentially no power was derived from the bypass portion of the turbine.

The fourth Detroit Diesel Allison concept evaluated was a variable geometry turbofan with a reverse pitch .an stage, remotely located upstream of the primary engine. This concept was configured to eliminate the need for lift engines, or fans, for vertical takeoff. For the vertical takeoff mode, the reverse pitch fan was used to pump airflow upstream of the engine. That flow was then vectored to obtain lift forward of the aircraft center of gravity. The reverse pitch fan was driven by the low pressure turbine of the turbofan engine in both the reverse mode (for lift) and conventional mode, where the reverse pitch fan supercharged the turbofan engine.

All the General Electric VCE designs were defined using one basic concept; the modulating bypass turbofan engine. These engines were designed with a modulating fan for airflow characteristics with various flows and discharge pressures. Variable geometry was included in the fan, compressor, turbine, and mixer. For high specific thrust operating conditions, the fan discharge pressure was maximized. During subsonic cruise, the fan produced a high operating bypass ratio. At these conditions, fan pressure ratio decreases and uninstalled fuel consumption improved, relative to a fixed cycle engine. Inlet flow could be modulated during subsonic flight to reduce installation losses.

Two General Electric concepts used axisymmetric exhaust nozzles. A third concept was the same engine design as the first, but with an ADEN exhaust nozzle. The fourth candidate concept incorporated the ADEN exhaust nozzle and an flow modulating fan which supplied flow to a remote lift system (RLS). The RLS was used to provide VTO thrust and, thus, eliminate the need for direct lift engines. This concept has the potential to significantly reduce the complexity of the powered lift system of a V/STOL fighter.

3. PROGRAM RESULTS

A systematic screening procedure was used to obtain a preliminary, but consistent, measure of the relative payoff for each of the candidate VCE concepts in advanced Navy V/STOL fighters. The following paragraphs briefly summarize this procedure and present the results of the engine screening for the DLI and SSS design missions described above. Results are presented for evaluations of VCE concepts with both axisymmetric and two-dimensional V/STOL nozzle installations.

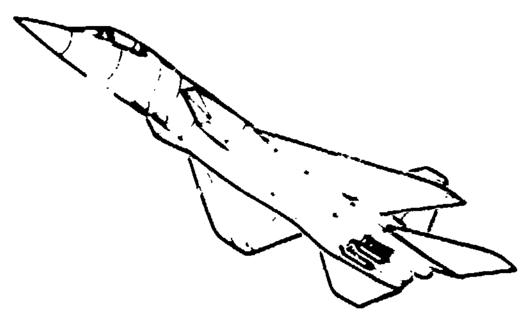
3.1 Screening Procedure

Aircraft TOGW sensitivities to engine size, weight and performance parameters were used to evaluate the candidate VCE concepts. These sensitivities were derived using a McDonnell Douglas advanced Navy V/STOL fighter design. That design, shown in Figure 7, was specifically optimized to achieve the Navy DLI and SSS missions using a fully integrated direct lift plus lift-cruise propulsion system. The design characteristics of the advanced technology, fixed cycle direct lift plus lift-cruise engines (FCE) are also noted in Figure 7. This aircraft was in the 30,000 lb TOGW class when sized to achieve the DLI radius and performance requirements and is capable of VTO with full internal fuel and DLI armament. With external fuel and SSS mission armament, the aircraft TOGW was in the 40,000 lb class, and was capable of achieving the SSS mission and STO requirements. TOGW sensitivities were determined at each critical mission and performance segment of the DLI and SSS missions.

FIGURE 7
LIFT + LIFT-CRUISE
V/STOL AIRCRAFT DESIGN

ENGINE DESIGN PARAMETERS

	LIFT CRUISE	DIRECT LIFT
OPR	23	11
SPR	0.7	--



TOGW CLASS

DLI (INTERNAL FUEL) ≈ 30,000 LB
SSS (EXTERNAL FUEL) ≈ 40,000 LB

The use of the TOGW sensitivities to assess the impact of the VCE concepts on DLI TOGW is illustrated in Figure 8. All increments in VCE fuel consumption, thrust, geometry and weight were referenced to the FCE lift-cruise engine sea level static thrust level.

The impact of VCE specific fuel consumption (SFC) on TOGW was evaluated at the DLI cruise, dash, and combat flight conditions using the same net propulsive force (NPF) (engine net thrust minus total inlet drag and throttle dependent aft-end drag) as required to perform those mission segments with the FCE. The VCE installed SFC was determined at each mission segment and the sensitivities used to determine impact on TOGW, e.g., $\Delta TOGW$ Fuel Consumption = $\frac{\partial TOGW}{\partial SFC}\left[SFC_{VCE} - SFC_{FCE}\right]$. The total TOGW variation caused by the difference between FCE and VCE SFC is equal to the summation of TOGW increments for each of the mission segments. The impact of VCE thrust characteristics on TOGW, the VCE NPF was defined at the subsonic and supersonic energy maneuverability flight conditions. Net propulsive force in excess of that required to achieve the desired performance indicates that the VCE size can be reduced. The sensitivities were then used to identify the resulting TOGW reduction.

Maximum engine diameter was used to assess the impact of VCE size on TOGW. Nozzle/aft-end drag increments were obtained from empirical data correlations by defining maximum engine/fuselage cross-section area ratio for each VCE candidate. The sensitivities were then used to assess TOGW increments caused by these geometry dependent drag increments at the thrust and fuel sizing mission segments.

The propulsion system weight TOGW increment was computed by multiplying the sensitivity times the change in lift-cruise and direct lift engine weight. This weight increment was determined with the engines sized to produce the same total lift as in the FCE powered aircraft. The same direct lift engine design was used in both the VCE and FCE aircraft.

The SSS mission was performed by adding external fuel tanks to the aircraft, which was sized to accomplish the DLI mission with internal fuel. Consequently, the difference between SSS and DLI TOGW was equal to the difference between SSS and DLI armament plus the external fuel and tank weight. There is, then, a direct relationship between DLI and SSS TOGW. This empirical relationship, $\Delta TOGW_{SSS} = 1.6\ \Delta TOGW_{DLI}$, was used to compute the change in SSS TOGW due to the change in DLI TOGW determined for each VCE. TOGW sensitivities were also used to determine the SSS TOGW changes resulting from VCE SFC and engine geometry dependent drag at the SSS cruise and loiter mission segments.

FIGURE 8
ENGINE EVALUATION PROCEDURE USING TOGW SENSITIVITIES

$$\Delta TOGW = \Delta TOGW_{\text{FUEL CONSUMPTION}} + \Delta TOGW_{\text{THRUST}} + \Delta TOGW_{\text{GEOMETRY DEPENDENT DRAG}} + \Delta TOGW_{\text{ENGINE WEIGHT}}$$

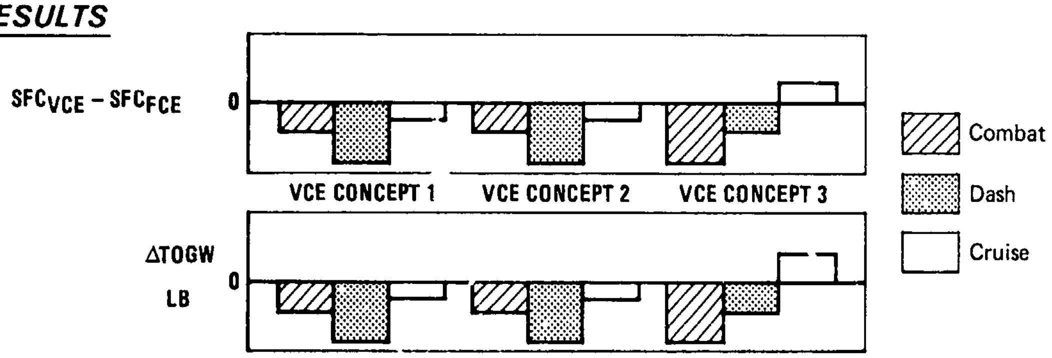
EXAMPLE: DLI MISSION

$$\Delta TOGW_{SFC} = \Sigma \left[\Delta TOGW_{\text{FUEL CONSUMPTION}} \right]_{\text{COMBAT, DASH, CRUISE}}$$

WHERE

$$\Delta TOGW_{\text{CRUISE FUEL CONSUMPTION}} = \frac{\partial\ TOGW}{\partial\ SFC_{CRUISE}} \left[SFC_{VCE} - SFC_{FCE} \right]_{\text{CRUISE}}$$

RESULTS



3.2 Evaluation Results for Axisymmetric Nozzle Installations

Six VCE concepts were evaluated with axisymmetric V/STOL nozzles (four Detroit Diesel Allison and two General Electric engines). With the axisymmetric nozzles, it was assumed that the DLI VTO was accomplished with non-augmented power, as was the case with the FCE aircraft. A summary of the DLI mission TOGW sensitivity evaluation is presented in Figure 9. Each concept was evaluated to determine the effects of maneuvering thrust, SFC, geometry dependent drag, and engine weight on TOGW. All six engines achieved TOGW reductions which were derived from their maneuvering thrust and SFC characteristics. It should be noted that none of the VCE concepts exhibited reductions in subsonic cruise SFC, but all six achieved reduced SFC at the supersonic dash and combat flight conditions. Only one VCE, the Detroit Diesel Allison VGT TF, resulted in an engine weight payoff and two exhibited prohibitive engine weight penalties. Finally, only the Detroit Diesel Allison VGT dry TJ produced an engine geometry dependent drag penalty. The total TOGW increment achieved by the VCE concepts is equal to the sum of the four separate increments shown in Figure 9. The Detroit Diesel Allison VGT TF clearly produced the largest potential DLI TOGW reduction of the six engines evaluated.

The results of the SSS mission TOGW sensitivity evaluation are summarized in Figure 10. As noted above, three TOGW increments were evaluated for the SSS mission, including those resulting from (1) DLI TOGW changes, (2) VCE SFC characteristics, and (3) VCE geometry dependent drag. Consequently, engines which produced DLI TOGW reductions resulted in a corresponding increment in SSS TOGW, as shown in Figure 10. Only one VCE, the General Electric 0.95 BPR Modulating Bypass TF, resulted in a TOGW reduction because of its SFC characteristics at the SSS mission subsonic cruise and loiter flight conditions. Again, the Detroit Diesel Allison VGT TJ was the only VCE which did not achieve a TOGW reduction due to the geometry dependent drag. The total SSS TOGW change is equal to the sum of the three increments shown in Figure 10. Two VCE concepts significantly reduced SSS TOGW.

The results of the DLI and SSS mission TOGW sensitivity evaluations identify the Detroit Diesel Allison VGT TF and the 0.95 BPR General Electric Modulating Bypass as the most promising of the VCE candidates evaluated. Although the 0.5 BPR General Electric Modulating Bypass TF produced greater DLI TOGW reduction

FIGURE 9
TOGW SENSITIVITY EVALUATION - DLI MISSION
AXI-SYMMETRIC V/STOL NOZZLE CONFIGURATIONS
BASELINE DLI TOGW ≈ 30,000 LB CLASS

	DETROIT DIESEL ALLISON				GENERAL ELECTRIC	
	ΔTOGW - LB				ΔTOGW - LB	
VCE CONCEPT / TOGW CONTRIBUTOR	VGT TF	VGT DRY TJ	PARALLEL TURBINE TF	REVERSE PITCH VGT TF	MODULATING BYPASS TF (BPR = 0.50)	MODULATING BYPASS TF (BPR = 0.95)
MANEUVERING THRUST - NPF	−650	−150	−720	−800	−1,180	−1,200
$(SFC)_{NPF}$	−1,680	−3,440	−4,710	−510	−1,820	−1,350
L/C + DL* ENGINE WEIGHT	−530	710	5,210	13,110	860	1,410
GEOMETRY DEPENDENT DRAG	−910	800	−1,040	−1,720	−510	−920
ΔTOGW TOTAL	−3,770	−2,080	−1,260	10,080	−2,650	−2,060

$* \dfrac{\partial TOGW}{\partial ENG\ WT} = 3.26$

FIGURE 10
TOGW SENSITIVITY EVALUATION - SSS MISSION
AXI-SYMMETRIC V/STOL NOZZLE CONFIGUATIONS
BASELINE SSS TOGW ≈ 40,000 LB CLASS

	DETROIT DIESEL ALLISON				GENERAL ELECTRIC	
	ΔTOGW - LB				ΔTOGW - LB	
VCE CONCEPT / TOGW CONTRIBUTOR	VGT TF	VGT DRY TJ	PARALLEL TURBINE TF	REVERSE PITCH VGT TF	MODULATING BYPASS TF (BPR = 0.50)	MODULATING BYPASS TF (BPR = 0.95)
CHANGE IN * DLI MISSION TOGW	−6,040	−3,330	−2,010	16,130	−4,250	−3,300
$(SFC)_{NPF}$	2,260	14,280	4,710	5,370	4,930	−530
GEOMETRY DEPENDENT DRAG	−1,030	900	−1,290	−1,610	−560	−890
ΔTOGW TOTAL	−4,810	11,850	1,410	19,890	120	−4,720

$*\Delta TOGW_{SSS} = 1.6\ \Delta TOGW_{DLI}$

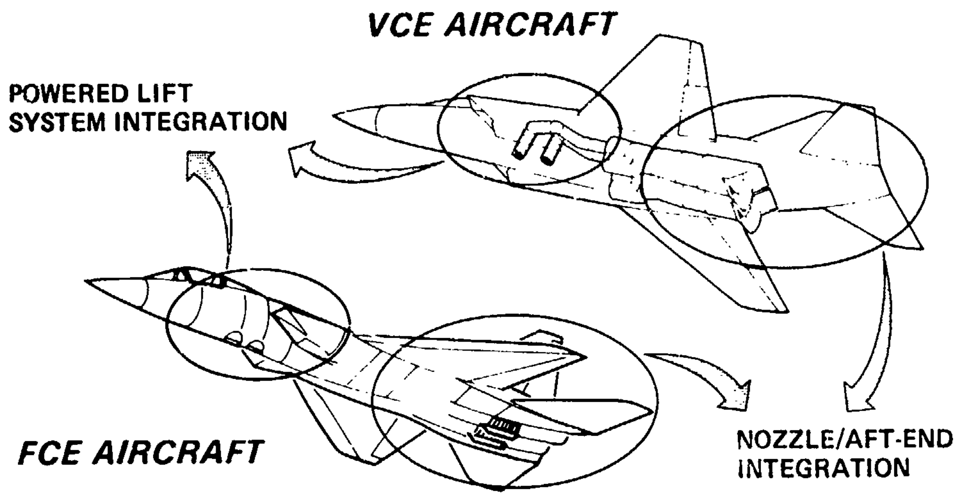
than did the 0.95 BPR engine, the SSS evaluation clearly identified the 0.95 BPR as the superior design for maximum payoff potential with multiple design mission requirements.

3.3 Evaluation Results for Two-Dimensional Nozzle Installations

General Electric provided two VCE concepts with two-dimensional Augmented Deflecting Exhaust Nozzles (ADEN) for evaluation in this program. Both designs were designed to permit full augmentation with vectored thrust for the DLI VTO and the SSS STO. The RLS concept uses fan discharge airflow, which is ducted forward of the aircraft CG to provide a powered lift system, and thus, eliminate the requirement for separate direct lift engines.

The installation of such propulsion systems results in a significant departure from the design integration of the FCE aircraft as shown in Figure 11. Therefore, the potential payoff for these VCE concepts was assessed by developing design layouts to determine VCE weight and geometry impact on TOGW and, then, using the FCE aircraft sensitivities to determine the TOGW impact of VCE thrust and SFC. To obtain consistent and comparable results, this procedure was also used to reassess the TOGW payoff of the two most promising VCE concepts (Detroit Diesel Allison VGT TF and 0.95 BPR General Electric Modulating Bypass TF) with axisymmetric nozzle installations.

FIGURE 11
VCE/AIRCRAFT DESIGN INTEGRATION



Payoff potential was estimated for both the DLI and SSS design missions and the results are shown in Figure 12 and 13 respectively. All four VCE concepts resulted in TOGW reductions, relative to the FCE aircraft, due to thrust, SFC and geometry dependent drag, but only the Detroit Diesel Allison VGT TF achieved a TOGW reduction due to engine weight. The total DLI TOGW change is equal to the sum of the four increments shown; and all four engines achieved a net DLI TOGW reduction. The linear relations between DLI TOGW changes and SSS TOGW changes discussed above were again used to assess the impact of VCE weight and geometry on SSS TOGW as shown in Figure 13. Only the 0.95 BPR General Electric Modulating Bypass VCE achieved a SSS TOGW reduction due to SFC. Once again, all four VCE concepts indicated potential TOGW reductions in the SSS design mission. The results of these evaluations are summarized in Figure 14. Two of the engines, the Detroit Diesel Allison VGT TF and the General Electric Modulating Bypass TF plus RLS, indicated significant TOGW payoff potential in both the DLI and SSS mission applications.

FIGURE 12
INTEGRATED DESIGN TOGW SENSITIVITY EVALUATION RESULTS
DLI MISSION

VCE CONCEPT	V/STOL NOZZLE DESIGN	ΔTOGW - LB				ΔTOGW TOTAL (DLI)
		NPR	SFC	AIRCRAFT DRAG	WEIGHT	
DDA VGT TURBOFAN	AXI-SYMMETRIC	−650	−1680	−210	−810	−3350
GE MODULATING BYPASS TF BPR = 0.95	AXI-SYMMETRIC	−1200	−1360	−600	+1760	−1400
GE MODULATING BYPASS TF BPR = 0.5	2-D ADEN	−1650	−1370	−510	+920	−2610
GE MODULATING BYPASS TF + RLS	2-D ADEN	−1160	−1670	−1170	0	−4000

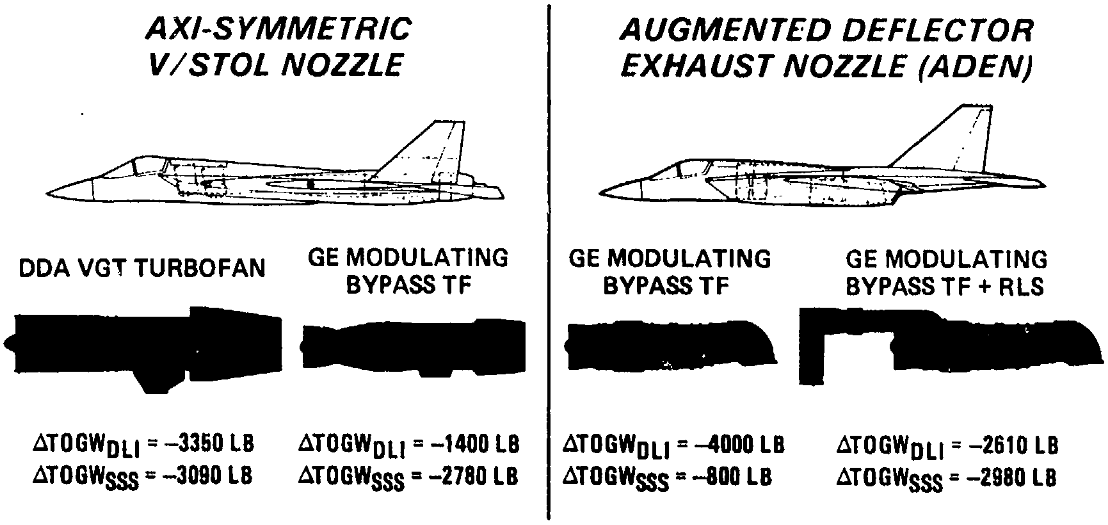
FIGURE 13
INTEGRATED DESIGN TOGW SENSITIVITY EVALUATION RESULTS

SSS MISSION

VCE CONCEPTS	V/STOL NOZZLE DESIGN	ΔTOGW - LB		ΔTOGW TOTAL (SSS)
		SFC	DRAG AND WEIGHT(1)	
DDA VGT TURBOFAN	AXI-SYMMETRIC	+2270	–5360	–3090
GE MODULATING BYPASS TF BPR = 0.95		–540	–2240	–2780
GE MODULATING BYPASS TF BPR = 0.50	2-D	+5600	–6400	–800
GE MODULATING BYPASS TF + RLS	ADEN	+790	–3770	–2980

Notes: (1) $(\Delta TOGW)_{SSS} = (\Delta TOGW)_{DLI}\ (1.6)$

FIGURE 14
INTEGRATED DESIGN TOGW SENSITIVITY EVALUATION SUMMARY



3.4 Evaluations of VCE Operational Flexibility

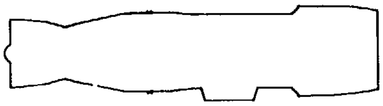
The fixed cycle engine used to define the aircraft TOGW sensitivities was a conceptual design incorporating technology projected for 1990 engines. However, due to individual engine company design techniques and schedule optimizations, this engine could introduce inconsistencies into the sensitivity evaluations. Thus, it was desired to assess the potential impact of such inconsistencies.

A General Electric modulating Bypass turbofan was evaluated and the results were compared with the VCE evaluation results. The TOGW sensitivities were used to determine the DLI and SSS TOGW for the General Electric FCE design.

demonstrates the advantage of variable cycle features. Both engine concepts provided a substantial TOGW reduction for the DLI mission, which emphasizes good supersonic flight engine performance at dash and combat. However, the VCE concept provided a substantially higher TOGW payoff for the SSS mission where the emphasis is on good subsonic cruise and loiter engine performance. On the basis of these results, it appears that the major payoff for VCE may be achieved in aircraft with multi-mission requirements.

FIGURE 15
DEMONSTRATION OF VCE OPERATIONAL FLEXIBILITY

FCE
OPR = 22
BPR = 0.95



VCE
OPR = 25
BPR = 0.95

	ΔTOGW$_{DLI}$				
	THRUST	SFC	WEIGHT	DRAG	TOTAL
FCE	−1200	−1000	950	−660	−1910
VCE	−1200	−1360	1420	−920	−2060

	ΔTOGW$_{SSS}$			
	DLI TOGW	SFC	DRAG	TOTAL
FCE	−3070	2900	−720	−890
VCE	−3300	−530	−890	−4720

VCE PAYOFF

MULTI-MISSION FLEXIBILITY

GP76-0509-5

4. CONCLUSIONS

The simplified analysis procedures used in Phase I of this program allowed a rapid and enlightening evaluation of several VCE concepts. The results, TOGW reductions of approximately 10%, showed sufficient VCE payoff potential to warrant proceeding to Phases II and III. The most promising VCE concepts were the Detroit Diesel Allison Variable Geometry Turbofan and the General Electric Modulating Bypass Turbofan engines. It was also apparent that an aircraft which is required to perform only one mission will probably not derive a significant TOGW reduction from the use of a VCE. However, an aircraft that must perform several missions of diverse nature will benefit from the use of a VCE.

5. REFERENCES

(1) J. Frederick, USAF Aero Propulsion Laboratory, R. E. Martens, McDonnell Douglas Corporation, R. Sutton, Boeing Aerospace Company, "Turbine Engine Cycle Selection Procedures", presented at the 3rd International Symposium on Airbreathing Propulsion, Munich, Germany, 7-12 March 1976.

DISCUSSION

J.Hourmouziadis

One of the engine concepts favored is the Detroit Diesel Allison VCE incorporating a variable HP Turbine. When does the author expect such a turbine to be available?

Author's Reply

We are presently engaged in advanced programmes with a number of different variable geometry turbines and Allison has in fact run one variable turbine, high temperatured and highly cooled, this being a preliminary version after some improvements. There will be one available for advanced type tests in the next year or so. Also it is to be mentioned that there is a similar programme on variable low pressure turbines going on.

J.F.Chevalier

Je voudrais faire deux remarques dans la comparaison du résultat de données par tous le moteurs. Il me semble que les bons résultats obtenus avec le premier de tous les moteurs, un VGT/TF d'Allison, résulte d'un optimiste certain sur le poids de ce moteur. Il devrait, certainement, dans les delta, être pénalisé plus par son poids. Quant aux derniers moteurs qui donnent des résultats assez intéressants aussi, c'est un retour finalement au Pegasus, à peu de chose très. C'est une remarque que je fais.

Author's Reply

I should point out that in the fixed and variable cycle engine comparison for the multi-mission flexibility comparison very consistent weight analysis procedures were used. And even though we can agree that we may not know in detail the weight penalties due to the variable components, I think that the use of consistent weight analysis procedures considering those components and then still indicating a pay-off is a very attractive result.

CYCLE VARIABLE ET TRANSPORT SUPERSONIQUE

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INTRODUCTION.

Ce n'est pas par hasard si le concept du moteur à cycle variable est né à l'occasion des études d'un avion de transport supersonique de deuxième génération.

Le domaine de vol d'un avion supersonique étant très étendu, le propulseur ne peut pas être optimisé dans tous les cas de vol et les qualités qui lui sont demandées peuvent même être contradictoires. En particulier, le principal obstacle au développement du transport supersonique est le problème du bruit au décollage. Il est impératif que les avions de transport supersonique futurs satisfassent aux règlements actuels ou en cours d'élaboration, concernant le bruit au décollage. Cette obligation, impliquant une vitesse d'éjection modérée, coïncide avec les exigences d'un bon rendement aux basses vitesses, mais est incompatible avec l'obtention d'un faible maître couple et d'un bon rendement aux vitesses supersoniques.

C'est pourquoi l'idée du moteur à cycle variable, qui réduirait comme par magie ces incompatibilités, apparaît comme particulièrement séduisante.

Mais le moteur à cycle variable, qu'est-ce que c'est ? Nous allons tenter de dresser le portrait robot du moteur à cycle variable idéal adapté au transport supersonique à Mach 2, en cherchant quelles devraient être ses caractéristiques principales du triple point de vue des cycles, de l'encombrement et de la constitution interne.

1. POSSIBILITES ET LIMITES DES MOTEURS CLASSIQUES.

Pour cela, il est tout d'abord nécessaire d'examiner quelles sont les possibilités et les limites des moteurs classiques.

En croisière supersonique, les principales qualités demandées au propulseur sont d'une part une faible consommation spécifique, d'autre part une forte poussée spécifique (poussée par unité de débit) et un faible maître couple, afin de réduire la masse de l'ensemble propulsif et la traînée de la nacelle.

1.1. Choix du cycle.

La planche 1 présente les résultats d'une étude des cycles monoflux et double flux mélangés sans réchauffe, dans les conditions de vol $M_0 = 2$ $Z \geqslant 11$ km   ISA + 5.

Les poussées spécifiques et consommations spécifiques sont présentées par rapport à un propulseur de référence de même cycle que l'OLYMPUS 593, propulseur actuel de CONCORDE.

Les cycles double flux mélangés dépendent de trois paramètres indépendants, par exemple : taux de dilution, température entrée turbine et taux de compression primaire, le taux de compression de la soufflante étant fixé par la condition d'égalité des pressions à la confluence.

Dans l'étude présentée, le taux de dilution est variable de 0 à 2,5 par pas de 0,5, tandis que la température entrée turbine est augmentée par pas de 50° jusqu'à 200° au-dessus de la valeur actuelle de l'OLYMPUS. Pour chaque valeur du taux de dilution et de la température entrée turbine, il existe une valeur du taux de compression primaire qui rend la consommation spécifique minimale. Ce taux de compression optimal est très élevé, irréalisable pratiquement, et conduit à des poussées spécifiques très faibles.

Dans la pratique, on recherchera donc une solution de compromis permettant, moyennant un léger sacrifice sur la consommation spécifique, d'augmenter la poussée spécifique et de diminuer le taux de compression primaire. C'est ce qui a été fait dans cette étude où, pour chaque valeur du taux de dilution et de la température entrée turbine, nous avons déterminé la valeur du taux de compression primaire permettant d'obtenir la poussée spécifique la plus élevée au prix d'un sacrifice de 2,5 % sur la consommation spécifique par rapport au moteur optimal correspondant.

L'examen des résultats montre que le taux de dilution peut varier dans des proportions importantes de 0 à 2,5, sans changement significatif de la consommation spécifique.

Par contre, la poussée spécifique décroît rapidement quand le taux de dilution augmente. De plus, dans un moteur double flux classique, on n'est pas maître de l'évolution du taux de dilution entre le décollage et la croisière. Un moteur double flux, avec un taux de dilution modeste de 0,5 au décollage, se retrouve en croisière supersonique avec un taux de dilution voisin de 1. Il en résulte, à la température entrée turbine de référence, une diminution de poussée spécifique supérieure à 40 % par rapport au moteur monoflux de référence.

Si on veut limiter cette perte de poussée spécifique à 30 %, il faut augmenter la température entrée turbine de 200°.

En conclusion, le cycle le mieux adapté à la croisière supersonique est le cycle monoflux sans réchauffe puisqu'il procure une excellente consommation spécifique et une poussée spécifique élevée.

L'utilisation de la réchauffe sur les flux primaire et secondaire, ou de la chauffe du flux secondaire, permettrait d'augmenter la poussée spécifique des moteurs double flux en croisière supersonique, mais au prix d'une augmentation prohibitive de la consommation spécifique.

1.2. Contraintes d'installation.

La poussée fournie à la cellule par le groupe propulsif est la résultante de la poussée du moteur et de la traînée de la nacelle, principalement la traînée d'onde qui représente une partie non négligeable (environ 6,5 %) de la traînée totale de l'avion.

L'optimisation doit donc porter sur l'ensemble du groupe propulsif et non sur le moteur seul.

La traînée de carène est proportionnelle au maître couple de la nacelle et dépend de la différence de section entre l'entrée d'air et le maître couple de la nacelle.

Dans un moteur classique, le compresseur basse pression est unique, son débit par unité de surface frontale est maximal au décollage. En première approximation, on peut dire que le maître couple de la nacelle est proportionnel à la section d'entrée du moteur et par conséquent au débit d'air dans les conditions décollage.

Par contre, la section d'entrée d'air est dimensionnée pour la croisière supersonique, elle est proportionnelle au débit d'air du moteur en croisière.

Finalement, la traînée de carène est proportionnelle au débit du moteur au décollage et à la loi de variation du débit entre le décollage et la croisière.

La planche 2 montre, pour un avion donné, de même conception aérodynamique que CONCORDE, quelle serait l'influence sur la traînée totale de l'avion d'un changement de la conception des moteurs et de leur loi de débit à travers la variation de la traînée de carène, en prenant comme propulseur de référence le moteur actuel.

L'ordonnée représente la variation de la poussée spécifique en croisière, l'abscisse le rapport débit croisière / débit décollage, les courbes en trait plein sont les courbes d'égale traînée totale de l'avion, les courbes en pointillé les courbes d'égale vitesse d'éjection décollage, donc de même niveau de bruit.

Pour réduire le bruit au décollage par rapport à la situation actuelle de CONCORDE et atteindre FAR 36, il faudrait diminuer la vitesse d'éjection décollage de 30 % minimum ; il en résulte, à même loi de débit, une diminution de la poussée spécifique croisière de 30 % et une augmentation de la traînée totale de l'avion d'environ 3 %. Ceci traduit simplement le grossissement relatif des nacelles par rapport à l'avion.

Un moteur tel que le point A représente donc une solution classique double flux sans réchauffe. Il permet de réduire sensiblement le bruit au décollage, mais au prix d'une augmentation de la masse et de la traînée des nacelles, car les possibilités d'action sur la loi de débit sont faibles.

2. MOTEURS A CYCLE VARIABLE : DEFINITION DES OBJECTIFS.

2.1. Objectifs relatifs au cycle.

L'optimisation du système pour la croisière impose un cycle ou ensemble de cycles monoflux. C'est cette configuration, considérée comme fondamentale, que nous appellerons "moteur de base".

Au décollage, le moteur de base fournirait une poussée insuffisante puisque sur l'OLYMPUS un taux de réchauffe de 20 % est nécessaire. De plus, la vitesse d'éjection du monoflux sans réchauffe doit encore être réduite d'au moins 20 % pour atteindre le niveau de bruit fixé par la norme FAR 36.

Le cumul de ces deux exigences nécessite une augmentation du débit au décollage d'au moins 50 % par rapport au moteur de base.

La puissance équivalente du jet est inférieure de 4 % seulement à celle du moteur de base. Le moteur utilisé en configuration décollage ayant vraisemblablement un rendement thermique moins bon, il est nécessaire qu'il ait un débit de carburant au moins égal à celui du moteur de base.

2.2. Objectifs relatifs à l'encombrement.

Le moteur à cycle variable doit avoir une faible vitesse d'éjection, donc fonctionner en double flux au décollage pour diminuer le bruit. On voit immédiatement sur la planche 2 qu'un tel moteur ne peut pas avoir un compresseur BP unique, car s'il en était ainsi, on obtiendrait un moteur tel que le point B. En effet, ce moteur devrait avoir le diamètre d'entrée, donc le maître couple du moteur double flux A, tandis que la section de captation de l'entrée d'air serait celle du moteur de référence.

Il en résulterait une augmentation inacceptable de la traînée de croisière (+ 13 %) par suite de l'augmentation de la traînée de carène due, d'une part au grossissement relatif de la nacelle, d'autre part à la déformation des formes extérieures.

La nécessité de disposer tout le système de telle sorte que le maître couple de la nacelle en croisière soit déterminé par la section d'entrée du moteur de base, implique que les compresseurs BP autres que celui du moteur de base soient disposés au niveau d'une partie étroite de ce moteur s'ils sont logés dans la nacelle, ou escamotables dans le cas contraire.

Les compresseurs BP supplémentaires logés dans la nacelle doivent être alimentés par des entrées d'air latérales disposées en paroi ou en "écopes" rétractables.

3. ENSEMBLE DE PLUSIEURS MOTEURS.

La première idée qui vient à l'esprit est d'utiliser deux moteurs :

- un moteur double flux à grand débit et à faible vitesse d'éjection pour le décollage,
- un moteur simple flux pour la croisière supersonique.

Un tel système n'est valable, du point de vue aérodynamique, que si le double flux peut être logé dans la nacelle du simple flux ou escamoté en croisière.

Il est évident que ce moteur ne peut pas être logé dans la nacelle puisque son diamètre est supérieur à celui du monoflux, et que ses dimensions le rendent difficilement escamotable. D'autre part, la masse d'un tel ensemble serait inadmissible.

On est conduit à envisager un ensemble dans lequel les dimensions du double flux seraient réduites grâce à l'utilisation simultanée au décollage des deux moteurs, le monoflux étant assez réduit pour que son niveau de bruit soit admissible. Les deux moteurs peuvent alors être disposés dans une même nacelle, comme le montre la planche 3.

Le monoflux, monocorps ou double corps, présente un resserrement entre la sortie des compresseurs et l'entrée de la chambre. Le double flux est monté autour du monoflux au niveau de cette partie étroite, il est alimenté par des prises d'air latérales.

Un tel ensemble ne pose pas de problèmes particuliers de fonctionnement puisqu'il est constitué de deux moteurs classiques indépendants.

La masse doit être élevée et supérieure à celle de deux moteurs respectivement de même cycle disposés côte à côte.

Le diamètre élevé des paliers du double flux, en plus des problèmes technologiques qu'il peut poser, impose une vitesse de rotation modérée et limite le taux de compression primaire, ce qui conduit à une température entrée turbine et un taux de dilution insuffisants pour obtenir un bon rendement.

Cependant, cet ensemble, constitué d'un monoflux à régime réduit et d'un double flux de performances médiocres, aura une consommation spécifique au décollage plus faible qu'un monoflux avec réchauffe.

Les systèmes composés de plusieurs moteurs indépendants du point de vue thermodynamique ne méritent pas véritablement le nom de moteurs à cycle variable, puisque chaque moteur réalise toutes ses fonctions (compression, combustion, détente, éjection) avec des organes qui lui sont propres.

Nous devons donc rechercher des systèmes plus intégrés dans lesquels le plus grand nombre de fonctions serait réalisé par les mêmes organes dans toutes les configurations.

Pour cela, nous allons passer en revue les procédés utilisables pour atteindre les objectifs fixés.

4. PROCEDES UTILISABLES.

Les deux objectifs principaux sont :

- une augmentation du débit au décollage de 50 % au moins par rapport au moteur de base monoflux,
- une diminution de la vitesse d'éjection au décollage de 20 % au moins par rapport au moteur de base.

Pour la clarté de l'exposé, nous étudierons séparément les procédés qui permettent d'atteindre chacun de ces objectifs. Cette séparation est artificielle, car un même dispositif entraîne simultanément une variation du débit et de la vitesse d'éjection.

4.1. Procédés d'augmentation du débit (planche n° 4).

4.1.1. Gavage.

Ce procédé, qui consiste à placer devant le compresseur BP du moteur de base un compresseur supplémentaire, peut être écarté immédiatement. En effet, le compresseur de gavage doit être dimensionné pour le débit total en configuration décollage, l'objectif d'encombrement ne peut donc pas être atteint.

4.1.2. Injection.

En configuration décollage, un débit d'air supplémentaire est injecté par un ou plusieurs compresseurs auxiliaires entre les deux parties numérotées 1 et 2 du compresseur BP. Le taux de compression du compresseur 1 est relevé et sa garde au pompage diminuée par cette opération, réciproque d'une décharge.

Un tel dispositif peut être réalisé de façon à ne pas dépasser le maître couple à l'entrée du compresseur 1 en plaçant les compresseurs d'injection au niveau d'une partie étroite du moteur de base.

Ce procédé employé seul ne fournit pas une augmentation de débit suffisante, car la quantité d'air injecté est limitée par la marge au pompage du compresseur 1.

4.1.3. Système série-parallèle.

Ce procédé est basé sur l'emploi de compresseurs assurant des fonctions différentes suivant la configuration.

L'augmentation de débit au décollage est obtenue en disposant en parallèle deux compresseurs 1 et 2 qui, dans la configuration de croisière, sont disposés en série. Ainsi, le compresseur 1 assure la compression secondaire au décollage, et le compresseur 2 assure la fonction de compresseur BP. La commutation d'une configuration à l'autre nécessite une vanne avec croisement de flux à débits élevés sous faible pression, dont la réalisation est délicate et l'encombrement important, tant en longueur qu'en diamètre.

4.1.4. Systèmes annexes débrayables.

Dans ce procédé, l'accroissement de débit est fourni par un système séparé, hors service en configuration à faible débit, qui n'augmente pas le débit à l'entrée du monoflux, et ne lui emprunte aucun organe, mais en reçoit la puissance qui lui est nécessaire.

Ce débit supplémentaire peut être aspiré par une soufflante. Les puissances mises en jeu étant trop élevées pour qu'un embrayage puisse être envisagé, la transmission mécanique n'est possible que si la soufflante est accouplée à un arbre monoflux qui est arrêté en configuration à faible débit ou encore, si la puissance prélevée par la soufflante peut être annulée en croisière, par un système de géométrie variable sur la soufflante.

Les transmissions électrique et pneumatique peuvent être utilisées.

La soufflante peut être intégrée à la nacelle (soufflante unique montée autour du moteur, ou petites soufflantes disposées en barillet) ou escamotable.

Enfin, l'accroissement de débit peut ainsi être obtenu par des trompes alimentées par de l'air ou des gaz fournis par le monoflux. Un tel dispositif présente peu d'intérêt par suite de son poids, de son encombrement, de son niveau de bruit interne élevé et de ses performances médiocres.

4.2. Procédés utilisables pour diminuer la vitesse d'éjection.

On peut agir au niveau du cycle du moteur de base ou au niveau du jet.

4.2.1. Modification du cycle du moteur de base.

On peut diminuer l'énergie disponible après les turbines en diminuant la pression dans la chambre de combustion. Ce procédé peut rarement être employé seul, il entraîne une dégradation du rendement du cycle. On peut aussi augmenter le taux de détente des turbines :

- en diminuant la température entrée turbine, ce qui entraîne une diminution de la poussée,
- en diminuant le rapport débit des turbines / débit des compresseurs, soit par prélèvement d'air, ce qui diminue également la poussée, soit en augmentant le débit des compresseurs,
- en prélevant de la puissance sur les arbres par un système débrayable. Ce procédé peut s'employer seul, sans modifier la partie amont du cycle.

Les premiers procédés, baisse de pression chambre, baisse de température, diminution du débit détendu, interviennent simultanément sans qu'il soit possible de les séparer, sauf si on introduit des géométries variables sur les distributeurs de turbine.

4.2.2. Prélèvement de puissance sur le jet.

On peut prélever cette puissance soit sous forme de chaleur, soit sous forme d'énergie mécanique.

Le prélèvement de chaleur ne peut se faire que par un échangeur. Un tel dispositif est lourd et encombrant, induit des pertes de charge importantes et ne permet qu'un prélèvement d'énergie modéré sous une forme difficilement utilisable.

Le prélèvement d'énergie mécanique peut se faire :

- par une turbine. Cette turbine doit pouvoir être mise hors service. La solution consistant à la retirer de la veine est pratiquement irréalisable. La solution consistant à la rendre transparente est difficilement réalisable, car elle implique une géométrie variable non seulement sur les distributeurs, mais aussi sur les aubes mobiles. De plus, les pertes de charge seraient importantes.
- par un système magnétohydrodynamique. Ce système serait lourd et d'un rendement médiocre.
- par une trompe.

L'énergie est ainsi communiquée directement à un autre flux, mais avec un mauvais rendement. De plus, la masse et l'encombrement d'un tel système sont inacceptables.

Nous allons maintenant illustrer l'utilisation des procédés qui viennent d'être décrits par des exemples de moteurs à cycle variable envisagés jusqu'à ce jour par nos concurrents, les avionneurs et par la S.N.E.C.M.A.

5.1. Applications du procédé série-parallèle.

5.1.1. Le système proposé par ROCKWELL (planche 5) constitue une solution intermédiaire entre une combinaison de moteurs monoflux et double flux totalement indépendants et des systèmes plus intégrés.

Il est composé d'un moteur double flux double corps classique et de plusieurs moteurs monoflux monocorps identiques disposés autour du double flux.

En croisière supersonique, les monoflux sont alimentés par la soufflante, le système se comporte alors comme un ensemble de deux cycles monoflux couplés entre eux par un échange de puissance.

Au décollage, les moteurs périphériques sont alimentés par des entrées d'air latérales et la soufflante du double flux débite dans une tuyère séparée. Le système se comporte alors comme l'ensemble d'un double flux et d'un simple flux indépendants.

En croisière subsonique, le moteur double flux est utilisé seul, ce qui procure une excellente consommation spécifique.

L'augmentation de débit obtenue au décollage est d'autant plus modérée que la commutation série-parallèle n'est effectuée que sur le flux secondaire du double flux. La diminution de la puissance thermique dans l'ensemble des chambres est elle aussi assez faible, puisque le dégavage n'affecte que les moteurs périphériques.

Le cycle obtenu en croisière correspond à une poussée spécifique élevée, donc à une traînée de nacelle acceptable si l'ensemble ne dépasse pas le maître couple à l'entrée de la soufflante. Même en optimisant le cycle en croisière, ce qui n'est pas facile puisqu'il dépend de six paramètres indépendants, la consommation spécifique en croisière est supérieure de 2 à 3 % à celle de l'OLYMPUS.

5.1.2. Sur le même principe, la S.N.E.C.M.A. a étudié la configuration représentée sur la planche 6, dans laquelle un moteur simple flux unique entoure le moteur double flux central.

Les paliers posent les mêmes problèmes que pour les systèmes de la planche 4.

5.1.3. La planche 7 montre l'application du procédé série-parallèle proposé par BOEING sur un moteur monoflux.

En croisière, le moteur fonctionne en monoflux double corps.

Aux basses vitesses, une partie du compresseur BP est utilisée comme soufflante, l'éjection se faisant par des tuyères disposées autour de la nacelle. La deuxième partie du compresseur BP est alimentée par des entrées d'air latérales.

En plus du problème d'encombrement posé par croisement des flux, le dégavage du flux primaire diminue la poussée réalisable au décollage.

5.1.4. Pour pallier cet inconvénient, PRATT & WHITNEY applique ce même principe au flux secondaire d'un moteur double flux (planche 8). L'arbre BP porte deux soufflantes séparées.

Dans la configuration croisière double flux, le flux secondaire est comprimé successivement par chacune des deux soufflantes.

Dans la configuration décollage, triple flux, le flux comprimé par la première soufflante est éjecté par des tuyères latérales, la seconde soufflante est alimentée par des entrées d'air supplémentaires.

L'augmentation de débit ne porte que sur le flux secondaire. Le cycle de croisière supersonique est un cycle double flux ; il faut donc utiliser la chauffe du flux froid pour rétablir une poussée spécifique élevée, ceci au détriment de la consommation spécifique.

5.2. Moteurs à débit augmenté par un système annexe débrayable.

5.2.1. Soufflantes mues par un prélèvement sur le jet.

La planche 9 montre une étude S.N.E.C.M.A. d'un moteur dérivé de l'OLYMPUS dans lequel l'accroissement du débit au décollage est obtenu par de petites turbosoufflantes disposées autour du canal de réchauffe, alimentées par un prélèvement de gaz derrière les turbines.

La contrainte encombrement conduit à limiter le prélèvement d'énergie à une partie du débit fourni par le générateur.

Avec un prélèvement de l'ordre de 30 %, on peut entraîner des soufflantes de taux de compression de l'ordre de 1,8 à 2 et réaliser un taux de dilution de 0,5 à 0,6, ce qui permet de ne pas utiliser la réchauffe au décollage. Toutefois, il reste un jet primaire rapide, donc bruyant.

5.2.2. Transmission pneumatique.

Les soufflantes sont entraînées par des turbines alimentées en air comprimé par le moteur de base. Les turbosoufflantes peuvent être indifféremment montées dans la nacelle (soufflante unique montée autour d'une partie étroite du générateur ou petites turbosoufflantes disposées en barillet) ou hors de la nacelle et rétractables dans une partie de l'avion.

Le principal problème réside dans le comportement du générateur hors adaptation et son aptitude à fonctionner avec un prélèvement d'air important. De plus, le prélèvement d'air tend à diminuer la poussée réalisable au décollage.

La planche 10 montre un des systèmes étudiés par la S.N.E.C.M.A. avec prélèvement d'air à la sortie du compresseur HP.

Sur ce générateur monoflux double corps, pour éviter la désadaptation des compresseurs, le prélèvement d'air est permanent.

En configuration croisière supersonique, ce prélèvement alimente une chambre de combustion secondaire et une turbine accouplée à l'arbre BP, puis est mélangé avec le flux principal.

Au décollage, cette chambre et cette turbine sont mises hors service et le prélèvement d'air alimente les turbosoufflantes.

Les performances en croisière sont détériorées par les pertes de charge dans le cycle secondaire.

6. CONCLUSIONS.

Ces quelques exemples constituent des tentatives plus ou moins heureuses en vue de se rapprocher du moteur à cycle variable idéal, tel que nous l'avons défini.

Nous avons montré qu'il y avait des exigences d'encombrement et de performances au décollage et en croisière, qui rendent le problème très difficile à résoudre. Toutefois, il convient de signaler que les critères qui déterminent le choix d'un moteur dépendent de la mission de l'appareil, en particulier de la vitesse de croisière.

Les problèmes d'aérodynamique externe sont plus difficiles à Mach 2 qu'à Mach 2,7, car la section de captation de l'entrée d'air est plus faible. De même, l'utilisation de la réchauffe ou de la chauffe du flux froid est moins pénalisante en consommation spécifique quand la vitesse de croisière augmente.

Dans tous les cas, la contrainte encombrement impose des configurations très complexes avec plusieurs compresseurs BP intégrés dans la même nacelle, fonctionnant simultanément ou non, suivant les conditions de vol.

En plus des difficultés évidentes d'installation, il en résulte également des difficultés de fonctionnement au niveau de la conception même, car il faut imaginer un système procurant une grande souplesse dans l'utilisation de la puissance produite par le moteur de base.

Il est souhaitable de rechercher des composants dont les caractéristiques soient satisfaisantes dans un large domaine de fonctionnement, notamment par l'utilisation de géométries variables sur les compresseurs ou les turbines, ce qui accroît encore la complexité.

Nous avons également défini les principes d'augmentation de débit ou de prélèvement de puissance utilisables dont nous n'avons pas épuisé toutes les applications. Il faudra encore déployer beaucoup d'imagination et d'invention avant d'aboutir à des solutions réalisables pratiquement.

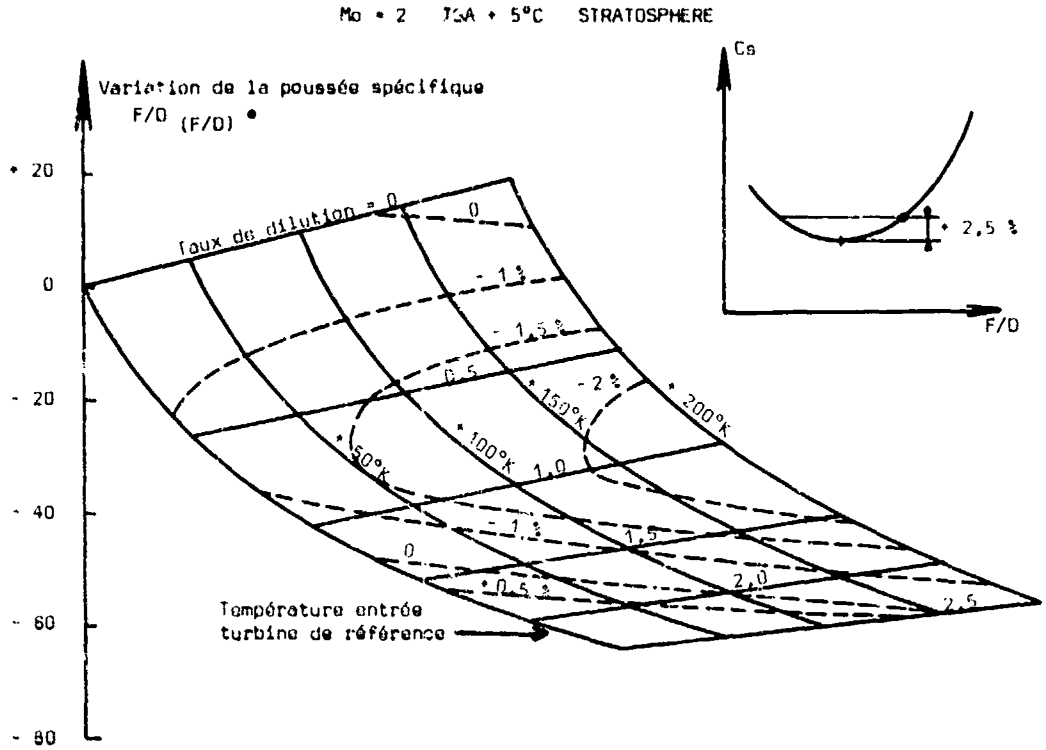


Planche 1 - Performances des moteurs double flux sans réchauffe optimaux

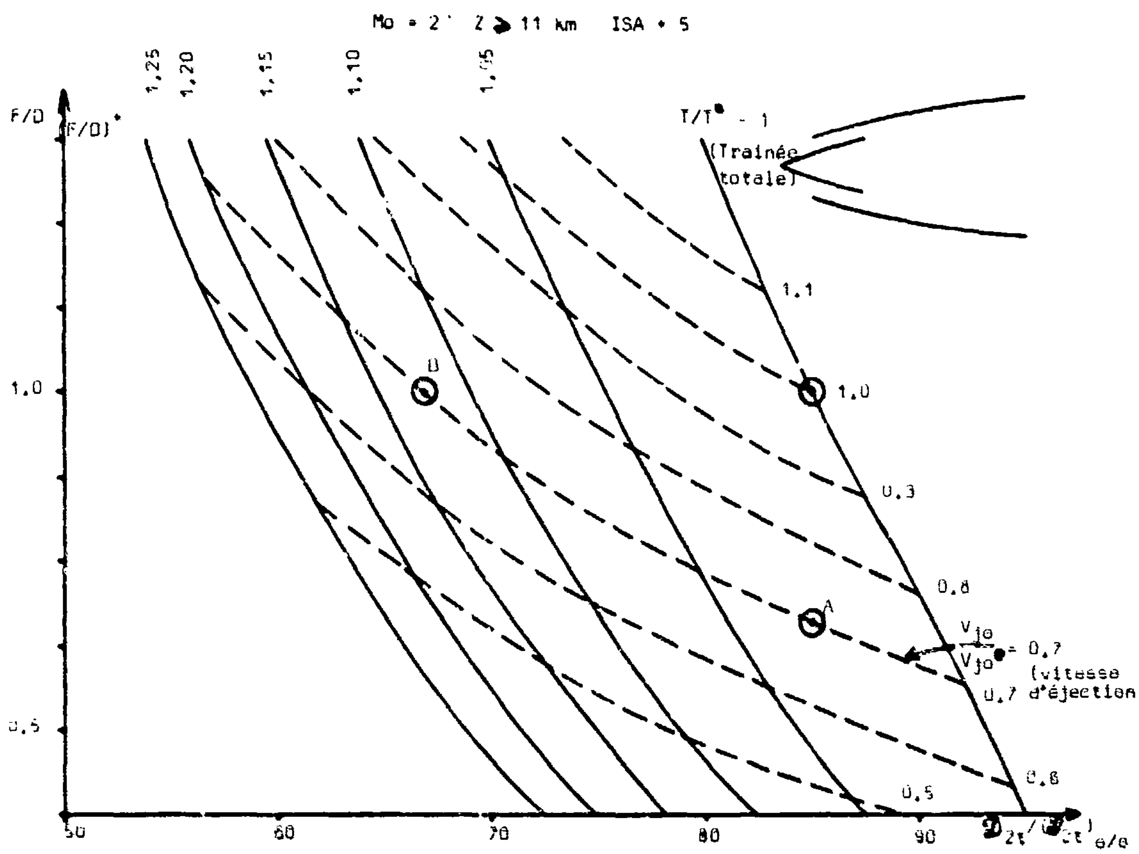


Planche 2 - Influence du propulseur sur la traînée de l'avion

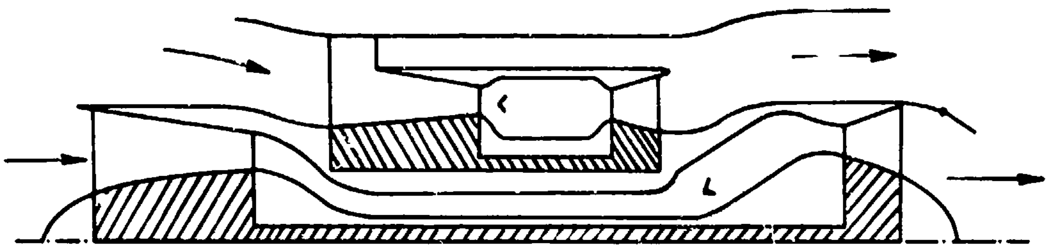
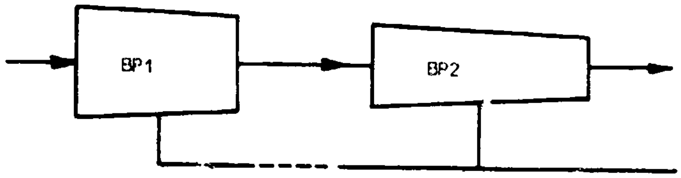
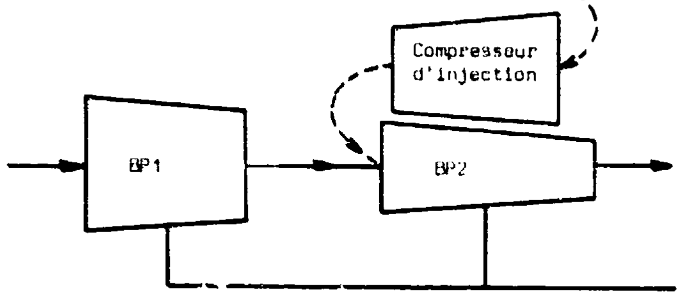


Planche 3 - Moteurs indépendants concentriques

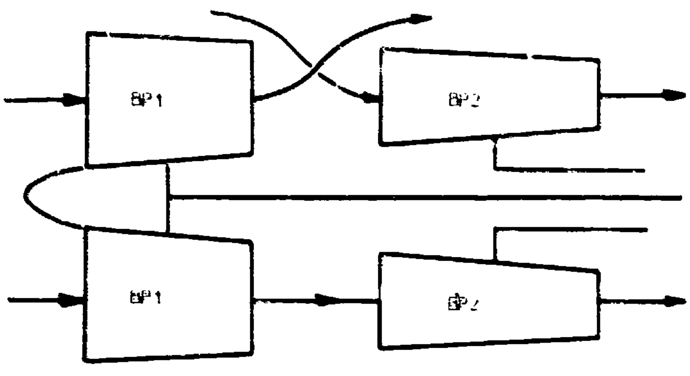
1°) Gavage



2°) Injection



3°) Série-parallèle
(Principe BOEING)



4°) Systèmes annexes débrayables
- soufflantes
- trompes

Planche 4 - Procédés d'augmentation de débit

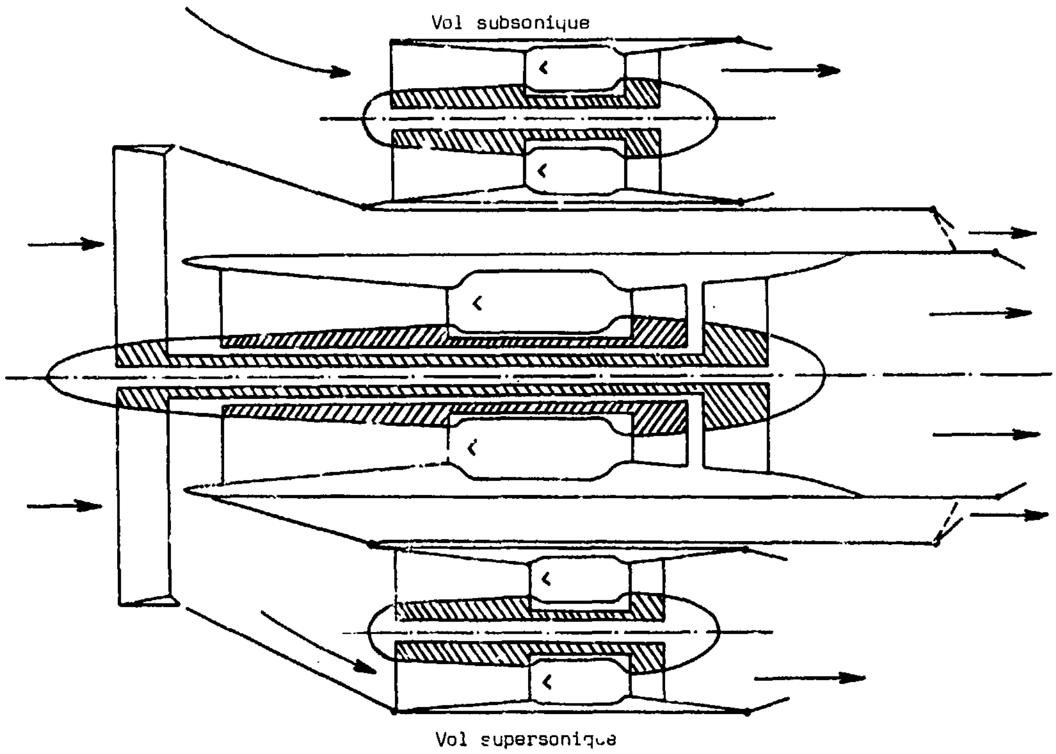


Planche 5 - Système ROCKWELL

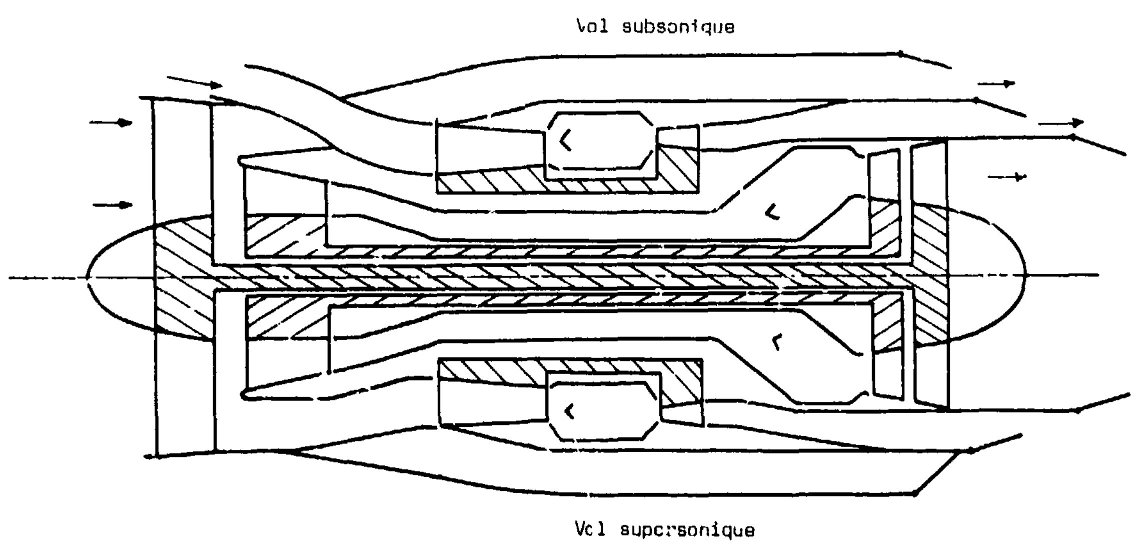
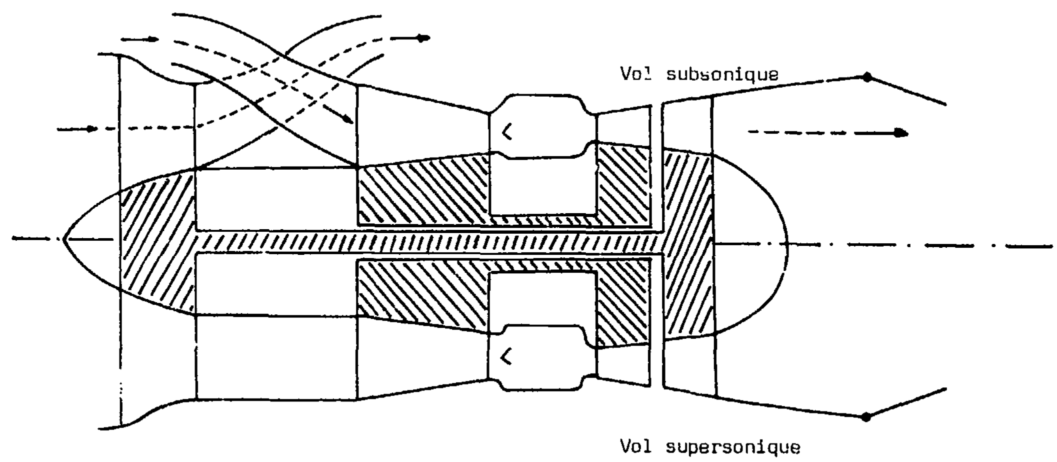


Planche 6 - Etude S.N.E.C.M.A. de moteurs concentriques



Pla 7 - Système série-parallèle application monoflux

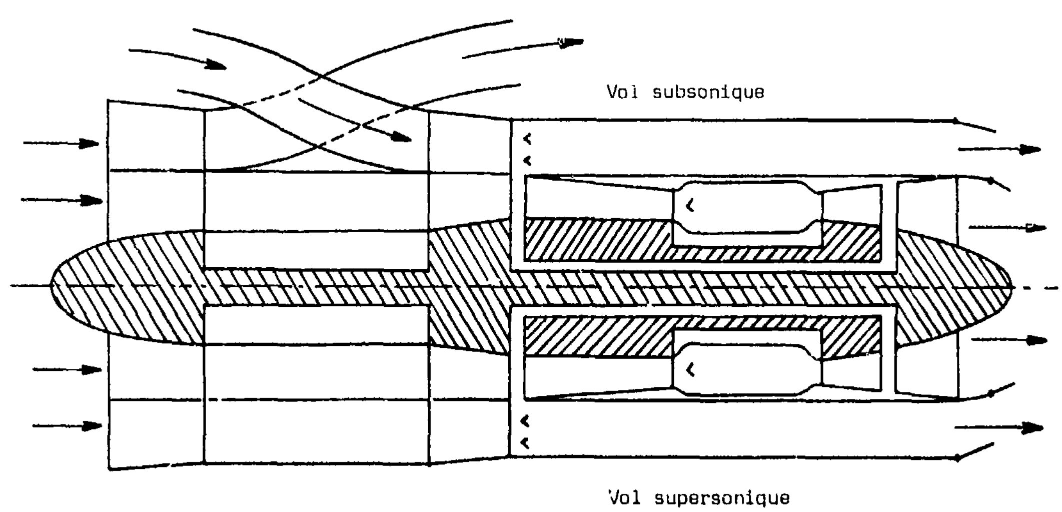


Planche 8 - Système série-parallèle application double flux (Etude P.W.A.)

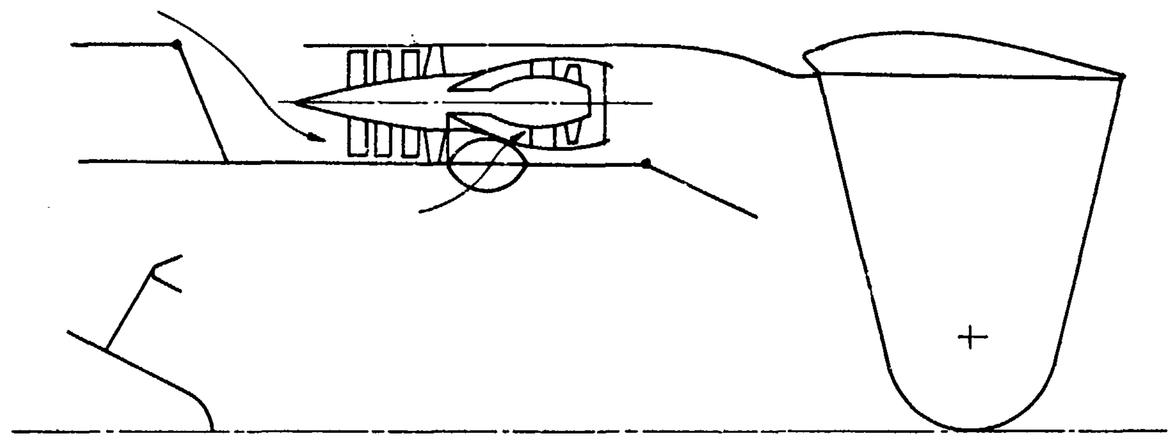


Planche 9 - Prélèvement de gaz (Etude S.N.E.C.M.A.)

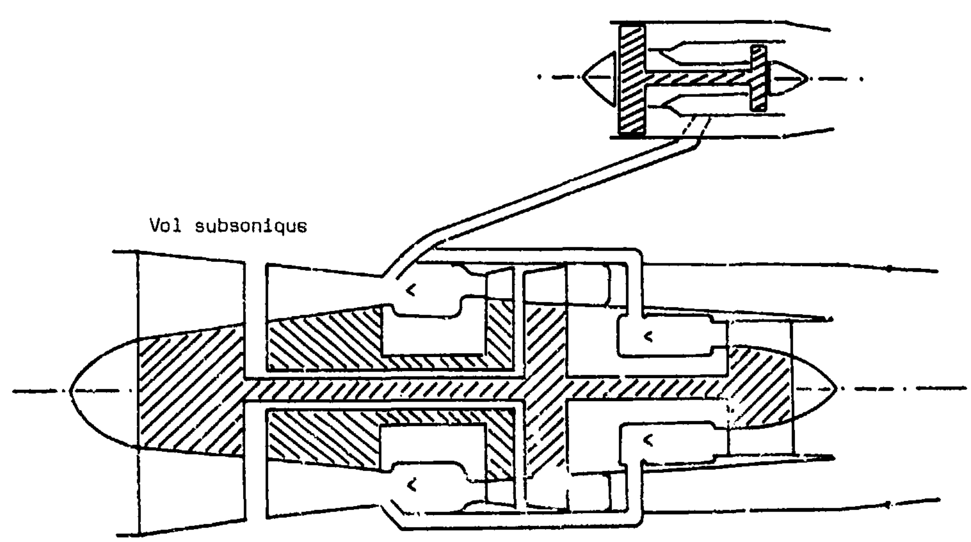


Planche 10 - Prélèvement d'air (Etude S.N.E.C.M.A.)

E.A.Willis

Have you estimated the weight of these engines and their performance over the full flight spectrum?

Author's Reply

Non. Pour l'instant nous n'avons fait que des études portant sur les performances. Il s'agit uniquement d'études théoriques et d'études thermodynamiques dans les différents cas de vol.

E.A.Willis

Do the thermodynamics studies include off-design as well as point design considerations?

Author's Reply

Nous cherchons à obtenir des performances à niveau de poussée donné en croisière, au décollage, et en transsonique, et dans chaque cas de vol nous cherchons à optimiser également les consommations spécifiques, enfin surtout pour les croisières.

J.F.Chevalier

Je prends la parole, Il est dommage que Mr Willis n'ait pas pu lire la totalité de son texte dans lequel il y avait un chapitre que expliquait que le moteur, qui doit avoir obligatoirement deux compresseurs BP d'après Mr Menioux, a été abandonné pour diverses considérations. Il serait apparu qu'une différence essentielle entre les deux études est la différence du nombre de Mach de croisière. Je crois que si Monsieur Menioux pouvait, pour la Table Ronde par exemple, refaire approximativement sa planche No 2 qui était la base de sa démonstration, pour le cas de Mach 3, il pourrait peut-être faire apparaître les différences liées aux deux applications. Croyez-vous, Monsieur Menioux que c'est possible?

Author's Reply

Non.

J.F.Chevalier

Dommage!

H.Grieb

In figures 7 and 8 you described engine arrangements with two compressors working both "in series" and "in parallel" as well. During the switch from the mode "in series" to the mode "in parallel", the pressure level in the rear compressors drops, i.e. the fluid within this compressor temporarily tries to flow in forward and rearward directions. This may cause surge of the rear compressor. Can you give a comment on this matter?

Author's Reply

Effectivement, il s'agit d'un problème très délicat. La configuration représentée sur la page 8 nécessite effectivement un système de croisement des flux avec une vanne permettant de passer d'une configuration à l'autre; mais ce problème a déjà été résolu par Boeing.

H.Grieb

Yes, the reason for my question is when you come from operation in series to operation in parallel, you have changed in the first case HP compressor to a low pressure level and you open an area in front of the HP compressor to a region where the pressure is lower. This could be a problem.

Author's Reply

Non, nous avons fait des études purement théoriques, je ne suis pas en mesure de répondre.

VARIABLE-CYCLE ENGINES FOR SUPERSONIC CRUISE AIRCRAFT

by
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SUMMARY

Since 1973, the NASA Lewis Research Center has been conducting studies of advanced civil supersonic engines, including Variable Cycle Engines or VCE's, as one part of the Supersonic Cruise Aircraft Research (SCAR) program. This paper reviews the progress and current status of the engine study work to date.

VCE rationale is first reviewed. It is pointed out that the VCE is a possible means of reconciling the necessary but sometimes contradictory performance, economic and environmental requirements that apply to modern supersonic-cruise aircraft. Early experiences showed, however, that VCE's may be excessively complex, heavy and expensive unless significant technology advances are accomplished. The SCAR engine studies were, therefore, designed to identify the most promising VCE concepts, simplify their designs to a more practical state, and define their advanced technology requirements.

The studies were conducted primarily via contracts, supplemented by a lesser amount of NASA in-house work. Initial efforts involved analyzing, optimistically but in little depth, a large variety of VCE concepts. In subsequent phases, a progressively-greater depth of analysis was applied to a decreasing number of surviving candidates. The line of development leading from initial to final concepts is reviewed with emphasis on the dual impact of technology advancements and design simplification. The presently-favored VCE's (two P&W concepts derived from a duct-burning turbofan and two GE engines based on a mixed-flow turbofan) are then reviewed. It is shown that all have benefitted significantly from recent SCAR technology advances, such as the "co-annular noise benefit" effect. The impact of each technology area is discussed. It is also shown that these simplified VCE cycles and technology advances, taken together, offer major performance, economic and environmental improvements relative to the 1970 U.S. SST predictions.

It is concluded that final choices among the current VCE candidates will depend on application and installation factors as well as further engine study/design and technology efforts. NASA's tentative plans in these latter respects are reviewed in the final section of the paper.

INTRODUCTION

Since early 1973, the NASA and its Contractors have been conducting studies of advanced supersonic Variable Cycle Engines (VCE's) as part of the Supersonic Cruise Aircraft Research (SCAR) program. This paper surveys the progress and current status of recent, unclassified engine study work.

Technical, economic and environmental problems were sources of major concern which eventually led to the cancellation of the U.S. SST program in 1970. Major environmental concerns were primarily focussed upon the engine's noise and exhaust emissions, as illustrated in Fig. 1. Other technical and economic problems were attributable partly to the propulsion system and partly to the airplane. These resulted in excessive weight and cost of the airplane, together with high fuel consumption and inadequate range. Consequently, this airplane would have been unable to serve many of the economically desirable city pair combinations. These factors would have caused the airplane to be costly to operate and to offer a relatively poor return on its investment. Inflation together with recent increases in the price of fuel would have made the situation even worse today.

The one unmistakable lesson to be learned from this experience is that any future U.S. civil supersonic airplane must be environmentally acceptable and economically viable. The sometimes-conflicting requirements of economic viability and environmental acceptability create major problems for the propulsion system. Their practical engineering solutions entail essentially contradictory design trends, e.g., high bypass vs. low bypass. Unfortunately we cannot turn to contemporaty engines for relief. The U.S. J58 and J93, although capable of cruising at Mach 3 or above, are relatively old designs and are not suitable for an advanced supersonic transport. Modern U.S. military engines such as the F100, J101, and F101 were essentially designed for sustained subsonic cruise efficiency with only a high Mach number dash capability. Their performance and service life characteristics for sustained supersonic cruise would be unacceptable for the applications envisioned now.

There are many ways to build a VCE and, as a matter of historical interest, some of the early ideas are described in refs. 1 & 2. For this discussion, however, a VCE is best defined by what it does rather than how it is built. Functionally, it is an engine which accommodates at least two distinct modes of operation: (1) a high airflow, low jet-velocity mode for low noise takeoff and/or efficient subsonic cruise; and (2) a turbojet-like, higher jet velocity, lower airflow mode for good supersonic cruise.

In more technical terms, the motivation for this "turbofan-convertible- to-turbojet" definition may be understood by reference to Fig. 2. There, weight and cruise-SFC trends for conventional supersonic engines are presented in terms of bypass ratio. Clearly, both weight and subsonic fuel economy favor a fairly high bypass ratio, about 1.5 (turbofan mode). Supersonic cruise on the other hand calls for a low bypass engine, 0.3 or below when fuel economy is considered, but this is tempered somewhat by the adverse weight trend. With a conventional engine, a compromise bypass ratio (usually in the 0.5 to 1.5 range, depending on the subsonic/supersonic mission mix) must be chosen, which is not really optimum for either requirement. The rationale for a VCE, then, is its potential ability to give use a better compromise. For this reason, the SCAR propulsion program was oriented to include VCE concepts and related technologies in addition to advanced conventional engines. It consists of studies and related technology subprograms which, collectively, were designed to identify, develop, and integrate together the technologies needed for a successful VCE. The study phase of the program is of primary concern in this paper.

The SCAR Propulsion studies were conducted primarily via contracts to GE and P&W, with a major subcontract to Boeing. Early phases of the studies involved analyzing, optimistically but in little depth; a large variety of VCE concepts. The results showed that VCE's may be prohibitively complex, heavy and expensive unless significant design and technology advances are accomplished. The final phases were, therefore, intended to identify, refine and compare the most promising VCE concepts, simplify their designs toward practicality and define their advanced technology requirements. The presently-favored and runner-up engines (a P&W advanced duct-burning turbofan, a P&W valved derivative of the duct-burner and two GE engines based on a mixed-flow turbofan) are first reviewed. Their performance in typical advanced supersonic transport airframes is then compared to that provided by first-generation SST engines. The impact of each major technology area is discussed and the technology needs of the preferred engines are reviewed.

The final fate of the VCE idea will depend on application and installation factors, further engine design and technology efforts, and the possible emergence of even more attractive VCE cycles from continuing studies. Future issues, options, and potential program plans in these areas are briefly reviewed in the final section of the paper.

THE SUPERSONIC CRUISE AIRCRAFT RESEARCH PROGRAM

The NASA Supersonic Cruise Aircraft Research (SCAR) program was instituted in early 1973 and is expected to continue into the 1980's. In contrast to the earlier SST project, the SCAR work is not aimed toward a production airplane, but rather, it is intended to establish a data base of advanced technology to be available for the design of future supersonic cruise aircraft if and when the U.S. determines it is desirable to build them. The various elements of the program are relevent in varying degrees to both potential civil and military applications. Elements of the program apply both to the airplane structure and aerodynamics and to the propulsion system; however, only the propulsion related aspects will be discussed here. As shown on Fig. 3, the SCAR propulsion program consists of two major, interrelated elements; namely, engine studies and technology sub-programs. These are so structured that one supports the other. The engine studies define the objectives and directions of research for the technology sub-programs. The results from the technology sub-programs in turn feed back into the engine studies and regenerate them. As indicated above, the engine studies have been conducted primarily by means of a continuing series of contracts to the Pratt & Whitney Co. (refs. 3 and 4) and the General Electric Co. (refs. 5 and 6), with a major sub-contract between P&W and The Boeing Co. (described in refs. 4, 7-9). Technology sub-programs involving these contractors as well as others have been launched in the areas of noise abatement (refs. 10-13), pollution reduction (refs. 14-16), inlet stability (ref. 17), and supporting component and material programs (e.g., ref. 18). References 19 and 20 survey the SCAR propulsion and airplane technology programs sponsored by the NASA Lewis and Langley Research Centers.

Before elaborating on these programs, we would like to illustrate the type of advancements are are considered possible now, based on results to date from the SCAR program. In Fig. 4, we have plotted airplane relative gross weight vs. relative noise footprint area (a typical measure of noise annoyance) for representative supersonic transport airplanes with different kinds of engines. These are approximate results taken from ref. 21 but are illustrative of the major trends. For reference, we have indicated on the horizontal axis the noise annoyance factors typical of the 1970 U.S. SST (at the right hand part of the scale) and also of a representative wide body subsonic transport. The performance of the 1970 technology turbojet powered airplane is illustrated by the right hand band on the figure. As mentioned previously, this was a heavy airplane and would have created a severe noise impact. Although the noise impact could be decreased by scaling the engine up in size and throttling it back for takeoff, this entails a substantial weight penalty as indicated. This in turn makes an already dubious economic payoff entirely unacceptable. But by taking advantage of the technology breakthrough termed the "co-annular noise reduction benefit" identified during the SCAR propulsion program, combined with variable cycle engine concepts to be discussed later, it now appears that the noise annoyance due to this type of an airplane can be reduced by a large factor compared to the 1970 U.S. SST. A less dramatic but still significant improvement in gross weight and airplane economics is also indicated and is due to a combination of many technology advances, in both the propulsion and airframe areas, that are considered possible.

Because of these promising developments we now feel, for the first time, that the noise objections that were leveled against the 1970 SST program can be met without incurring prohibitive economic penalties. An equivalent statement cannot yet be made in the exhaust emissions area, despite the achievement of significant improvements, because realistic standards applicable to an SST do not exist at present.

Engine Studies

Let us now turn to the SCAR engine studies themselves. Beginning in 1973, the studies have been divided into 4 distinct phases as indicated in Fig. 5. Phase 1 was organized in such a way as to exclude no reasonable candidate engine from consideration. Many engines were studied optimistically but in very little depth, see refs. 3 and 5. Only those engines which were obviously unacceptable under this optimistic approach were excluded from further consideration. Our deliberate intent was to give the Variable Cycle Engine its day in court. After the unpromising concepts had been screened out, a smaller number of survivors received a more refined analysis in Phase 2 (refs. 4 and 6). Four finalists survived into Phase 3 which has just recently been completed and is as-yet unpublished. In this phase a greater depth of analysis was accomplished and we initiated preliminary design activities. Based on the results, we have now tentatively identified two engines which appear to be most promising. (Their margins of superiority, however, are not overwhelmingly large; the runners-up are being retained as backups and will also be described.) In Phase 4 we are initiating airframe integration activities, continuing with preliminary design and developing a series of technology recommendations relative to the favored engines. These provide the engine manufacturers with an opportunity to define, for NASA's consideration, what is needed in terms of future technology programs in order to bring these paper engines into being. As illustrated by the arrow in the upper right we expect that these activities will eventually result in demonstrator engines which will prove the concepts that are being contemplated.

Before proceeding to a discussion of the currently-favored engines, it seems appropriate to briefly review the evolution of the VCE idea and describe how it may be impacted by two major technology area.

Early VCE Concepts

According to our previous definition, a VCE is an engine that <u>does</u> the right things. The many attempts that have been made to actually design one may be broadly classified into two generic approaches. One would rely upon valves or equivalent means to create two or more discrete flowpaths upon demand within the same engine structure - each flow path presumably being tailored to the flight condition at hand. The alternative approach would rely primarily upon component variability and spool speed variations to achieve equivalent results.

A typical early example (Pratt & Whitney, ref. 3) of the changing-flowpath approach is shown in Fig. 6. Here a valve is inserted between the fan and compressor of an otherwise-conventional 2-shaft machine. In the "turbojet" mode, the valve is set in its straight-through position. The fan and compressor flow in series, and we have in effect a two-spool, high overall-pressure-ratio (OPR) turbojet. As such, it can provide very good supersonic performance. In the "turbofan" mode, the valve mechanism is moved to the "crossover" position suggested by the lower sketch. Fan air supplied by the normal inlet is bypassed around the compressor and into an auxiliary bypass duct. Meanwhile, additional air from an auxiliary inlet is drawn through a second set of channels in the valve, into the compressor, and hence, through the combustor and turbines. Thus, the engine is now operating at a much higher (up to 2X) airflow than before and without augmentation its jet velocity is significantly decreased. In this mode, the engine provides a low-noise takeoff mode and potentially good subsonic SFC.

By the standards of our functional definition, this engine does the right things. Numerous objections, however, were found upon closer examination. From an engine manufacturer's viewpoint, it developed that the weight and pressure-loss penalties associated with the valve were significantly larger than had been expected. Since the core is desupercharged in the turbofan (parallel) mode, the OPR is considerably below the optimum value for subsonic cruise. For the same reason a variable (and probably multi-stage) low-pressure turbine would be needed to provide high relative work extraction in the turbofan mode, and lower extraction in the turbojet mode. From the airframe point of view it was observed that the requirement for an efficient auxiliary inlet implied a major design and development task and a significant additional installed-weight penalty (above that required to enclose the engine's greater length and diameter). The closed-off bypass duct also would entail a sizable base or boattail drag penalty during supersonic cruise.

Subsequent efforts were aimed at removing or minimizing these complications. As described in ref. 4, many alternatives involving front valves, rear valves, front and rear valves, and improved valve concepts were evaluated iteratively by Pratt & Whitney and Boeing. An historical review of this process is given in ref. 22, where it is shown that the lessons learned also apply, to some degree, to more conventional engines. The rear-valved VCE to be described later herein, is the latest and apparently best example of this particular line of VCE evolution, but probably not its end-point.

The variable-component/variable speed approach is most attractively represented by the Pratt & Whitney Variable Stream Control Engine. Essentially a high-technology duct burning turbofan incorporating some of the component and control features discussed in

Ref. 22, it is currently the favored P&W VCE and will be more fully described later.

Another historically-significant and perhaps more spectacular example is the General-Electric 3-Spool Double Bypass or Modulating Airflow Engine (ref. 5) depicted in Fig. 7. It is a representative sample of the early variable-component approach, although there are many others. It is of particular interest here because it was not only the best VCE identified in the initial GE studies (ref. 5), but also because many of its characteristic features have survived into their currently-favored, much-simplified version of the Double Bypass VCE.

The design approach for this engine was to incorporate the maximum practicable amount of turbomachinery variability into a basic duct-burning turbofan. By utilizing differential speed control among the three rotors, variable stator geometry and properly controlling the three variable nozzle exit areas, it provides (1) a high-airflow, unaugmented mode for low-noise takeoff; (2) a constant-airflow throttling mode for efficient subsonic cruise; and (3) a relatively low-bypass augmented mode for good supersonic performance.

At takeoff, the front fan block or group of stages was high flowed by means of variable geometry, speed control (i.e. speeding-up the inner spool) and opening the outer bypass stream's exit area. The duct burner is not lit. Without using either a mechanical suppressor or the "co-annular benefit" (which was unknown at the time), the Modulating Airflow engine was capable of meeting FAR 36 when sized to be competitive with a conventional reference engine.

Subsonic cruise throttling is accomplished by running the inner rotor at essentially constant speed; the front fan block then maintains its constant nominal airflow over a wide range of conditions. The intermediate and high pressure rotor speeds are varied to modulate the thrust. The excess air provided by the front block (above the intermediate block's air-swallowing capacity) passes through the outer duct to the third nozzle exit. The duct burner is not lit. In this fashion, constant airflow could be maintained down to approximately 50% of maximum dry thrust. This provided a significant (~15%) improvement in subsonic SFC.

At supersonic cruise, the rotor speeds and variable geometry features are modulated to approach turbojet operation as closely as possible. That is, the high pressure and intermediate rotors are run at maximum speed to swallow most of the front block's airflow. The outer nozzle meanwhile is at or near the closed position to minimize the outer bypass flow. The core is run at maximum speed and is high-flowed to swallow as much as possible of the intermediate block's air. This reduces the bypass ratio of the duct-burner portion of the engine and hence the need for augmentation. When run in this manner, the engine's supersonic cruise performance was found to be within 1 or 2% of that of the reference turbojet.

Similar measures applied during the mission's climb/accel segment resulted in a consistently good match to the inlet's flow schedule and hence fuel savings via reduction of installation drags.

Thus, the 3-Rotor Double Bypass or Modulating Airflow engine also does everything required of a VCE: low noise takeoff; fuel savings subsonically and during the climb/accel phase; and competitive supersonic performance. Unfortunately, these desirable features were essentially offset by a major weight penalty (amounting to over 20,000 lbs per airplane, when installed). Depending upon the flight Mach number, the resulting airplane's performance ranged from just competitive to somewhat poorer. Because of the weight penalty together with very legitimate concerns over the engine's complexity, the 3-rotor approach was not continued pas the Phase I SCAR studies. Instead, an effort was made to incorporate its most desirable features into a lighter, less complex and more conventional 2-shaft machine. The concept was retained of dividing the fan into two distinct blocks or groups of stages, with the interblock region ventilated by an auxiliary bypass duct. As will be seen, this progress in design simplification, coupled with the technology advances discussed in the next two sections, has finally resulted in a highly attractive VCE.

The Co-annular Noise Benefit

As previously implied, the "Co-annular Noise Benefit" effect is considered to be the major "break through" in the SCAR propulsion technology program. Figure 8 illustrates what is meant. Attention is first directed to the lower right hand corner of the figure. In brief, it has been found that: (a) if the flow streams of a two stream coaxial nozzle are so arranged that the high velocity stream is one the outside and the low velocity stream is on the inside; and (b) if in addition the outer nozzle has a high annular radius ratio; then the noise produced by this arrangement is significantly lower than would be predicted for two conventional conical nozzles which individually have the same airflows and velocities as in the two coaxial streams. This effect was first noted by Pratt & Whitney during SCAR parametric acoustic testing that commenced in 1974 (refs. 10-13) and was later confirmed by parallel independent testing at General Electric (as-yet unpublished). It is of the utmost significance for SCAR VCE concepts since these inherently involve (or can be so arranged as to provide) a coaxial, high radius ratio two stream nozzle flow configuration at takeoff.

It should be noted that both the coaxial flow configuration and the high annular radius ratio are necessary to obtain the maximum benefit. The term "co-annular" is, therefore, used as a reminder of this fact.

The rest of the chart illustrates the sideline noise produced by either conventional or co-annular nozzles as a function of the jet velocity averaged over the two streams. Two bands are shown, the upper one for conventional nozzles and the lower one for co-annular nozzles. As indicated by the vertical line, the 1970 turbojet operated at a relatively high jet velocity and created a noise signature 12 to 15 dB above the FAR 36 requirement. This could be reduced to some degree by oversizing the engine and operating it throttled back to lower jet velocities for takeoff purposes. As previously mentioned, however, this results in severe airplane weight and economic performance penalties; so severe, in fact, as to be unacceptable. When a co-annular nozzle is used, on the other hand, it is immediately seen that the noise signature is 8 to 10 dB lower than that of the conventional model. If in addition, the engine is a variable cycle engine which is capable of taking-off at reduced jet velocities without otherwise penalizing the airplane, it may be seen that a noise signature below FAR 36 can be anticipated. The combination of the two concepts, namely, the co-annular nozzle and the variable cycle engine, results in perhaps 10 - 12 dB lower noise than that of the conventional nozzle combined with the conventional turbojet engine. This, it is felt, will have a decisive impact on the environmental acceptability of any future SST.

The application of this revolutionary concept to a duct-burning turbofan engine is straightforward. The flow stream configuration is already the proper one, it is only necessary to tailor the cycle to provide the correct velocity and radius ratios. It is also adaptable to some mixed-flow engines via the use of a ventilated plug nozzle of the general type discussed in refs. 23-25. In essence, fan air or inlet ram air is ducted to the plug by some means and exhausted from an annular slot in the afterbody. The abovementioned General Electric acoustical research program has shown that, depending on radius ratio and flow conditions, most of the benefit illustrated in Fig. 4 may be achieved by this arrangement.

Pollution - Reduction Technology

Let us now turn to the second area of environmental concern, namely, exhaust emissions. Of the various emission criteria, that of high altitude cruise NOX is of greatest concern for the supersonic transport. In Fig. 9, we illustrate the comparative performance of several combustor concepts in terms of its relative NOX emission index at supersonic cruise. As indicated by the top bar, a conventional combustor such as was used in the 1970 SST and is still used today in current airplanes, shows the highest emission level and is normalized to 1.0 on this relative scale. (The normalizing factor varies from about 20 gm/kg to 50+ gm/kg depending on cycle conditions.) This may be compared to a value of 3 gm/kg (0.16 to 0.06 relative) which ref. 26 tentatively suggests may be appropriate for the avoidance of appreciable stratospheric pollution by a future SST fleet. The clean combustor concepts developed by Pratt & Whitney and General Electric under our recent SCAR Experimental Clean Combustor Program (refs. 14 & 15) show relative emission indices of approximately 0.4 to 0.5, on the same scale, in burner-rig experiments. This level of performance could be incorporated in a new engine program starting now. Further improvement is predicted for NASA's swirl can combustors and various lean combustor concepts. Probably the most hopeful concepts for the future, however, are in the area of pre-mix combustors and the catalytic combustor concept (e.g., ref. 16). NOX indices as low as 1 gm/kg (0.05 to 0.02 relative) have been demonstrated in small scale, idealized laboratory experiments. But it is clear that a large, lengthy and probably expensive program, including both fundamental research work and applied development, will be required to translate these promising concepts into reality. Assuming that the necessary programs will be forthcoming, we anticipate that relative values as low as 0.25 may eventually be attainable in practical engines. (Absolute levels of course will also depend upon the specific cycles chosen.) It should be recognized, however, that this involves our entering a new and relatively unknown area of technology, and this has yet to be done in a serious way. The above estimates are therefore uncertain, as are the projected requirements; either or both may change significantly in the future.

Although NOX emissions are most critical for an SST, it must be recognized that local (airport-area) emissions must also be environmentally acceptable. It is believed, however, that all of the advanced technology primary burner concepts would be capable of meeting the "proposed" standards for future SST's.

This is not necessarily the case for augmentors, however. The search for a locally-acceptable augmentor will again require us to enter an uncharted technology area.

CURRENT VCE's

Having reviewed early VCE concepts and two major impacting technology area, it is now appropriate to turn to the currently favored VCE's themselves. These "paper" engines are the "final product" of the SCAR engine studies. Further, more refined definitions of these engines must await the outcome of hardware oriented programs.

Pratt & Whitney Concepts

The currently-favored Pratt & Whitney VCE is illustrated in Fig. 10. This Variable Stream Control Engine (VSCE) has the flow path of a conventional duct burning turbofan. But it incorporates an unique main combustor power schedule and makes extensive use of rotor speed control and variable geometry in the fan, compressor, primary nozzle, and secondary nozzle to control its operating bypass ratio. Because of this capability, the VSCE qualifies to be termed a variable cycle engine. Yet it is of striking simplicity in comparison with the approaches illustrated previously in Figs. 6 and 7.

Under subsonic cruise conditions the duct burner is not lit. The engine then is precisely a conventional separate flow medium bypass turbofan engine (bypass $\approx$ 1.5) and it provides relatively good subsonic cruise performance.

For takeoff, acceleration and supersonic cruise, however, additional thrust is required. This is obtained by lighting the duct burner. During takeoff, the additional energy supplied by the duct burner results in higher velocity in the nozzle's outer annular stream. But the additional noise implied by this condition is offset by the coannular noise reduction benefit that was discussed earlier. Thus, the engine, when taking off, should sound more like a conventional turbofan engine than like a high-performance supersonic engine. During supersonic cruise operation the core is speeded up by increasing the temperature in the main combustor and by manipulating variable geometry features. Thereby, the bypass ratio is decreased and the need for augmentation is decreased, resulting in specific fuel consumption approaching that of a well designed turbojet engine.

The second Pratt & Whitney VCE is depicted in Fig. 11. This Rear Valve VCE (VCE-112C) is derived from the duct burning turbofan through the addition of a mixer/crossover valve followed by an additional aft turbine stage - both located downstream of the normal LPT. The VCE-112C has two distinct operating modes depending on the valve position. For takeoff, acceleration and supersonic cruise, the valve is in the "crossover" position. I.e., core air bypasses around the aft turbine and exits through the outer annulus of the nozzle. Thus, the core cycle is that of a turbojet.

The fan air meanwhile passes through the duct burner (which is lit), and is directed by the crossover valve into the aft turbine, where a significant amount of energy is extracted to help drive the LP system. The fan air's cycle is also that of a turbojet; hence, this mode of operation is referred to as the "twin-turbojet mode." Its supersonic performance, however, is not quite as favorable as this name implies, because neither "turbojet" cycle is of the optimum pressure ratio and because of pressure losses and weight/volume penalties due to the valve and aft turbine. Its advantages are relatively low weight (due to the high "bypass" ratio of about 2.5) and an advantageously-shaped supersonic throttle curve. I.e., since the duct burner is upstream of a turbine stage, high augmentations can be accomplished for significantly less SFC penalty than in the VSCE's case. The resulting "flat" throttle curve in turn provides the airplane designer with additional flexibility in terms of engine sizing.

Subsonically, the valve is in the "mix" position and the duct burner is not lit. The combined fan and core streams pass through the aft turbine. The corrected flow is about the same as that provided by the augmented fan stream alone in the supersonic twin-turbojet mode. The aft turbine, however, extracts relatively little power. The engine thus behaves as if it were a conventional mixed flow turbofan for subsonic cruise.

A major disadvantage of the VCE-112C is that the earlier-discussed coannular noise benefit may not apply fully. That is, the nozzle's central stream at takeoff (which originated in the duct burner) is relatively large and of high velocity compared to that of the VSCE. There is hence a core jet noise "floor" which will probably limit the coannular benefit to no more than 50% of that shown in Fig. 5.

A third Pratt & Whitney engine of interest (but not illustrated herein) is a modernized conventional mixed flow turbofan with a relatively low (0.4) bypass ratio known as LBE-430. Although lacking obvious VCE features such as valves or coaxial flow streams, it incorporates the identical general technology assumptions (materials, temperatures, component efficiencies, stresses, cooling techniques, etc.) that were built-into the Pratt & Whitney VCE's. It also utilizes rotor speed control and variable geometry features (to the extent possible) as in the VSCE-502B, to maintain a degree of control over the operating bypass ratio. As will be seen later, it provides excellent performance at low airflow sizes if noise constraints are ignored. Unfortunately, the coannular benefit does not apply to this engine in its present form. Hence, this engine, alone among those considered herein, would require either the use of a mechanical noise suppressor (with its attendant risks and penalties) or a greatly-oversized engine for throttled-back takeoff. It is a useful yardstick, however, for evaluating the merits of the coaxial-stream VCE concepts.

General Electric Concepts

The other preferred VCE concept is the General Electric Double Bypass Engine (DBE) shown in Fig. 12. Like the Pratt & Whitney engine, it is designed to take full advantage of the annular/coannular noise benefit, clean primary burners and augmentors, advanced materials and other SCAR technology developments. But where the Pratt & Whitney engine originated as a duct burning turbofan, the double bypass engine is derived from a conventional mixed flow turbofan by adding features from the 3-rotor engine previously discussed.

The low bypass mixed flow engine can provide excellent supersonic performance, but is prone to be excessively heavy when its airflow is sized for low noise takeoff. As with all conventional turbofans, it also suffers from a significant throttle dependent drag penalty at part power subsonic cruise because airflow decreases along with thrust when the engine is throttled back. To offset these penalties, the double bypass engine provides a temporary high airflow mode for low noise takeoff and the capability to throttle at constant airflow for part power subsonic cruise.

As the figure suggests, this is physically accomplished by dividing the fan into two distinct blocks or groups of stages, and providing an auxiliary duct leading from the interblock region. The resulting flow path is similar to that of the 3-rotor engine, but major progress in design simplification has been achieved - as may be inferred by comparing Fig. 12 with Fig. 7. Although not illustrated here, some of the auxiliary flow can discharge into the plug and exit from the aft surface through an annular slot. This provides the flow configuration and geometry needed to obtain the coannular noise benefit discussed earlier.

Three distinct operating modes may be recognized, depending on the fan block flow settings and whether the auxiliary duct is open or closed.

In the low noise takeoff mode, the auxiliary duct is open, the front fan block is in its high flow setting, the core is operated at maximum takeoff power, and maximum energy is extracted by the low pressure turbine. The tailpipe heater is not lit. In this mode, the double bypass engine provides thrust, airflow and jet velocity characteristics that would be typical of a larger but throttled back conventional engine, or a higher bypass engine. Note, however, that only the front block is high flowed. Hence, there is significant weight savings compared to an equal noise conventional engine. The combination of lower mean jet velocity with the coannular noise benefit results in an engine that is remarkably quiet for its power.

For part power subsonic cruise, the auxiliary duct is again open, and passes the excess airflow provided by the front block. In this fashion, a wide range of throttling may be accomplished at constant airflow, thereby eliminating or minimizing spillage, boattail, and other throttle dependent drags.

In the high power mode for climb, acceleration and supersonic cruise, the auxiliary duct is closed, the core is at or near maximum continuous power, and the tailpipe heater is used as needed. In this mode, the double bypass cycle is identical to that of the conventional low bypass engine, and offers essentially the same performance.

A second General Electric VCE of potential interest is the Dual Cycle Engine or DCE (not illustrated herein). It is also a derivative of the low-bypass mixed flow turbofan, but in this case a relatively simple one. As its name implies it has two modes of operation - mixed flow and separate flow. The conventional mixed flow mode is used for climb, acceleration and supersonic cruise. For takeoff or subsonic cruise, the bypass stream is diverted from the normal mixer and instead exits through a separate nozzle opening. This allows the engine to throttle at constant airflow over a range about midway between the conventional turbofan and the DBE. Since the separated bypass flow could also be led to the plug as in the DBE, the coannular benefit is believed to be applicable. As will be seen, this less-complex VCE is fairly attractive at small airflows but is of less interest in a high-airflow, low noise setting.

ENGINE COMPARISONS

Experience has taught that the engine and airplane cannot be created in a vacuum, that is, developed separately from each other. The intent of engine and airplane studies has been to cause innovation by identifying problems in missions, installations, engine technical constraints, and finally aircraft performance and range. Figure 13 shows the flow-path of the studies conducted under the SCAR program; ref. 27 elaborates upon the method of analysis and presents some preliminary NASA results. We have demonstrated significant progress by this approach. Subsequent charts will show that both the Pratt & Whitney and General Electric engines have improved significantly as the SCAR studies progressed. In each case, the engine concepts have changed significantly, driven at least in part by the airplane requirements. It will be recalled that at the start of the engine studies, there were many engine concepts; but in all cases the requirements have tended towards variable cycle engine concepts as the best overall solution.

Pratt & Whitney Results

The performance of the Pratt & Whitney engines is illustrated in Fig. 14. Here we have plotted total range as a function of engine corrected airflow. For reference, the lower curve labelled "CTJ" shows the performance obtained by a hypothetical current-technology turbojet engine. The airframe, in this case, is representative of modern NASA and contractor thinking derived from the SCAR program. It is an arrow-wing configuration weighing approximately 700,000 pounds at takeoff and would carry 275 to 300 passengers over ranges up to 4,000 or 4,500 nautical miles. The curve labelled "LBE-430S" is for the modern Pratt & Whitney conventional low bypass mixed-flow engine which embodies SCAR technology advances, but no variable cycle engine features. It represents a major advance over the early engine. In unsuppressed form (the dashed curve) it would appear to be a "winner" at low airflows, but is less attractive at high airflows. Unfortunately, this engine in its present form would require a mechanical sound suppressor; its suppressed performance illustrated by the solid curve, is significantly degraded. Illustrated next is the performance of the variable stream control engine, VSCE-502B. Clearly, it provides excellent performance even at low engine airflows. Its major advantage, however, occurs at higher airflow levels that correspond to lower noise performance. Finally, the rear valve VCE-112C is also fairly competitive at low airflows but less attractive in larger sizes. As previously mentioned, this engine because of its inherent cycle and nozzle geometry characteristics does not receive the full coannular noise benefit. It therefore is less attractive than the curve might suggest for civil uses. For other applications, however, or if a solution to this problem is found, it could well merit further consideration.

The overall results are summarized in bar chart form on the other part of the figure. Here we have shown the range obtainable for several different engines as a function of sideline noise (estimated by the simplified methods of ref. 27) and takeoff field length constraints. The results are shown for a long and short field length and for noise levels of FAR 36 and FAR 36 minus 5. For ease of comparison, both the early turbojet and the LBE-430 have been credited with a mechanical suppressor which confers a 8-dB noise reduction (about the same level as obtained via the coannular benefit). In both cases it is clear that the SCAR conventional engine represents a significant advance over the early turbojet and that the variable stream control engine, the prefered P&W VCE, represents a further advance over the modern conventional engine at airflows corresponding to low takeoff noise.

General Electric Results

Similar results for the General Electric engines are illustrated in Fig. 15. Here are plotted the total range as a function of corrected airflow for the 1970 GE-4 SST engine, for the GE Dual Cycle Engine (which, but for its presumed ability to use a coannular nozzle, is essentially a modernized low bypass mixed flow turbofan engine) and for the Double Bypass Engine. Again for ease of comparison, the GE-4 is credited with an 8 dB high-performance suppressor, while the two VCE's presumably receive about the same benefit from the coannular effect. As was the case with Pratt & Whitney engines it is clear that the modernized turbofan or Dual Cycle engine has achieved a significant improvement over the 1970 SST engine, but the Double Bypass engine in turn represents a major further advance - especially in the high airflow regime which corresponds to low takeoff noise. The bar chart in this figure illustrates exactly the same trends. Range again is shown for long and short takeoff field lengths and for FAR 36 sideline noise and FAR 36 minus 5Db. Noise is again computed by the simplified NASA method of ref. 27, individual contractor's estimates may vary somewhat. It is clearly evident that the Double Bypass variable cycle engine represents the major advance, although the Dual Cycle is fairly competitive at the lower airflows that correspond to greater field lengths and higher noise.

TECHNOLOGY REQUIREMENTS AND PROGRAMS

Mentioned earlier was the fact that one objective of the SCAR engine studies was to define the technology requirements for making these paper engines real. Figure 16 is a summary of the major technology recommendations presented by Pratt & Whitney and General Electric. Clearly needed are quiet coannular nozzles, underlined on the figure because they are not only critically needed but are unique comments for these engines and not likely to be developed under other programs. In the same category is the low emmissions, efficient duct burner which is characteristic of the Pratt & Whitney engines alone. Also needed are variable geometry fans, flow control valves, advanced low pressure turbines and advanced inlets. There is a major need for low-emmissions primary burners as well as for advancement in hot section technology in general. As previously mentioned, the favored engines obtain improved supersonic performance by increasing the primary burner temperature and speeding up the core as the engine accelerates toward supersonic cruise operation. A consequence of this inverted temperature profile, is an inverted duty cycle in which the engines must spend perhaps 80% of their life times operating at or near the maximum possible turbine inlet temperature. By comparison, a conventional subsonic engine would take off at maximum temperature and then throttle back several hundred degrees when it reaches cruise conditions. Thus, advanced cooling techniques and advanced high temperature materials are of the greatest importance in these engines. Finally, because of the engines' many adjustable features that must be continuously monitored and controlled in flight for safe and efficient operation, there is also a need for advanced digital electronic controls as indicated.

To address some of these needs, NASA has instituted test bed engine programs with both Pratt & Whitney and General Electric. The current program is austere and is relatively slow paced. The basic Pratt & Whitney test item is a rear end assembly comprising a duct burner and a coannular nozzle. In lieu of a large facility air supply, this assembly will be driven by an F100 engine rematched to approach the Variable Stream Control Engine cycle. The duct burner configuration will be selected on the basis of an analytical screening study followed by segment-rig tests of the most promising configurations, before the boiler-plate burner is assembled. Similarly, the quiet coannular nozzle will be evaluated by means of aerodynamic performance and acoustic model tests before the boiler-plate nozzle is constructed.

NASA is also addressing the General Electric technology needs by a test bed engine program. This is being closely coordinated with military programs involving related concepts. Following parallel logic with the Pratt & Whitney work, an existing military engine (J-101 derivative) will be used as an air supply to test a new aft-end assembly incorporating a quiet coannular nozzle. The military demonstrator includes or can be made to simulate some but not all of the desirable Double Bypass features identified by the SCAR studies. It can be rematched to provide an excellent simulation of the selected cycle at takeoff conditions, and a more limited simulation at other conditions. The design of the quiet nozzle will be established by further analysis and aeroacoustic model tests before the full-sized assembly is constructed. In addition, a new variable geometry front fan will be rig tested separately from the engine/nozzle test. The fan rig test assembly will be sized to be compatible with a future, more advanced testbed engine embodying all significant features of the Double Bypass Engine concept.

In summary, the presently planned testbed programs will accomplish several objectives; namely, to test for each company - General Electric and Pratt & Whitney - the two most critical, most unique technology requirements identified by their SCAR engine studies. At Pratt & Whitney this comprises a clean, efficient duct burner and a quiet coannular nozzle. For General Electric it includes a variable-flow front fan block and a quiet annular nozzle. It is emphasized that these are critical items, unique to the favored engines, and not likely to be developed elsewhere. Hopefully, additional needs appearing in Fig. 16 will be at least partially addressed by other NASA or military programs. If not, a sizeable augmentation of the testbed and related SCAR programs may be necessary in the future.

CONCLUDING REMARKS

At this point, we have reviewed the evolution of two groups of VCE concepts and shown how they have been favorably impacted by design simplification and by technology advancements in many areas - particularly in the area of acoustics. Parallel advancements have been achieved in the airframe area by other parts of the SCAR program.

What is the overall payoff from these developments? In Fig. 17 is shown a plot of subsonic mission leg length versus the airplane's total range capability. Several city-pair combinations of economic interest are spotted on the figure. The line at the left indicates the estimated performance of the 1970 United States SST at one point near the close of that program. The nearly vertical band at the right indicates the performance now predicted for an advanced supersonic transport using variable cycle engine concepts and taking advantage of the SCAR technology advancements that have been discussed. As indicated by the arrows between the lines, these advancements are due to improved engine technology, aerodynamic and structural technology advances and the variable cycle concepts. Clearly, a major improvement in the airplane's ability to serve potentially attractive markets has been identified on paper.

What can be done to make these paper engines real? By the SCAR studies we believe that we are identifying what needs to be done to develop a viable option for some future date. By the testbed programs we are addressing the unique and most critical components for each of the favored VCEs. Admittedly, there are other needs which are not now being addressed. But we believe that if the testbed programs are steadfastly pursued to their successful conclusions, the logical next steps will be forthcoming.

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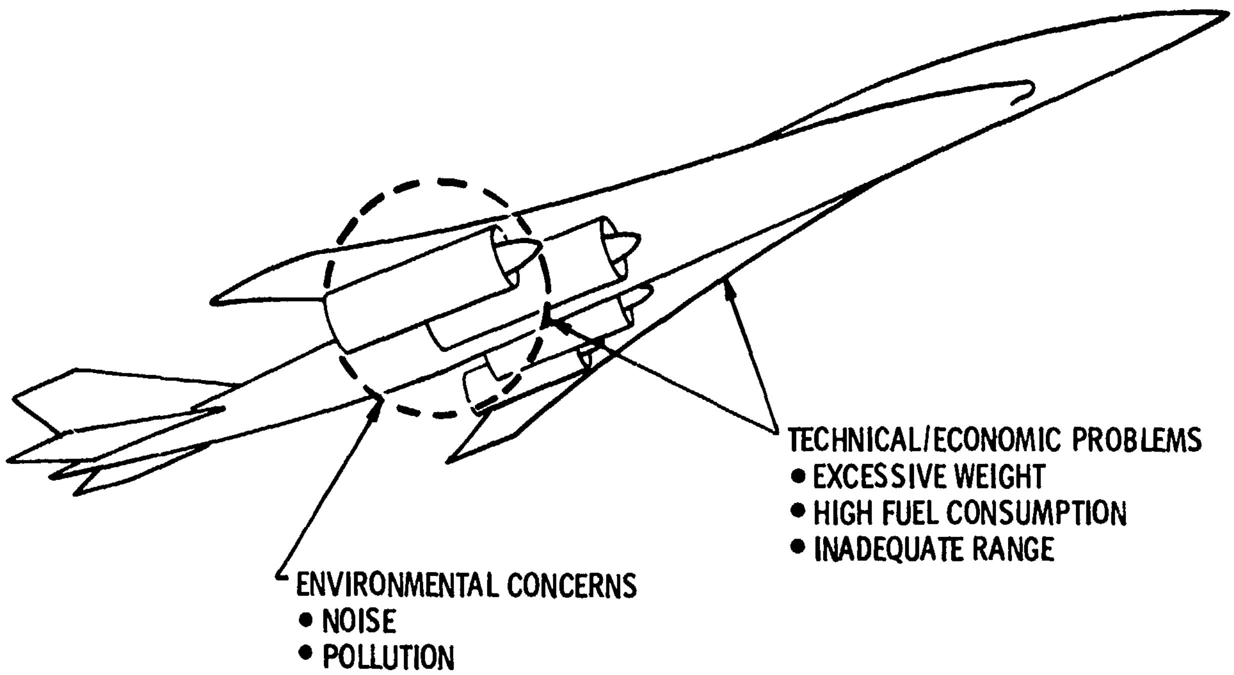


Fig.1 The 1970 U.S. SST program

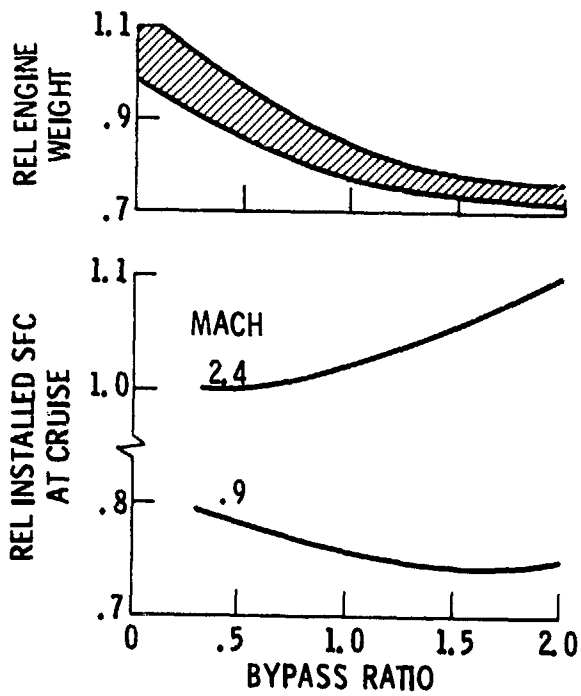
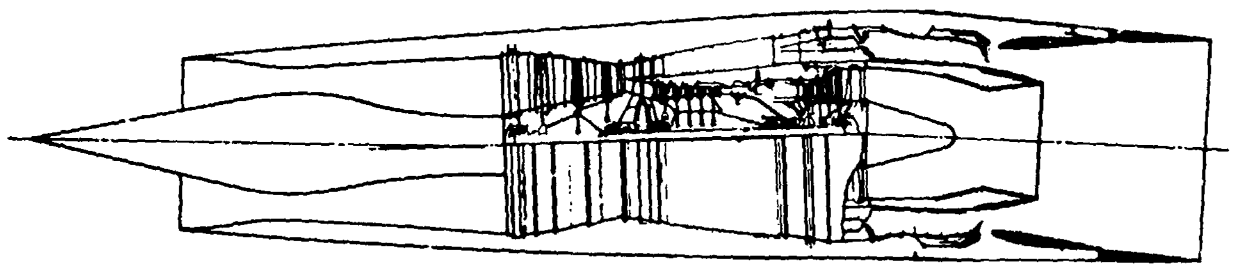


Fig.2 Factors to consider in cycle selection



ENGINE STUDIES
- P&W CONTRACTS
- G.E. CONTRACTS
- P&W/BOEING SUBCONTRACT

TECHNOLOGY SUBPROGRAMS
- NOISE REDUCTION
- POLLUTION REDUCTION
- INLET STABILITY
- MATERIALS

Fig.3 The SCAR propulsion program

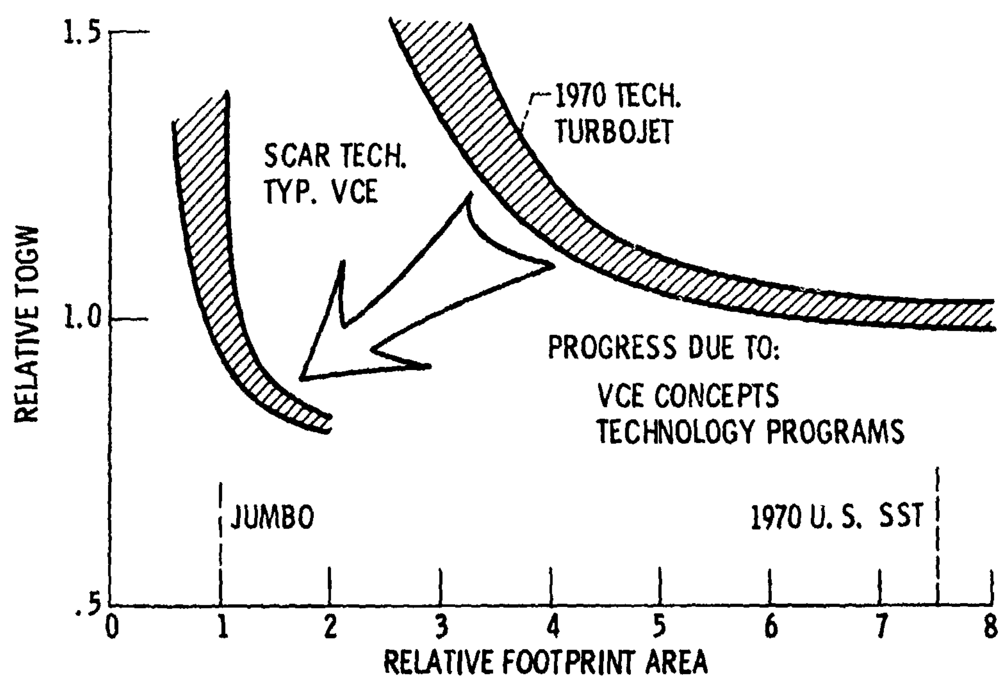


Fig.4 SST progress since 1970

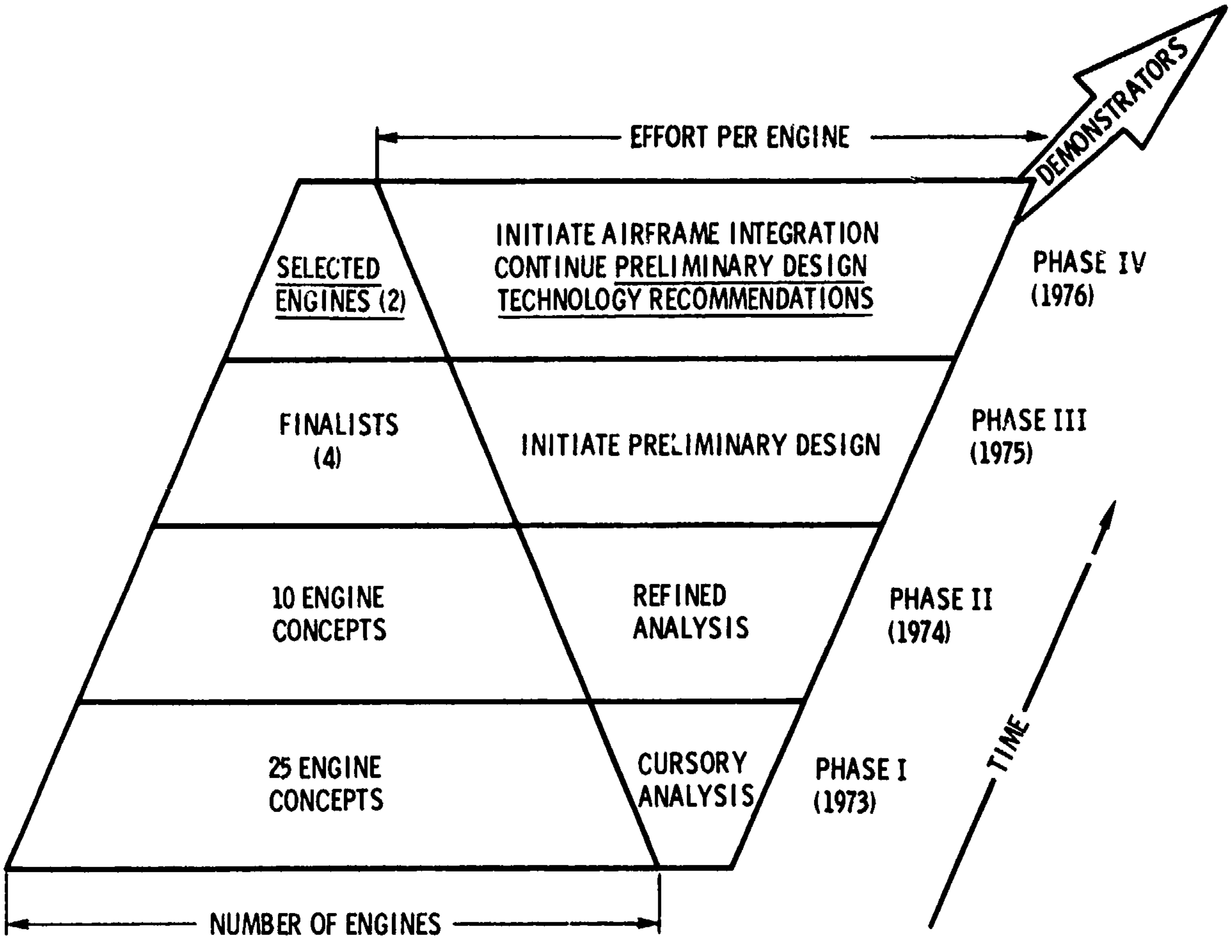


Fig.5 Evolution of SCAR engine studies

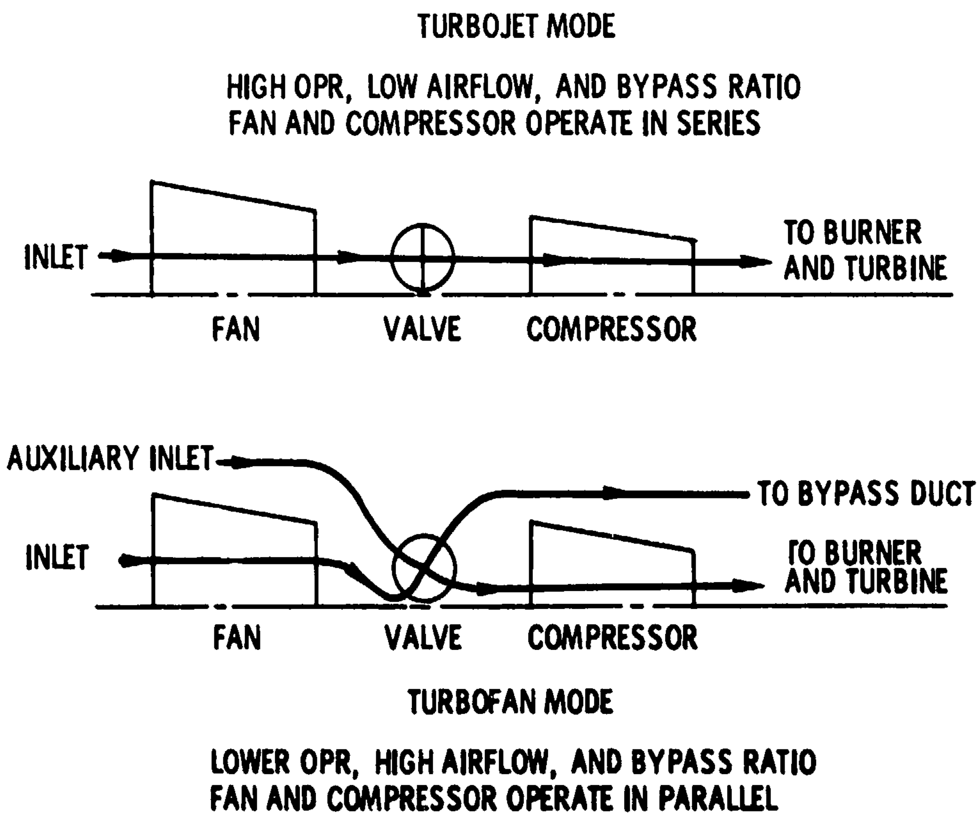


Fig.6 Series-parallel valved VCE concept

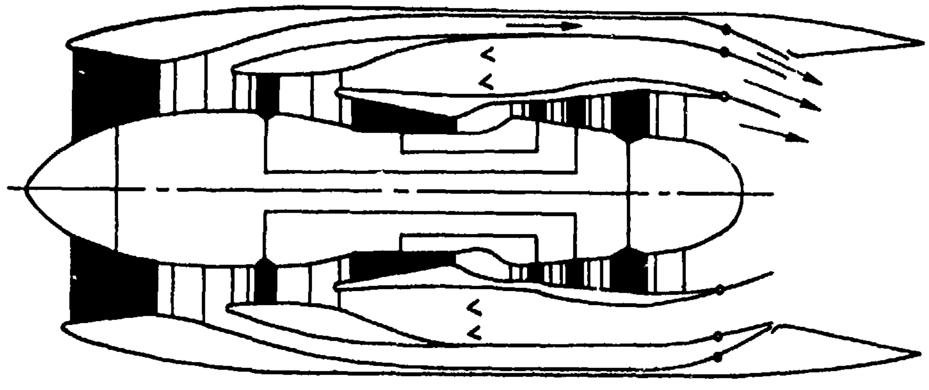


Fig.7 3-Rotor double bypass or modulating airflow engine concept

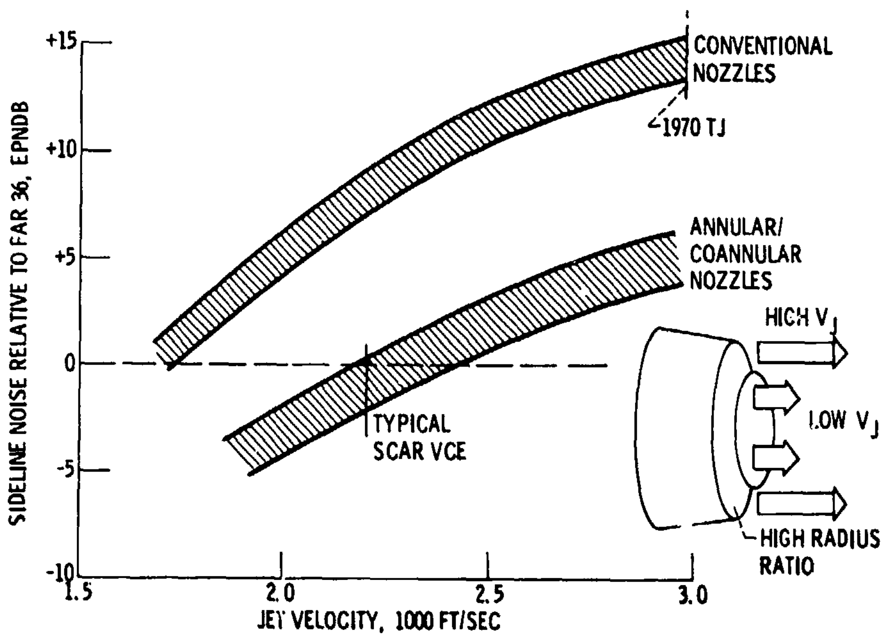


Fig.8 Coannular noise benefit

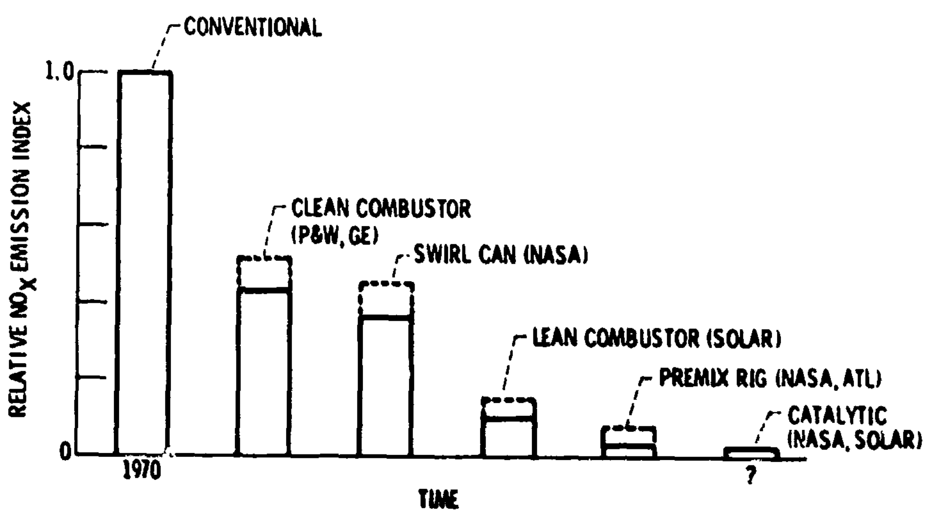


Fig.9 Status of cruise NO_x emission experiments

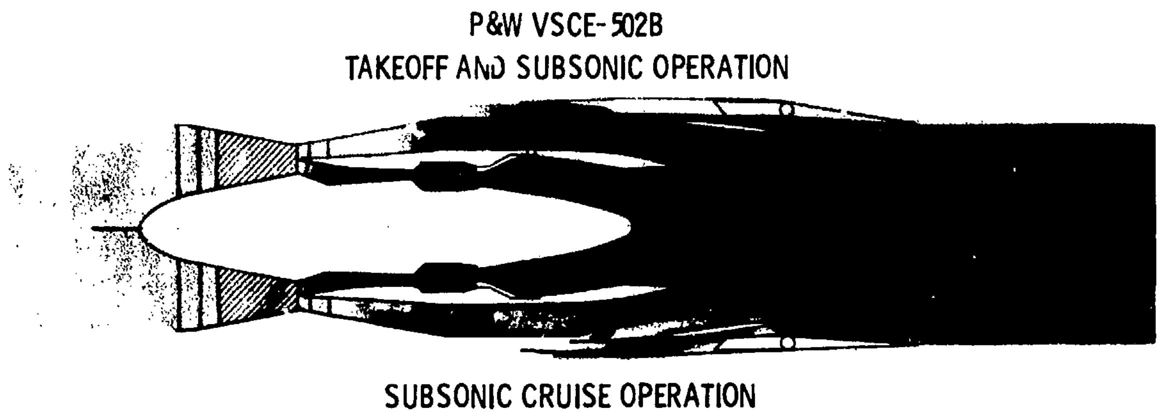


Fig.10 Variable stream control engine

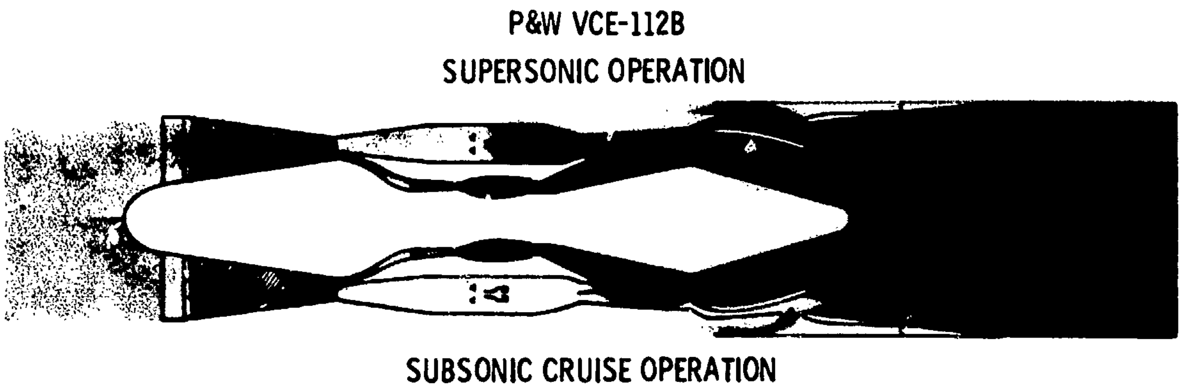


Fig.11 Rear valve variable cycle engine

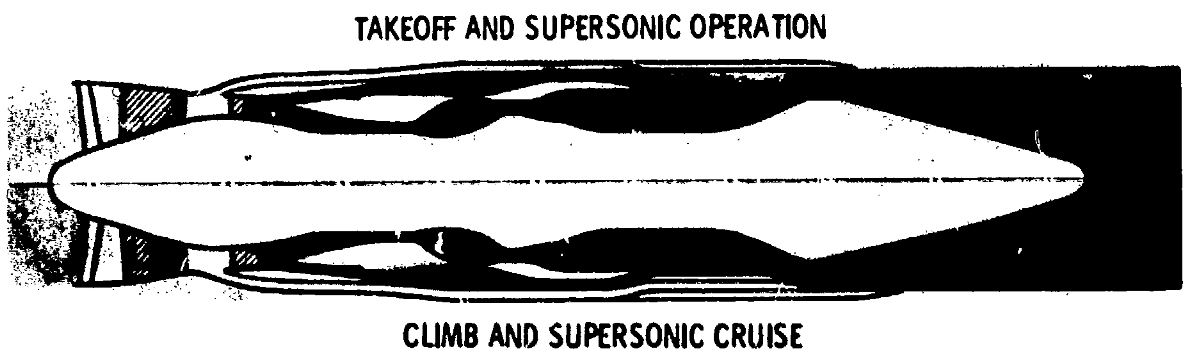


Fig.12 Double bypass engine GE VCE concept

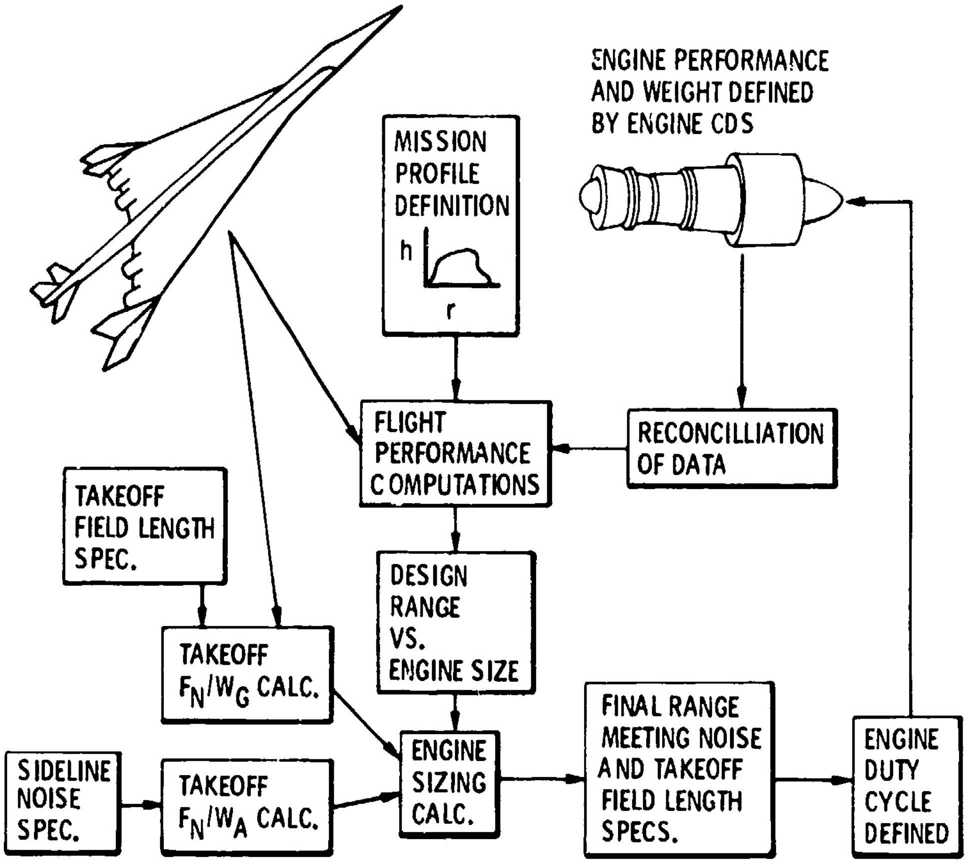


Fig.13 Engine evaluation scheme

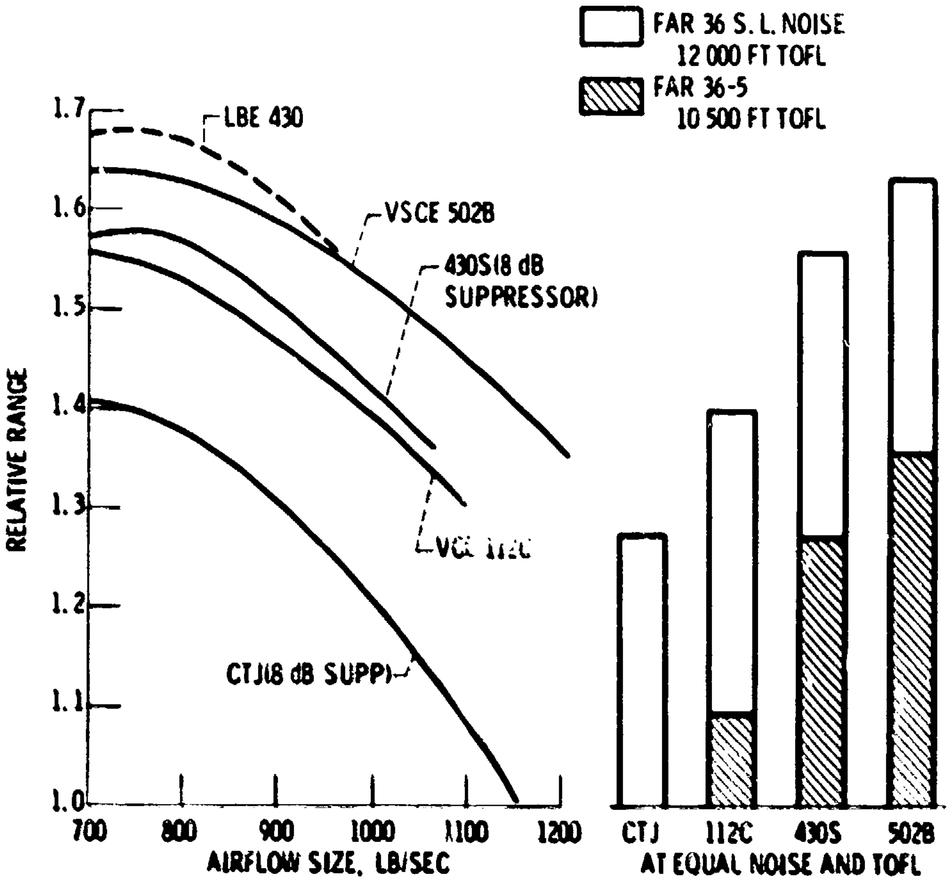


Fig.14 Pratt & Whitney engine comparison

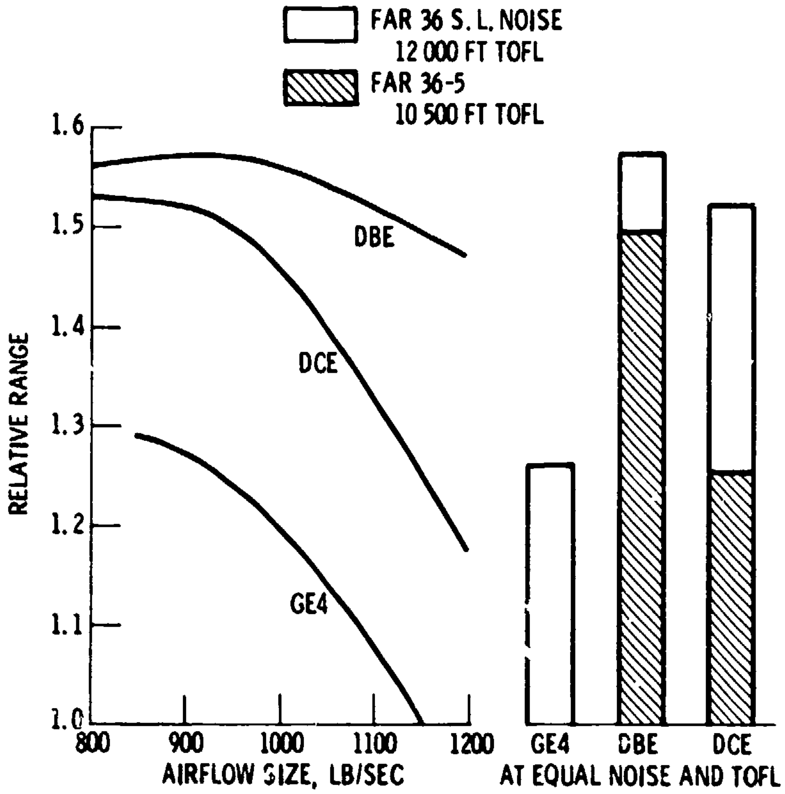


Fig.15 General Electric engine comparison

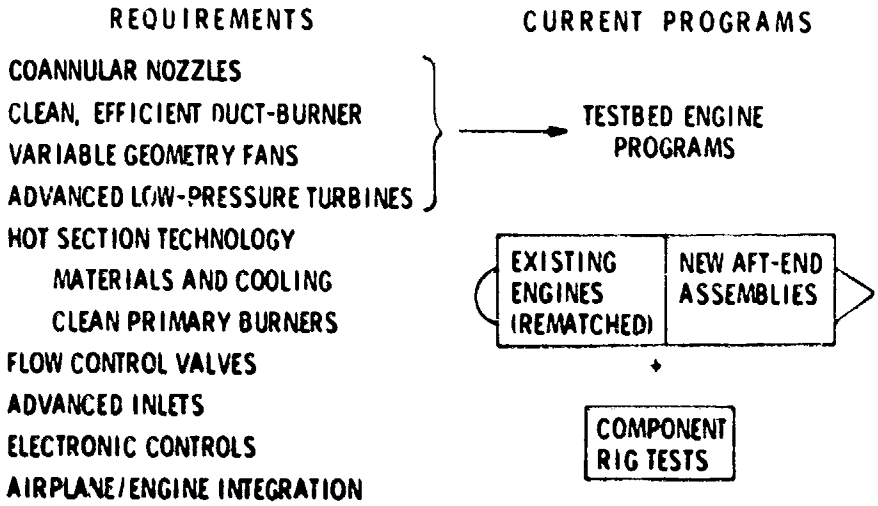


Fig.16 Summary of VCE technology requirements and current programs

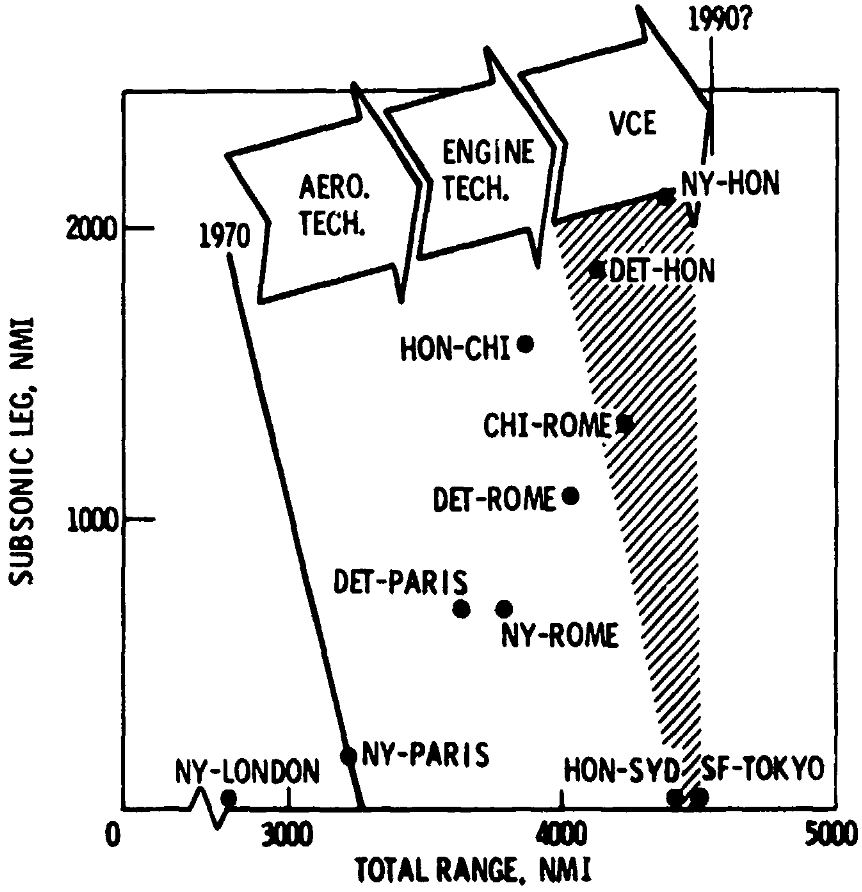


Fig.17 SCAR technology payoffs

DISCUSSION

J.Kurzke

It is known that surrounding a subsonic jet by an about 50 percent slower, cold jet with a bypass ratio of ca 5 will give a noise reduction of about 5dB. The *slower* jet is outside. Now, your report quite the opposite, namely a high speed jet surrounding a slower jet would result in significant noise reduction. Would you explain the different physical phenomena which seem to be the basis for this fact?

Author's Reply

The physical phenomena are not completely understood at this time. Hopefully we will know more definitely in 1½ – 3 years as a result of further testing now in progress or planned. Meanwhile, I am convinced the limited amount of testing done so far has been conducted in a scientific manner. I feel fairly comfortable about using the data in hand but would caution against *extrapolating* into new regions.

Apparently, one finds some noise reduction, for both cases, where outer stream is *slower* or *faster* than core, but none when the two streams have equal speeds.

J.F.Chevalier

Je voudrais poser un question complémentaire de la question précédente. Dans la planche qui compare le bruit des jets co-annulaires et le bruit des "conventional nozzles" – pour la courbe des "conventional nozzles" s'agit-il d'un moteur simple flux qui, pour une vitesse de jet donnée, en abscisse, donne la même poussée que le moteur double-flux VCE examiné? S'agit-il d'un moteur double flux?

Author's Reply

For coannular nozzles, the abcissa refers to the velocity of the outer stream only. The inner stream is moving slower, hence, for a given total mass-flow, there is indeed a loss of thrust. To be rigorously fair about it, we should equalize the thrusts before drawing final conclusions. One way to do this is simply to scale up the coannular system in mass-flow to the desired thrust level. For the cases of concern here, this doesn't decrease the benefit significantly. For example, consider the data shown at 3000 ft/sec on the chart (Fig. 8 of the preprint). Then for the coannular system the inner velocity would be about 2000 ft/sec and the mass-flows of the two streams would be about equal. To give the same thrust as the conventional nozzle does at the original total mass-flow, the coannular nozzle thus has to be scaled up by 20% – which gives a noise penalty of less than 1 dB. In this case, which is quite representative of the comparison between a turbojet and a VCE, the benefit is more realistically 8 rather than 9 dB.

On the other hand, we could also equalize the thrust by reducing the conventional nozzle's jet velocity to about 2500 ft/sec. Comparing this point to the coannular value at 3000 ft/sec on the chart, we still see a benefit of 6 to 7 dB. On either basis of comparison, the reduction of jet noise is significant.

J.F.Chevalier

Ce n'est pas une question. J'ai une remarque à faire. Dans les comparaisons de bruits entre moteurs, il faut faire très attention aux chants sonores de ces moteurs – on peut avoir une réduction de bruit à un certain angle et ne pas l'avoir à un autre angle très important pour le bruit de l'avion et des réductions peuvent être différentes en vol de ce que l'on a au sol. Est-ce que vous pourriez nous dire si il y a confirmation aujourd'hui que, en vol, ces réductions seront obtenues?

Author's Reply

No, we have considerable uncertainty on that point. I mentioned in the preprint that there are some caveats that have to be considered in regard to these results. One being the question that still surrounds the issue of forward velocity effects, the other being simply the question of scale effect. In conclusion I only can say that nobody knows for sure, but based on theoretical considerations we are optimistic.

D.R.Higton

Regarding the VCE schemes that you have presented, and in particular the GE double by-pass engine, could you please comment on what variables could be required to enable the fan to switch from the "low flow" mode to the "high flow" mode?

Author's Reply

All of the VCEs I've discussed would require fans beyond the present state-of-the art. We would expect to use some combination of stator-geometry and rotor-speed control in order to: (a) maintain airflow while throttling, and, maybe (b) to get increased airflow at take-off for lower noise. In the latter case we have to play a very complex game to trade-off fan variability features vs inlet complications. In the case of the DBE we are just starting to work this problem and I can't predict now or when the answers will come out.

NUMERICAL PREDICTION OF THE UNSTEADY FLOW IN VARIABLE GEOMETRY ENGINES - PRELIMINARY INVESTIGATION (*)

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Summary

This paper presents a numerical methodology which enables engineers and designers to get a prediction of the unsteady flow, through turbojet engines, due to the variation in time of the geometry of some engine component. The approach is based on the idea of replacing the actual bladings with actuator discs, which deflect the gas stream according to the real geometry. Other surfaces of discontinuity simulates the in... and the exaust of the engine and the combustion chamber as regards the heat input. The computation is carried on by integrating in time (finite difference method) the hyperbolic partial differential equations which describe the gas motion, with a second order of accuracy integration scheme. Numerical examples show the simulated unsteady flow originated by variation in time of stators stagger angles (compressor and turbine) and of exaust nozzle area.

INTRODUCTION

Large attention is paid today to the so called "variable geometry turbojet engines", owing to the advantages the variable geometry components offer as regards the flexibility of the propulsion system to different operating conditions. During the variation in time of the geometry of some engine element, unsteady flow is originated and pressure waves travel up and downstream along the engine.

The aim of this paper is the presentation of a computational procedure which enables engineers and designers to get useful informations on the flow field through the engine during these transients. It should soon be pointed out that the transients we refer in this paper, may be ascribed to the class of high frequency transients, in comparison with those where the low frequency is related to the relativelylarge inertia of the rotating components. We will then refer here-after to the unsteady flow in the meaning usually accepted in gasdynamics, the one characterized by pressure wave propagation.

Because of the complexity of the flow field in a turbomachine (even under the assumption of steady flow) it is clear that we have tomake a certain number of hypothesis in order to create a model of the machine simple enough and suitable to be studied with reasonable computations, as regards computer times. Some of these assumptions may be probably appear questionable. However we are now at the first stages of this investigation and we are looking for some better ways for modeling the actual complicated phenomena. It should even be kept in mind that too much sophisticated and accurate models may require a large amount of computational points and, therefore, too long computational times.

The numerical procedure, here presented has been tested with some numerical examples describing the unsteady flow originated by the geometry variation in time of engine components such as the stagger angles of stators in the compressor, the stagger angle of the turbine nozzle and the exaust nozzle area.

BASIC ASSUMPTIONS

We neglect here the radial gradients of any fluid-dynamic or geometrical parameter. We assume also axisymmetric flow. We neglect the effects related to viscosity and secondary flows. We assume then, that each blading (of both stators and rotors in the turbomachines) may be replaced by actuator discs; these are surfaces of discontinuity and the continuous aerodynamic loading along the actual blade, is concentrated on them. The two basic phenomena (deflection through the blade and pressure wave propagation along the machine) are splitted: the deflection occurs through the disc, while the wave propagation takes place along the actual axial length of the blade.

The combustion chamber (and the after-burner also) is concentrated in a surface of discontinuity as regards the heat input.

All these assumptions have been made in order to keep as low as possible the number of computational points and, therefore, the computational times. In fact, we did in the past, many numerical experiments where we avoided to make all these assumptions together. Ref. 1 refers to the case where the actual bladings were replaced by a field of forces such that the flow tangency to the blade was imposed at any time and at any point, instead of concentrating the continuous blade loading in the actuator disc. The numerical examples reported there refer to the unsteady flow in axial flow compressors. In ref. 2 we took into account the radial gradients in order to extend the methodology to axial compressor, with high tip to hub ratio bladings. Moreover we are presently working on the prediction of the influence of peripherical distorsion at the inlet of axial compressors and, in this case, we are considering explicitely peripherical gradients., avoiding the assumption of axisymmetric flow. We are also working on models for simulating combustion chamber and afterburner, with a continuous heat release along the engine axis, according to the chemical reaction process.

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Of course all these more sophisticated approaches require a relatively large number of computational points, and computer times may results unacceptable when the prediction of long transients through an entire turbojet engine is required. Moreover we point out that the mentioned hypothesis may fail in describing the unsteady flow in the small local scale, but, we think, it is possible to get a reasonable good description of the most important features of transients in the engine.

Under these assumptions we may then consider the turbojet engine as a sequence of regions with continuous flow, which are separated by discontinuities simulating bladings or heat inputs. The numerical procedure will be then carried on by integrating in time the unsteady flow equations and by computing the flow properties on the two side of each discontinuity, by matching the unsteady flow equations in the neighboring regions with those describing the physical process at the discontinuity.

THE EQUATIONS OF THE MOTION

According to the above mentioned assumption, the problem is formulated in terms of one-dimensional, unsteady, inviscid, compressible flow. Because of the tangential component of the velocity (v), we should add, to the usual set of equations, the tangential momentum one. All the parameters are normalized with respect to reference values: pressure p_∞ and temperature T_∞ in the front of the engine, reference velocity $c_\infty = \sqrt{p_\infty/\rho_\infty}$, reference length (axial length of the engine) l_∞, reference time $t_\infty = l_\infty/c_\infty$.

The flow equations are:

$$
(1)\qquad \begin{cases} P_t + uP_x + \gamma u_x + \gamma u\alpha = 0 & \text{(continuity)} \\ u_t + uu_x + TP_x = 0 & \text{(axial momentum)} \\ v_t + uv_x = 0 & \text{(tangential momentum)} \\ S_t + uS_x = 0 & \end{cases}
$$

where: $P = \ln (p/p_\infty)$; $T = \exp\left(\frac{\gamma-1}{\gamma} P + \frac{S}{\gamma}\right)$

$$\alpha = \frac{dA/dx}{A} \quad \text{(duct divergence)}$$

The set of Eq. 1 represents hyperbolic partial differential equations. By working on the first two equations, one may find two characteristic lines with slopes:

$$(2)\qquad \lambda = u \pm a \; ; \; (a = \sqrt{\gamma T})$$

The compatibility equations along these lines are:

$$(3)\qquad (P_t + \lambda P_x) \pm \frac{\gamma}{a} (u_t + \lambda u_x) = - \gamma u\alpha$$

The third and the fourth equations give a third characteristics line; its slope is:

$$(4)\qquad \lambda' = u$$

Two compatibility equations hold on it:

$$(5)\qquad v_t + \lambda' v_x = 0 \quad ; \quad S_t + \lambda' S_x = 0$$

Eq. 1 are used for computing the variation in time of pressure (P), the two components of the velocity (u,v) and entropy (S) in the continuous flow regions bounded by the surfaces of discontinuity. Eq. 1 are treated according to the finite difference method. The integration follows the two level (predictor - corrector) scheme suggested by Mc Cormack. This algorithm is used at all the interior points; different procedure are adopted at those computational points which are located at the discontinuities.

COMPUTATION AT THE DISCONTINUITIES

We discern different discontinuity surfaces in relation with the physical element replaced by the discontinuity: we have then inlet and exaust discontinuities where the unsteady flow inside the engine is matched with the outer flow, actuator discs replacing stators and rotors of the turbomachines and discontinuity surfaces to simulate the heat input (combustion chamber or afterburner).

At each discontinuity the algebric quasi-steady relationships, which describe the flow evolution through it, are written and differentiated in time; these equations are then matched with the flow equations, written in the compatibility forms (Eq. 3 and 5). The resulting derivatives in time of the flow properties at the discontinuity are then integrated in time, following the same integration scheme as at the interior points. We show here the detailed procedure at each boundary.

a) <u>Inlet</u> (fig. 1)

The quasi steady relationships at the inlet are provided by:

$$
(6) \quad \begin{cases} T_1 + \frac{\gamma-1}{2\gamma} u_1^2 = T_o + \frac{\gamma-1}{2\gamma} u_o^2 \\ S_1 = S_o \\ v_1 = 0 \end{cases}
$$

where the subscript "o " refers to the free stream conditions, which may vary in time as prescribed inputs.

Eq. 6 are differentiated in time and matched with the compatibility equation (3) on the left running characteristic (u-a). The time derivatives of the velocity (u_{1t}) is obtained and integrated in time, according to the two-level scheme as well as at the interior points. The updated value of the pressure (P_1) follows from Eq. Eq. 6.

b) Bladings (Fig. 2)

The quasi-steady relationships at the disc, which replaces the blading, are:

$$
(7) \quad \begin{cases} \frac{u_1 P_1}{T_1} = \frac{u_2 P_2}{T_2} & \text{(continuity)} \\ \left[T_2 + \frac{\gamma-1}{2\gamma}(u_2^2 + v_2^2)\right] = \left[T_1 + \frac{\gamma-1}{2\gamma}(u_1^2 + v_1^2)\right] + \frac{\gamma-1}{\gamma} \sigma\omega \, (v_2 - v_1) & \text{(work balance)} \\ S_2 = S_1 \\ v_2 = \sigma\omega - u_2 \tan\beta & \text{(geometrical condition)} \end{cases}
$$

The second term, on the right hand side of the second equation, accounts for the mechanical work exchanged between the blades and the gas stream. The blade speed is represented by ω (for rotor $\sigma = 1$, and for stator $\sigma = 0$). In the third equation we assume here that no losses occur through the blanding; this is not a very restrictive condition and the procedure may be easily implemented by introducing the relative total pressure losses, depending on the incidence, once experimental data on cascades are provided. On the fourth equation, the flow tangency at the trailing edge is imposed. (Fig. 3); deviation effects may be taken into account (as for the losses), if, even here, experimental data are provided.

Eq. 7 may be differentiated in time. On the other hand, four compatibility equations (Eq. 3.5) (three in the upstream region and one down stream of the disc) are given on the corresponding characteristic lines (Fig. 2). The set of all these equations enable us to evaluate the time derivatives of the flow properties in front of the disc (P_{1t}, u_{1t}, v_{1t}, S_{1t}), which may be integrated in time as usually. The downstream flow conditions follow from Eq. 7.

This procedure holds in case of unchoked flow through the blading.

However if, by computing the upstream flow, the mass flow rate becomes larger than the critical value (corresponding to the area at the trailing edge), the computation at the point 1 (Fig. 4) is carried on as it follows. The continuity equation is written by introducing the critical condition and, therefore, unaffected by the downstream pressure:

$$
(8) \quad \frac{u_1 P_1}{T_1} = f(P^{\circ}_{1\ rel}, T^{\circ}_{1\ rel}, \beta)
$$

where $P^{\circ}_{1\ rel}$ and $T^{\circ}_{1\ rel}$ denote the total values of the flow relative to the blading. Eq. 8 is differentiated in time and the flow at the point 1 is computed, by taking into account the compatibility equations upstream of the disc. As regards the point 2, we write the continuity equation:

$$
\frac{u_2 P_2}{T_2} = \frac{u_1 P_1}{T_1}
$$

We assume here isentropic process, so that T_2 in easily related to p_2. This equation is differentiated with respect to the time, and then matched with the compatibility equation on the downstream region (u-a). One may now compute the updated values of P_2 and u_2. The modulus of the relative velocity w_2 at the trailing edge follows as:

$$
w_2^2 = \frac{2\gamma}{\gamma-1} (T^{\circ}_{1\ rel} - T_2)
$$

The direction of w_2 will be, in general, different from the one given by β. A similar model is suggested in Ref. 3.

Finally the downstream tangential velocity is computed as:

$$
v_2 = \sigma\omega - \sqrt{w^2 - u^2} \cdot \frac{\beta}{|\beta|}
$$

It is clear that this, or similar procedure, can not be applied in the case of supersonic axial flow.

c) Combustion Chamber (Fig. 2)

We have here the following quasi-steady relationships:

$$(9)\quad \begin{cases} \dfrac{u_1 p_1}{T_1} = \dfrac{u_2 p_2}{T_2} & \text{(continuity)} \\ p_1 \left(1 + \dfrac{u_1^2}{T_1}\right) = p_2 \left(1 + \dfrac{u_2^2}{T_2}\right) & \text{(momentum)} \\ \dot{m}_a \, \gamma \, (T_2^\circ - T_1^\circ) = H_i \cdot \dot{m}_b & \text{(energy)} \end{cases}$$

The energy equation takes into account the heat release ($\dot{m}_a$ and $\dot{m}_b$ represent the air and fuel mass flows and H_i the fuel heating value refered to c_v T_*). As usually, Eq. 9 are differentiated in time and matched with the four compatibility equations along characteristics.

d) Exaust Nozzle (Fig. 4)

The quasi steady relationships are:

$$(10)\quad \begin{cases} \dfrac{u_1 p_1}{T_1} = \eta \dfrac{u_2 p_2}{T_2} & \text{(continuity)} \\ T_1 + \dfrac{\gamma-1}{2\gamma} u_1^2 = T_2 + \dfrac{\gamma-1}{2\gamma} u_2^2 & \text{(energy)} \\ S_1 = S_2 \\ v_1 = v_2 = 0 \end{cases}$$

The constriction factor η of the nozzle may vary in time as prescribed (for example during the afterburner starting procedure). Even now Eq. 10 are differentiated in time and matched with the compatibility equations along the right running characteristics $(u + a, u)$. The value of p_2 is kept equal to the external pressure. This procedure in used for unchoked flow. In case of choking, the pressure level p_2 is imposed as the critical value with respect to the total pressure at the point 1.

NUMERICAL SIMULATION OF TRANSIENTS

We have chosen a simple scheme of turbojet engine: a six stages axial compressor is driven by a single stage axial turbine.

The distribution of the computational points as well as the location of the discontinuities is shown in Fig. 5. On the base of this scheme of the engine we did, in the past, some computations related to the starting operation of the engine and to the after burner ignition; the results of these numerical simulations are reported in Ref. 4.

In the present paper we report the results of different computations, describing transients which may occur in variablegeometry engines.

Steady configurations in the engine are assumed as "initial values" at time zero. The change in geometry of some component is supposed to take place in a given range of time and this fact originates pressure waves travelling along the engine. As result of this transient, all the thermodynamic properties of the flow will vary in time. In particular the air mass flow rates at the inlet and exit of the engine show a difference because of air mass storage inside the engine; the combustion temperature and the thrust fluctuate; the instantaneous torques on the compressor and turbine become unbalanced and the rotor accelerates according to the dynamic law. Once the transient vanishes a new steady configuration is acieved. We have simulated three different transients:

- Run n. 1: the stagger angle (θ_s) of the first two stages in the compressor is varied by a deflection of 30°.
- Run n. 2: the exit angle of the turbine nozzles (θ_N) is changed by 10°, at the same time the fuel injected in the combustion chamber is reduced by a factor of 20%.
- Run n. 3: the area of the exaust nozzle is reduced from a constriction factor of .70 to .65.

The are many interesting values, as output of these computations, which may be shown. We have reported only few of them in Fig. 6 (Run n. 1), 7 (Run n. 2), 8 (Run n. 3) as function of the time. It may be noted that the new steady configurations are almost achived for time nearly equal to 5 or 6.

These figures report:

- the input data which originate the transients:

i) the stagger angles (θ_s) in the first two rotors of the compressor (Fig. 6)

ii) the geometrical exit angle of the turbine nozzle (θ_N) and the fuel mass flow rate ($\dot{m}_b$) (Fig. 7)

iii) the constriction factor η of the exaust nozzle (Fig.8).

- the air mass flow rates at the inlet ($\dot{m}_{ai}$) and at the exit ($\dot{m}_{ae}$) of the engine.
- the instantaneous torques required by the compressor (TQ_c) and given by the turbine (TQ_t)
- the rotor peripherical speed (ω)
- the total temperature (T_b°) at the turbine inlet.
- the thrust computed as all the forces acting, in the axial direction, on solid walls and bladings.

It should be mentioned that in the Run N. 2 (Fig. 7) the geometrical angle θ_N (trailing edge of the turbine nozzles) determines the air angle so far the nozzle operates as unchoked; in the case of critical flow, however, the effective air angle θ_N may differ from θ_N, according the physical arguments and the procedure reported above (computations at the bladings).

Finally we make clear that the inertia of the rotors we have assumed in our computations, is order of magnitude lower than the actual ones, but such a low value allowed us to test the methodology also during accelerating operations, without requiring too long computer times.

CONCLUSION

A method has been presented for the numerical simulation of high frequency transients, which may occur in variable geometry turbojet engines, due to the change in time of the geometry of some components. A complete description of the transients may be achieved from numerical computations and a very large amount of informations are available.

Some drastic assumptions have been made, in modeling the actual complicated flow, and some input numbers (very low inertia of the rotors, few stages in the turbomachines, etc) may appear unrealistic. However this paper represents just only a preliminary step of the investigation and we are working on more sophisticated and reliable models. The final goal is the computation of transients in complete engines, including the interference, between the engine and the inlet (See Ref. 5) in propulsion systems for supersonic aircrafts.

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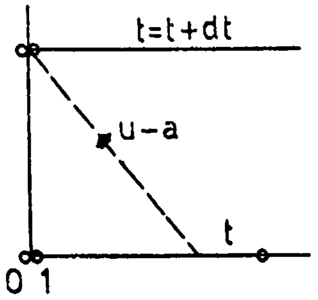

FIG.1

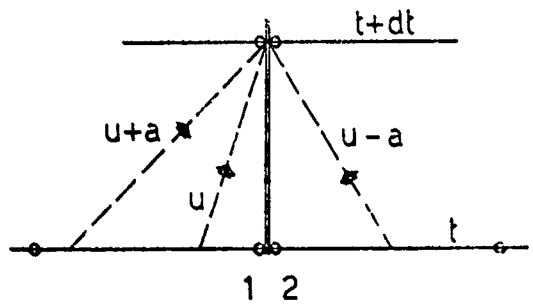

FIG.2

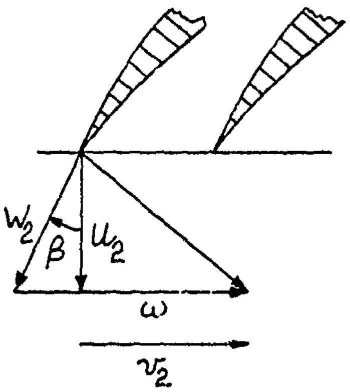

FIG.3

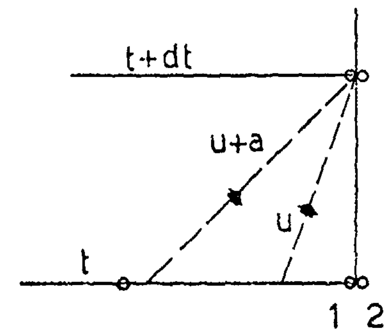

FIG.4

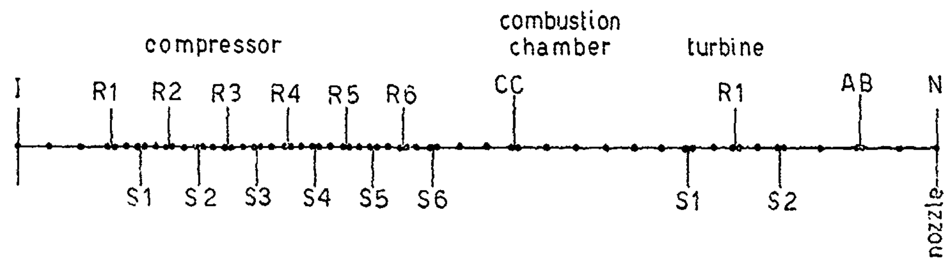

FIG.5

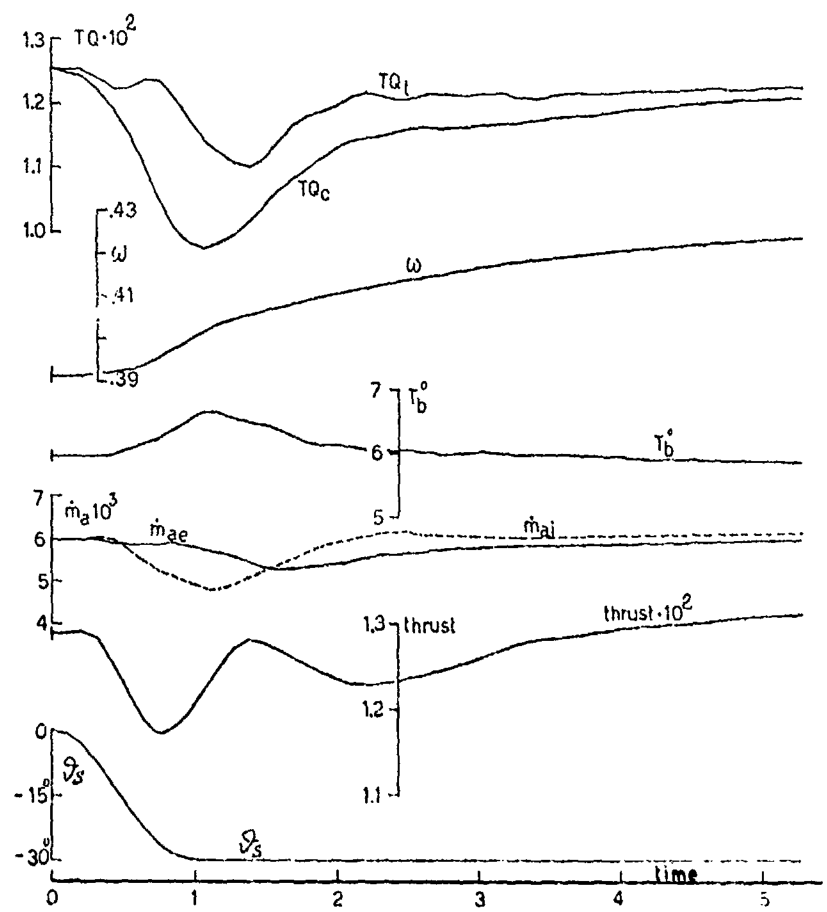

FIG.6

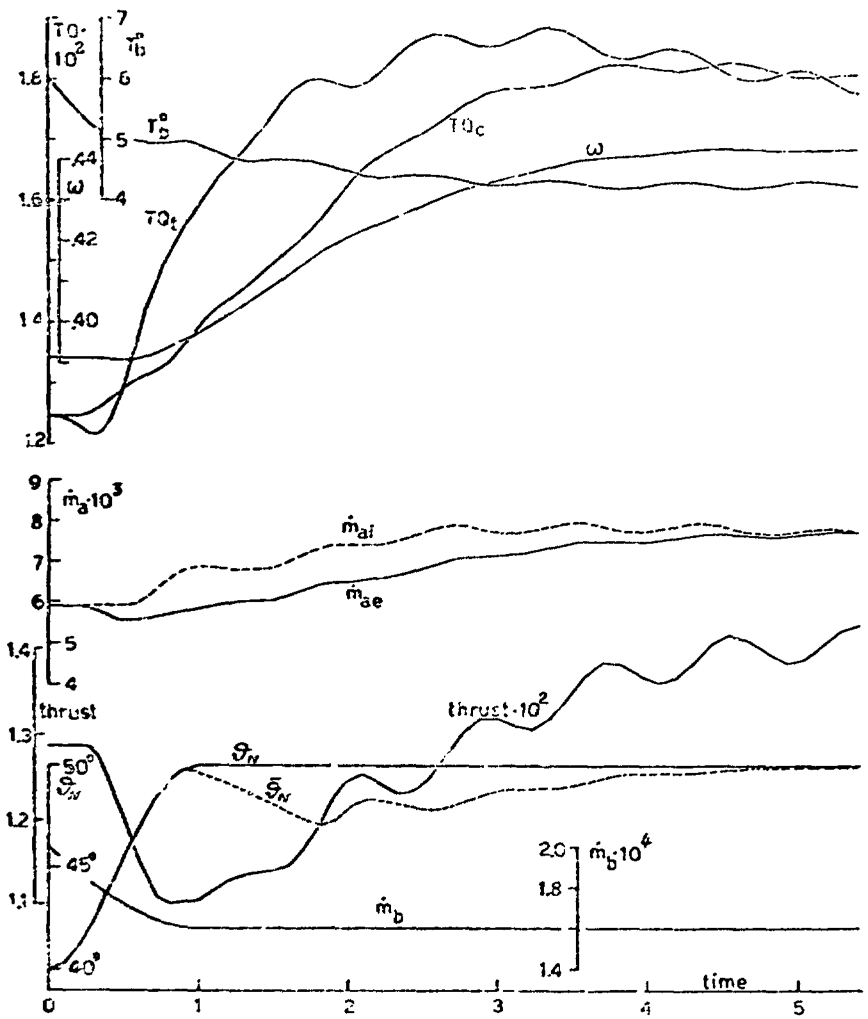
TQ·10²
Tb°
ω
TQc
TQt
ṁa·10³
ṁai
ṁae
thrust
thrust·10²
ϑN
ṁb
ṁb·10⁴
time

FIG.7

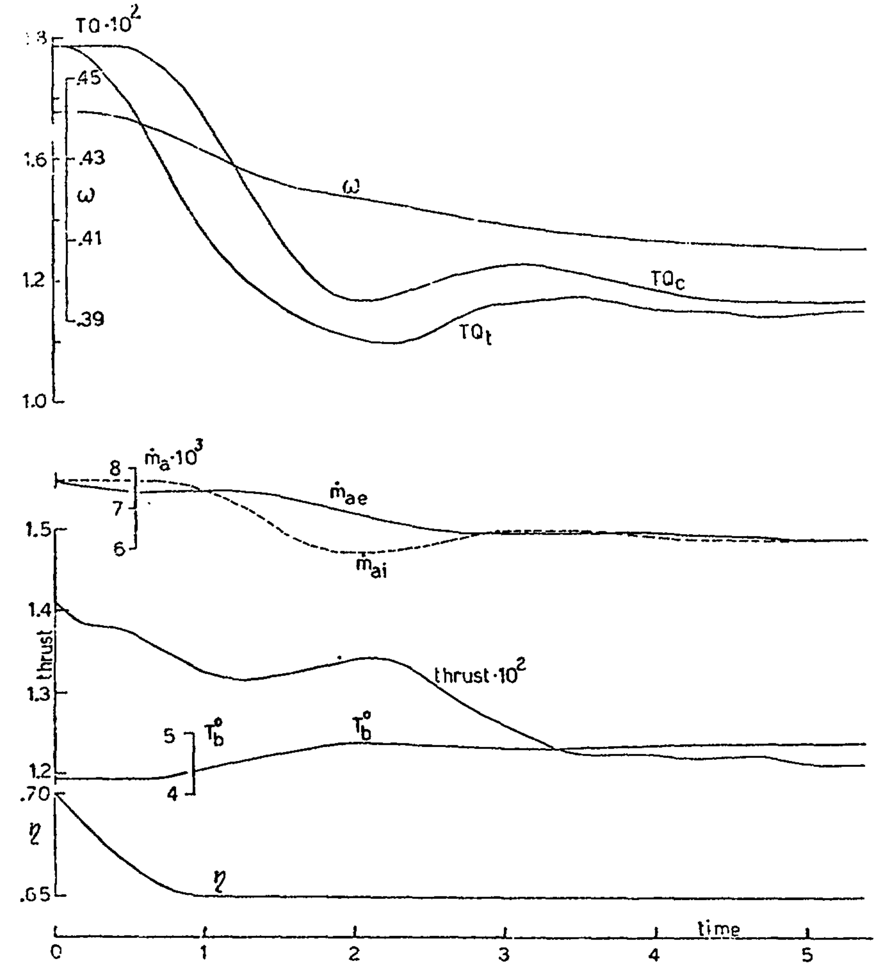
TQ·10²
ω
TQc
TQt
ṁa·10³
ṁae
ṁai
thrust
thrust·10²
Tb°
η
time

FIG.8

DISCUSSION

J.F.Chevalier

Je voudrais faire une remarque. Je trouve que c'est assez sympathique que dans une Université on fasse des calculs qui ont pour but d'être assez simples pour représenter approximativement ce qui se passe dans un moteur. Nous voyons trop souvent des Professeurs parler de choses trop sophistiquées pour que nous puissions les utiliser. Ces calculs que vous avez faits me semblent intéressants. En particulier, par exemple, pour étudier les problèmes de "scritch" et de couplages éventuels de modification de conditions de scritch dans une vibration de post-combustion, couplée avec la combustion elle-même; il me semble que votre calcul peut servir à établir de bonnes conditions limites dans un tel calcul et donc serait très utilisable par des chercheurs en turbomachines.

Quel est le temps de calcul et sur quelle machine faites-vous un tel calcul?

Author's Reply

The computer time necessary is depending on the kind of machine you are using. We performed this kind of computation on a IBM370/158 or CDC/6600. The computational time does not mean too much except if you compare the computational times with real times, that means to compare the computational times with the times related to physical problems. If I say to you three minutes with CDC/6600, it does not mean too much because we have to look at what are the physical time we are investigating on. I think that it will be quite good to have a computation which will be performed in real time. However, we are at the moment quite far from this goal. In fact the computational times are, on a CDC 6600, about 60 times larger than the real physical times.

R.E.Peacock

In Figure 7 the integration with respect to time between what appear to be two equilibrium conditions ($t = 0$ è $t = 5$) of the curve of m_{ai} and that of m_{ae} do not give the same result. Apart from possible changes of fuelling level, why is this so?

Author's Reply

I think one should not expect equal integrals in both cases. In fact the difference is just the air which has been stored in the engine from one state to the new state of configuration. In other words, the difference is directly related to the different distribution of density in the engine, in the two different steady conditions.

ASSESSMENT OF VARIABLE-CYCLE ENGINES FOR SUPERSONIC TRANSPORTS

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SUMMARY

The NASA Supersonic Cruise Aircraft Research (SCAR) program has sponsored extensive work to define technology improvements which could lead to an economically and environmentally viable advanced supersonic transport. One element of the program involved the generation of a multitude of advanced conventional and variable-cycle "paper" engines by the engine manufacturers and screening them in a typical transport mission to find the most promising engine cycles. These latter cycles were evaluated by the airplane manufacturers in more detailed mission studies with the result that three promising candidate engine cycles were identified for further study and refinement. The present paper evaluates each of these three proposed SCAR propulsion systems in terms of relative aircraft range for a fixed payload and take-off gross weight with a design cruise Mach number of 2.7. In order to put the performance of these engines in perspective, a comparison of these engines and the former U.S. SST engine (GE4) is made with an idealized variable-cycle engine whose performance at all operating points matches that of an optimized point-design cycle within specified limits. In addition, range comparisons are made with and without noise level constraints to determine the influence of noise upon cycle selection. Finally, the critical areas requiring new or improved technology for each cycle are delineated.

1.0 INTRODUCTION

With the cancellation of the United States Supersonic Transport (SST) program in 1971, an enormous concentrated developmental effort for civil supersonic flight technology came to an abrupt halt in the U.S.A. This did not deter dedicated engineers from examining the problems which led to the demise of the U.S. SST and prophesying new technology requirements for a second-generation SST which would lead to substantial improvement in performance, economics, safety, and social acceptability. A paper presented by Nichols later in 1971 (Ref. 1) indicated that substantial range improvements were possible with advances in aerodynamics, structures, materials, propulsion, and flight control within the restraints imposed by take-off noise considerations. Although the gains shown were those for an advanced dry turbojet engine equipped with a noise suppressor, Nichols called for inventiveness to define a variable-cycle engine to "have the airflow characteristics at take-off of the turbofan combined with the good cycle efficiency of the turbojet in supersonic cruise." As a result of studies such as this, NASA in 1972 elected to establish a low-keyed effort now known as the Supersonic Cruise Aircraft Research program (SCAR) to define, foster, and fund research efforts to develop the technology needed to support any future attempt to build a second-generation SST.

Shortly after the start of the SCAR program, Swan (Ref. 2) indicated that a weight reduction equivalent to that of the entire payload would have been possible for the U.S. SST had a variable-cycle engine been available. The implication was that the variable-cycle engine would be capable of generating a large airflow in a turbofan mode with low specific thrust levels to meet both the take-off field length and regulated noise level without a suppressor. It would cruise supersonically as a dry turbojet and would maintain high inlet flows when operating at part power to eliminate throttle dependent spillage, bypass, and boattail drag. Swan further made the point that "the propulsive system concept must be treated as an entity, including inlet and exhaust systems such that reduced weight, drag, and complexity of these latter components may be traded for increased weight and complexity of the variable cycle." At about the time Swan presented his paper, the results of Boeing's JT8D variable bypass engine test (Ref. 3) were made known to the staff of NASA. This test demonstrated the ability to increase airflow 70% and vary the bypass ratio from 1.1 to 3.5 through the use of an air inverter valve. Partly as a result of this information, the on-going SCAR engine studies performed under contract with General Electric and Pratt & Whitney, and directed by NASA Lewis Research Center, were expanded to include studies of a family of unconventional variable-cycle engines. The results of these studies, which are still underway, are described in Reference 4.

The purpose of this paper is to assess, on an integrated mission basis, the performance of three variable-cycle engine concepts resulting from the on-going SCAR program and to delineate those areas of technology which must be developed to achieve such performance. The engine cycles selected have differing degrees of variability and complexity as well as differing advantages and disadvantages with respect to each other. The figure of merit employed is the maximum range achieved at a cruise Mach number of 2.62 on a hot day for a given take-off gross weight and payload. The results are compared with a fantasized completely variable-cycle engine which represents an upper bound of the potential for a variable-cycle engine. In addition, comparisons are made with the GE4, the engine selected for the U.S. SST, to illustrate the improvements afforded by advanced engine technology. These comparisons are made for optimum performance-sized engines and for vehicles with engines sized to meet FAR noise regulations both with and without suppression.

The engine data are as supplied by the engine manufacturers, and no independent evaluation of the validity of the data or ability to perform as specified has been made. Where opinions are expressed in the paper, they represent those of the authors alone.

2.0 ENGINE SELECTION AND DESCRIPTION

Many engine cycles of both conventional and unconventional types were generated and examined by the Pratt & Whitney and General Electric companies under the auspices of the NASA/SCAR program. Each of these engines was screened in a mission simulation program to pinpoint the more desirable cycles. Mission simulation was a necessary tool since it is well known that comparisons of the usual performance parameters of thrust-weight ratio, specific fuel consumption, installed thrust level are not necessarily indicative of the best cycle in view of the conflicting requirements imposed by noise restraints, weight, thrust margin, and subsonic versus supersonic cruise fuel consumption rates. The performance data of the higher ranked engine cycles were provided to the airplane companies to evaluate in their SCAR-sponsored system studies. Based upon the results of these studies, the most promising cycles were further refined and analyzed in more detail by the engine manufacturers. The net result of this iterative process was the definition of three candidate variable-cycle engines of differing degrees of variability. For each of these engines, a technology base available for a certificated engine in the late 1980's was assumed. For conventional engine components, an extrapolation based upon historical data was used by the engine manufacturers to predict weight and performance. For nonconventional components, estimates of performance and weight were based upon results using small-scale models wherever possible and in all cases with a degree of restrained optimism in keeping with an objective of the SCAR to delineate potentially attractive areas for new technology research.

To establish bench marks against which to assess the performance of these candidate variable-cycle engines, the aircraft performance with two alternate engines was generated. The first is the engine selected for the former U.S. SST (the GE4/J5P). It is included to indicate the gains possible due to technology advances since 1969 in conventional components and due to cycle variability. The second engine used for reference is in reality not an engine. It represents, within limits, a cycle optimized over the flight spectrum for best fuel economy without regard to physical restraints. The five engine cycles are briefly described below. Each of the variable-cycle engines have been optimized in terms of overall pressure ratio and a fan pressure ratio for a standard day flight Mach number of 2.7 for direct comparison with the GE4 which was designed specifically for this flight Mach number. The engine cycle parameters and performance are listed in Table 1.

2.1 Pratt & Whitney — Variable Stream Control Engine

The variable stream control engine (VSCE), shown in Figure 1, is a two-spool duct-burning turbofan employing a convergent-divergent ejector nozzle. In essence, it is very similar to the JTF17, the P&W entry for the U.S. SST program but differs primarily in employing higher turbine inlet temperature, a variable area throat for the primary stream, and a greater degree of variable geometry in the fan and compressor. The use of this variability permits a more complex throttle schedule to be used. This throttle schedule essentially matches the engine and inlet flow schedule at maximum dry and augmented power settings at all flight Mach numbers to minimize spillage, bypass, and boattail drag. In addition, for take-off, the primary burner is throttled back with the duct burner lit and full airflow maintained to achieve a tailored exhaust gas profile. This technique maximizes the coannular noise relief at the required take-off thrust level. A detailed description of the engine and explanation of the coannular noise relief are contained in Reference 5. The VSCE represents a conservative approach toward achieving the objective of a variable-cycle engine. Its performance at both supersonic and subsonic cruise conditions is quite similar to that of a conventional duct-burning turbofan.

2.2 Pratt & Whitney — Rear Valve Variable-Cycle Engine

The rear valve variable-cycle engine (RVE) has been found to yield the most attractive application of the air inverter valve concept. It is used as a means of cycle conversion from turbofan to turbojet and vice versa. A description of the air inverter valve and its use in this and other arrangements is given in Reference 3. The operation of the RVE is described in detail in Reference 5 and is briefly reviewed here. The RVE is a two-spool nonafterburning engine employing a variable geometry fan and a split low-pressure turbine and incorporates a convergent-divergent ejector nozzle. The air inverter valve functions as a diverter/mixer and is located before the last element of the low-pressure turbine. In the twin turbojet mode, the duct burner is lit and the valve is in the inverting position such that the core flow is bypassed around, and the heated duct flow expanded through the rear low-pressure turbine. In the turbofan mode the valve is used to mix the unheated duct flow with the core flow before expanding through the rear turbine element. The inner stream nozzle throat is fixed. In the turbojet mode of operation, variation of the outer stream nozzle throat area and fan burner temperature are used to maintain constant corrected airflow at supersonic cruise part power thrust levels. Airflow regulation in the turbofan mode is uniquely defined by turbine inlet temperature since all the flow exits through the fixed area inner stream nozzle throat. Thus, in the turbofan mode a greater degree of spillage must exist compared to the turbojet mode. This is a result of lower fan speed due to the lower flow energy level at the rear turbine because of the mixing of both streams. The RVE exhibits the greatest variability of any variable-cycle engine in this group in that it operates like a turbofan engine at subsonic cruise and as a turbojet at supersonic cruise speeds.

2.3 General Electric — Double Bypass Variable-Cycle Engine

The double bypass variable-cycle engine (DBE) is a low bypass-ratio two-spool mixed-flow afterburning turbofan engine. The fan is divided into two separate elements. These elements are designed so that engine air can be bypassed downstream of each element. The configuration is shown schematically in Figure 1 and is described in detail in Reference 4. The DBE engine used in the present investigation is a later version of the engine described in Reference 4. In this later version, both bypassed streams are mixed and a portion of the mixed flow is exhausted through an auxiliary nozzle in the take-off and low-speed cruise modes. For take-off, variable turbomachinery geometry is used to overspeed the fan and increase the airflow approximately 20%. This overspeeding in combination with the translating shroud convergent-divergent plug nozzle (Ref. 6) and annular noise relief (Refs. 7 and 8), significantly reduces the jet noise as compared to a conventional C-D nozzle equipped low bypass ratio turbofan engine. The engine throttle modulates the variable stators and bypass flow paths to provide inlet engine airflow match at all flight Mach numbers at both maximum dry and augmented thrust levels and to provide full airflow down to approximately 50% of dry thrust. The DBE represents a degree of variability midway between the VSCE and RVE.

2.4 General Electric — GE4

The GE4 engine, which is used to illustrate the technology base available in 1969, is a single spool afterburning turbojet equipped with a convergent-divergent two-stage ejector nozzle. The engine has a variable area nozzle and employs a two-position compressor stator schedule. The engine weight and performance parameters were taken from the model specifications for the GE4/J5P for a standard day.

2.5 Reference Variable-Cycle Engine

The reference variable-cycle engine (RE), as stated previously, is not a specific engine. It represents a series of point-design cycles, each optimized, within limits, to achieve minimum fuel consumption. Its purpose is to provide a goal for the class of variable-cycle engines to strive for, yet never reach. Admittedly, the RE presents impossible hardware problems and is intended only to serve as a measure of the ultimate range obtainable within the restraints imposed as indicated below.

It is assumed that at Mach numbers below 1, the cycle would be that of a simple turbofan with a bypass ratio of 3 as an upper limit. Above Mach number of 1, it would transform into a dry turbojet. At selected Mach numbers and altitudes, the overall pressure ratio and fan pressure ratio were varied at a constant combustor exit temperature of 1537° C (2800° F) to find the minimum specific fuel consumption through the use of a design point engine cycle program. If the minimum specific fuel consumption occurred at an overall pressure ratio above 45 or compressor discharge temperature exceeding 727° C (1340° F), the specific fuel consumption value at these limit points was assumed as the minimum value. The component efficiencies and pressure losses for this engine were those assumed achievable in the mid-1980's. The exhaust nozzle gross thrust coefficient was held at a value of 0.985 at all flight conditions.

The engine airflow schedule is presumed to vary such that the thrust would match that of the afterburning GE4 for climb and acceleration. At all other conditions, the airflow varies to produce the thrust required by the airplane. At take-off, the engine is high flowed to produce take-off thrust without exceeding the FAA noise regulations. It is further postulated that the weight of the engine pod would be equal to that of the GE4/Boeing installation scaled to 110% of the required airplane cruise airflow rate.

3.0 MODELS AND METHODS

A comparison of the various variable-cycle engines can best be achieved if the airplane-engine characteristics are optimized for each of the individual engines. The maximum performance is then obtained subject to the operational restraints imposed by such factors as take-off field length, noise, approach, and take-off velocity. The techniques used in the present paper to obtain this objective are discussed below. In all cases, the maximum range achieved for a given take-off gross weight was calculated for a simple 8° C hot day; that is, the temperature at any standard day altitude is increased by 8° C and the speed of sound is calculated for the increased temperature. All other state variables are assumed to be the same as for a standard day. To avoid stagnation temperatures in excess of that of standard day flight Mach number of 2.7, the maximum flight Mach number for the hot day assumption is limited to a value of 2.62.

The figure of merit used to compare the performance of the various engine cycles is the maximum relative range achieved for a fixed take-off gross weight and payload. The relative range is defined as the range for any given engine normalized by that of the reference engine (RE). It was adopted as a means of emphasizing the effects of engines cycle upon performance and to avoid comparison with other studies in terms of absolute range which necessarily is dependent upon the level of structural and other technologies assumed.

3.1 Airplane

The airplane configuration chosen to "fly" with the candidate engines is shown in Figure 2. It is designed for a cruise Mach number of 2.7 standard day. For maximum aerodynamic efficiency, it incorporates an arrow-wing planform mounting four engines in separate pods aft beneath the wing for favorable interference effects. It has been sized to have a design 8° C hot day range of 7348 km (3968 n. mi.) carrying 292 passengers and equipped with an advanced single-spool nonafterburning turbojet engine. It meets the design range with a take-off gross weight of 325679 kg (718000 lbm) which is the value assumed throughout this paper.

The airplane characteristics for a wing loading of 352 kg/m^2 (72 lbm/ft^2) are fully described in Reference 9. As a result of recent wind-tunnel tests, more efficient flaps were developed which indicated that the use of a wing loading of 415 kg/m^2 (85 lbm/ft^2) would meet the take-off field length criteria and result in improved range. Therefore, the wing area was reduced to yield this value while maintaining the same aspect ratio. The aerodynamic characteristics were recalculated and the airplane rebalanced. The resulting baseline airplane maximum lift-drag ratio as a function of Mach number is shown in Figure 3. The airplane drag includes that due to both nacelle interference and nacelle skin friction. All other propulsion system drag items are included in the installed engine performance.

The baseline operating weight empty less that of the propulsion system is 33.8% of take-off gross weight.

3.2 Mission Profile

The mission profile flown for each engine is illustrated in Figure 4. Fuel reserve allowance from FAR 121.648 SST Fuel Requirement (tentative standard proposed by FAA) was modified for a change in holding altitude from 457 m (1500 ft) to 4572 m (15000 ft). The cruise portion of the mission was assumed to be entirely supersonic for the baseline mission. However, the necessity of avoiding sonic boom over populated areas may require a portion of the flight to be conducted at subsonic cruise speeds. This requirement makes the development of a variable-cycle engine especially attractive for supersonic transports. Therefore, two alternate mission profiles were examined which incorporated a 1111-km (600-n.-mi.) subsonic cruise range at either the departure or arrival portion of the flight. The subsonic cruise leg is assumed to be at a Mach number of 0.9 at best cruise altitude. The Mach number-altitude climb schedule for the all-supersonic cruise mission is shown in Figure 5. This schedule has been used in previous studies and has been checked to insure that it did not cross any flutter boundaries. Full climb thrust was employed during the accelerating climb without any attempt to optimize for any given engine cycle.

3.3 Engine-Airframe Matching

The relationship between engine size and wing area for maximum range can best be determined through the use of the so-called "thumbprint" or "knothole" diagram. Such a diagram is illustrated in Figure 6 for the RVE engine. Contours of constant relative range are shown as a function of installed thrust loading and wing loading. The contours were developed with the aid of the computer program described in Reference 10 which generates performance for a matrix of input wing loading and thrust loading values. Engine weight and dimensions are scaled in accordance with information provided with the engine performance decks. The airplane operating weight empty is adjusted for wing loading changes by assuming a constant fuselage and empennage weight and adjusting wing weight as a function of wing loading and engine weight in accord with previously determined parametric scaling laws. The airplane aerodynamic polar diagrams are adjusted for the effects of wing area changes and for the effects of both altitude and nacelle size on skin-friction drag.

Superimposed on the thumbprint diagram are limit lines which represent physical or operational restraints. Areas on the shaded side of each line represent portions of the diagram that violate the constraint. The balanced take-off field length, excess thrust, approach and take-off speed limits lines are assigned based on operational consideration at the values shown.

The maximum range of the reference variable-cycle engine (RE) used to normalize all other engine range performance values was determined for these same operating limits. For the case illustrated in Figure 6, the maximum relative range for an all-supersonic cruise mission without noise restraint is limited at the intersection of the take-off field length and transonic and supersonic excess thrust limit lines. Only a small sector of the knothole diagram bounded by the approach speed, take-off field length, and supersonic excess cruise thrust meet all operational restraints for airplane equipped with the RVE. For all engines, the maximum unrestrained range at the eye of the "knothole" is indicated.

4.0 ENVIRONMENTAL AND ECONOMIC FACTORS

The ranking of engine cycles in the SCAR program included projections of engine cost, maintainability, complexity, as well as performance in order to determine the most economically attractive cycle. The economic factors are ignored in this paper because they represent an area of greatest uncertainty. An engine cycle with a clear performance superiority should be economically competitive.

4.1 Emissions

The impact of engine emissions upon the design of combustors or duct burners cannot be assessed until such time as emission regulations are set forth; however, the goal of achieving low emissions both in flight and in the vicinity of the airport is of paramount importance. The development of low emission combustors is a problem shared to the same degree by all candidate engines. Projections based upon recent research indicate the possibility that low emission combustors can be developed within the volume and length of current practice. Therefore, the effect on performance of designing low emission combustors for the candidate variable-cycle engine is ignored in this paper.

The development of a low emission duct burner or afterburner presents a more difficult problem particularly with regard to hydrocarbon levels. Burner efficiencies very much higher than those achieved to date are required without sacrificing low-pressure loss performance. In addition, the current on-going research in low emission combustors for today's commercial engines is not directly applicable to burners because the velocity, pressure, and temperature levels are not comparable. However, it is assumed that the optimism expressed by the engine manufacturers is justified and that timely research and development will yield a low emission duct burner or afterburner with no engine performance or weight penalty.

4.2 Noise

The environmental factor that has the greatest impact upon engine size is the sideline and/or flyover noise level. To illustrate the effect of noise restraints, the maximum range for each engine has been determined first by means of the "thumbprint" diagram for maximum performance as previously described without consideration of noise. Secondly, the maximum range with noise restraints applied was determined from the "thumbprint" diagram for an engine sized to meet the maximum allowable noise level of 108 EPNdB at either the sideline or flyover measuring point. In this exercise, the variable-cycle engines were sized to meet the noise restraint both with and without suppression due to annular/coannular noise relief. For these variable-cycle engines, no consideration was given to any additional relief made possible through the use of acoustically treated liners or mechanical stream-immersed suppressors in an effort to demonstrate the potential benefits due solely to the annular/coannular effect. The suppression level assumed was provided by the engine manufacturers and was based upon small-scale static acoustic tests. The noise relief varied with throttle setting reaching a value at maximum throttle of 10 EPNdB for the VSCE, 5 EPNdB for the RVE, and 9 EPNdB for the DBE. The GE4, which is used to represent first-generation SST technology, was presumed to be equipped with an 8 EPNdB mechanical suppressor that weighed 7% of bare engine weight and created a 5% net thrust loss at take-off. This approach was taken presuming the annular noise relief effect was unrecognized at the time of the planned entry into service. The reference variable-cycle engine was assumed to be operated such that no suppression was required and thus incurred no range penalty due to noise.

5.0 PROPULSION SYSTEM PERFORMANCE

Engine performance data supplied by General Electric and Pratt & Whitney for their engine designs provided the net internal thrust and specific fuel consumption (uninstalled performance) at key altitudes, Mach numbers, and power settings. These data reflect only the effects of inlet pressure recovery, nozzle gross thrust coefficient, horsepower extraction, and engine bleed. As noted previously, all propulsion system related drag with the exception of nacelle friction and interference drag are charged to the engine. The drag due to inlet spillage, boundary-layer bleed, bypass, and boattail as functions of power settings were treated as thrust decrements in generating installed engine performance data decks for each engine. The isolated boattail drag as a function of power setting was provided by the engine manufacturers.

Each of the variable-cycle engines was presumed to operate in conjunction with the inlet described in References 11 and 12. This inlet is of mixed compression axisymmetric type designed for a Mach number of 2.65 and incorporates a translating center body and bleed ports on both cowl and center body to minimize shock-boundary-layer interaction. It is operated in an unstarted (external compression) mode up to a Mach number of 1.6 at which point the shock is swallowed. This inlet is sized to pass 2% greater airflow than that required by the engine and bleed system at supersonic cruise on a standard day. The total pressure recovery, bleed flow requirements, and maximum flow schedule as functions of Mach number are presented in Figure 7.

The GE4 is presumed to operate in conjunction with the Boeing-developed inlet whose performance and drag buildup is essentially as given in Reference 2. The reference variable-cycle engine (RE) is presumed to be operating with a rubberized infinitely variable inlet such that the only chargeable drag is that due to the boundary-layer bleed flow.

The performance for each of the considered engines at the subsonic and supersonic cruise Mach numbers at altitudes above 11 km (36000 ft) are demonstrated in Figure 8. Both installed and uninstalled data are plotted to indicate the effect of installation drags at these conditions. The net thrust has been non-dimensionalized with respect to each engine's maximum thrust at the given altitude and Mach number to define a thrust ratio. This technique was used to eliminate engine sizing effects.

The installation penalty at the supersonic cruise Mach number is entirely due to bleed drag except for the DBE which has a boattail drag approximately equal to a third of the bleed drag caused by a rearward facing faired step just upstream of the translating shroud. Boattail drag accounts for approximately 70% of the installation penalty for all engines at subsonic cruise Mach number. At this flight condition, however, the DBE with its translating shroud-plug nozzle and high airflow at part power exhibits about half the installation penalty as compared with the VSCE.

At the supersonic cruise Mach number, the minimum installed specific fuel consumption of all engines shown are quite comparable with a maximum difference of approximately 4%. At the subsonic cruise Mach number, the spread of the minimum fuel consumption values increases to approximately 50%, with the highest bypass ratio engine, RVE, exhibiting the lowest value and the turbojet engine, GE4, the highest value. The true ranking of these engines, in terms of fuel economy, depends upon the thrust ratio required to balance drag and engine size required to meet the operational restraints and not upon the indicated minimum value of SFC.

6.0 RESULTS AND DISCUSSION

Comparison of the performance of the various engine cycles in terms of maximum relative range are presented in Figures 9 and 10 to show directly the effects of cycle, noise, and subsonic cruise requirement on range. The incremental range, fuel usages, and propulsion system performance for each of the maximum range configurations are given in Tables II and III. All engines and airframes were sized to meet the operational limitations imposed by take-off field length, approach velocity, and excess thrust; however, the limitations imposed by the noise criteria were treated separately. To show the effect of noise, the engines were first sized for maximum performance without consideration of take-off noise level (no noise restraints). The engines were then resized and matched to the airframe to determine the maximum range with a noise limitation of 108 EPNdB without any noise relief due to mechanical or annular/coannular suppression (108 EPNdB, no suppression). Finally, the effect of the assumed suppression levels were included and the engines resized, where necessary, to determine maximum range (108 EPNdB with suppression).

The variable-cycle engines, as a group, exhibit substantial range improvements as compared to the GE4 (Fig. 9). On the basis of maximum performance alone without consideration of the effects of noise, a range improvement of 25% is provided by the RVE. This engine has a range of 76% of the ultimate as represented by the RE. When compared on the basis of engines sized to meet the 108 EPNdB noise restriction, including the effects of noise suppression, the improvement in range of the RVE as compared to the GE4 increases to 40%. This is accomplished with only a 1% penalty in range for the RVE sized to satisfy the noise restrictions. The GE4 range for these conditions is only 53% of that for the RE. It should be remembered that the variable-cycle engines were designed with the FAA noise regulations in mind, whereas the GE4 was selected for the U.S. SST with much less restrictive noise requirements. Therefore, the imposition of the 108 EPNdB limit upon a high specific thrust cycle (high jet velocity) such as the GE4 results in the need for a large oversized engine and a consequent significant range penalty. The variable-cycle engines, on the other hand, exhibit relatively small decreases in range when sized to meet the noise standards even without the benefit of suppression. It is interesting to note that the range of all the variable-cycle engines sized to meet the noise restrictions without the effects of suppression exceed that of the GE4 sized for maximum performance without noise considerations. An increase in the level of mechanical suppression beyond the 8 EPNdB for the GE4 will be of no help because the airplane with the resulting smaller engine will be incapable of take-off within the full runway length of 3200 m (10500 ft).

A 1111-km (600-n.-mi.) subsonic cruise to avoid sonic boom at either the departure or arrival portion of the flight for engines sized to satisfy the noise restrictions with the assumed suppression levels included (Fig. 10) indicate relatively small reductions of 3 to 6% in range for the variable-cycle engines.

The GE4 sustains a 19% loss in range under the same conditions, primarily as a result of the poor cycle efficiency for heavily throttled low-pressure ratio turbojets at subsonic speeds and the large installation penalty due to high spillage and boattail drag.

Insight to the factors which have a bearing upon the overall performance noted above may be obtained from Figures 11 to 13 as well as from Table II. The DBE uses the greatest amount of fuel and travels farthest during the climb to supersonic cruise for the case of engines selected with no noise restrictions (Fig. 11). As can be seen in Figure 12, the range of the DBE is limited by the excess thrust available in climb. An increase in the afterburner operating temperature could increase its range because the engine could then be sized to meet the take-off field length and have more fuel onboard for the more efficient supersonic cruise leg. The RVE, which also exhibits a relatively long climb distance, is limited in engine size by both field length and climb thrust. It cannot climb more rapidly to the supersonic cruise point because the presence of the rear valve sets the limit on auxiliary burner temperature. It is interesting to note that the VSCE which is both slightly lighter and exhibits lower specific fuel consumption at both supersonic and subsonic cruise points has 3% less range than the RVE. This is a result of accelerating in the less efficient full augmented power mode at all speeds. Although no attempt was made to optimize the climb throttle schedule, the range of the VSCE improved 1% by restricting duct burning during climb to Mach numbers above 0.8.

Both the RVE and DBE engines, as used in this study, are sized by the requirement to meet the noise limit of 108 EPNdB in the suppressed mode for the given runway length. Increasing the take-off field length limit for either of these engines would result in an insignificant range improvement because the climb thrust requirement would then become the engine sizing parameter. The climb thrust limit is not an absolute operational requirement; however, any reduction in the assumed minimum thrust-to-drag ratio value of 1.2 will adversely affect acceleration capability to start of cruise and increase trip time. The VSCE, on the other hand, is limited only by take-off field length and not by noise. The sideline noise which is the dominant factor for this engine is actually less than 108 EPNdB and an increase in take-off field length would permit the use of a smaller engine and a consequent improvement in range. Relaxation of the take-off field length limit would, therefore, favor the VSCE such that its range would equal or slightly exceed that of the RVE.

Within the accuracy of defining the performance of the engines based upon projections of future technology development, the RVE and VSCE can be viewed as having equal potential for use in a second-generation supersonic cruise airplane having a standard day cruise Mach number of 2.7. The DBE, on the other hand, although superior to the GE4 in range, has a range 13% less than that of the RVE when compared for the noise limited suppressed all-supersonic flight condition. The supersonic cruise specific fuel consumption and fuel reserves of the two engines are not too dissimilar (Table II). As noted earlier, an increase in afterburner temperature limits would only slightly improve the range of the DBE. Therefore, the prime reason for the difference in range is that the installed weight of the DBE propulsion system is 19% greater than that of the RVE and, as a matter of fact, is equal to that of the GE4.

The VSCE and the RVE exhibit approximately equal range potential yet represent widely divergent variable-cycle concepts. The VSCE is essentially a turbofan engine with controllable primary and secondary nozzle-throat areas which can be scheduled to provide engine-inlet flow match at maximum nonaugmented power over most of the flight spectrum. The RVE, on the other hand, employs a unique flow path schedule which provides a cycle change from what is essentially a relatively high bypass turbofan to a dual turbojet. Unfortunately, it does not exhibit fully the favorable fuel economy of the conventional turbofan at subsonic cruise nor that of a conventional turbojet at the supersonic cruise Mach number. This is a result of the compromise required in the selection of fan pressure ratio. In the turbojet mode, the overall cycle pressure ratio for the bypassed flow turbojet is equal to the fan pressure ratio and is too low for best fuel economy. In the turbofan mode, the fan pressure ratio is too high and the overpressurization and consequent expansion through the rear turbine reduce the thrust potential of the bypass stream due to the additional rise in entropy through the fan and turbine. In addition, the RVE cannot generate sufficient thrust for take-off in the turbofan mode which is desired for reasons of noise, because its air handling capacity is not increased to compensate for its low specific thrust. Thus, for these reasons, the RVE does not meet the full objectives of a variable-cycle engine set forth by Nichols (Ref. 1).

The DBE is the only one of the three considered variable-cycle engines which maintains full engine airflow at the subsonic cruise power setting. The other two variable-cycle engines must spill or bypass from 12 to 21% of full throttle airflow. The fixed airflow of the DBE in combination with the translating shroud nozzle, provides throttle independent bypass, spillage, and boattail drag down to one-half of the maximum nonaugmented thrust level. The elimination of throttle dependent drag is one of the goals for variable-cycle engines advocated by Swan (Ref. 2). However, the specific design of the translating shroud incurs a basic boattail drag which, if it were possible to eliminate by redesign, would yield subsonic as well as supersonic cruise specific fuel consumption rates equal to the VSCE. Although the range of the DBE would increase as a result of eliminating the boattail drag, it would not equal that of the VSCE or RVE because of its higher propulsion system weight fraction.

It can be concluded that the range increase noted for all engines, as compared to the GE4, may not be due as much to meeting the goals outlined for variable-cycle engines as that due to advanced technology represented by higher turbine temperature, improved component performance, annular/coannular noise relief, structural efficiency, and to the selection of cycle parameters and methods of operation to best match the conflicting performance requirements.

Comparison of the relative range of the three candidate variable-cycle engines to that of the reference engine indicate that further significant improvements are possible. However, these improvements in potential fuel savings are contingent upon finding a means of varying cycle parameters for each flight condition which are not too costly in terms of weight. For example, if it were possible to achieve the low fuel consumption rates of the reference engine, it would be pointless to do so if the installed propulsion system weight were equal to or greater than 173% of the longest range variable-cycle engine, for then there would be no gain in range.

It should be emphasized that the SCAR engine program was undertaken with the goal of determining the potential gains possible for supersonic cruise aircraft equipped with advanced technology engines and if found attractive to foster research on those components critical to achieving that goal. As a group, the variable-cycle engines were found to be superior to the more conventional cycles. However, the relative ranking of the three variable cycles considered in this paper does not necessarily reflect the desirability of choosing the highest ranking cycle for development at this time for several reasons. First, the flight Mach number of 2.7 chosen for the purpose of comparison with the GE4 may not be the most desirable flight speed. The three airplane companies have performed independent studies which indicate the most economically desirable flight Mach number varied, depending upon company, from a low of 2.2 to a high of 2.55. Second, the relaxation of the restrictive take-off field length would favor the VSCE which is the only cycle whose sizing is strictly limited in range by take-off field length and not by noise. Last, the rankings ignore the complexity, maintainability, and cost factors and are dependent solely upon achieving the flexibility, performance, and weight assumed for each engine. A case in point is the low ranking of the DBE due in large measure to its relatively heavy weight. Whether this is caused by a conservative weight extrapolation philosophy by one manufacturer or to an optimistic estimate for the other two engines by the second manufacturer or whether either or both estimates are valid cannot be determined until a detailed part-by-part engine design is completed. Before initiating any such costly detail design, the basis for the assumed component performance must be examined. For conventional components such as fans, compressors, combustors, and turbines, the historical background and on-going research and development programs applicable to all types of engines provide a firm base from which to project performance and weight estimates. However, it is necessary to verify the performance of new and advanced technology items which affect overall engine performance or weight. Some of the more critical items in this category are reviewed below.

A new technology item that has an important influence on the selection of engine size and thus range is the jet noise relief due to the coannular/annular effect. Although this effect was noted many years ago, its potential benefits were not recognized until recently. Small-scale static tests have established the suppression level over a range of bypass and velocity ratios for the coannular nozzle and radius ratios for the annular nozzle and have been used as a base for estimating the relief for the subject engines. The effects of forward velocity, size, and internal stream mixing upon noise suppression levels are as yet not well known. Therefore, NASA has established a phased experimental program to determine their influence upon noise suppression and to provide a firmer base for future noise prediction studies.

The development of a highly efficient secondary burner is a particularly critical item from the standpoint of meeting the anticipated hydrocarbon emission standard as well as its effect upon fuel consumption. Of all the variable-cycle engines, the performance of the VSCE is most vulnerable to duct burner design changes that may be needed to meet the combustion efficiency levels assumed. The higher pressure and temperatures associated with the secondary burners of the other two engine cycles makes the problem of attaining high efficiency only slightly less difficult. To provide insight and guidance, NASA's Experimental Clean Combustor Program was enlarged to include the study of duct burner concepts leading to high efficiency and low emissions.

The variable-cycle engines employ scheduled stator angle settings in both the fan and compressor elements which in combination with spool speed and exhaust nozzle throat area variation is used to essentially match the inlet airflow schedule at maximum turbine inlet temperature. In addition, the stator angles are scheduled to maintain the engine operating line on the fan and compressor maps near regions of best efficiency. For the DBE, as the cycle changes from double to single bypass operation, the compressor must accept approximately 25% greater airflow. This produces a difficult design problem of maintaining good efficiency and sufficient stall margin over a wide pumping range. The assumed performance of the DBE depends to a greater degree upon the resolution of this design problem than do the other cycles. Support for a research program in this area has been funded by NASA.

The use of airflow path control valves for cycle flexibility is uniquely identified with the RVE and DBE cycles. Estimates of valve pressure loss for these cycles were based upon model tests, however, the trade off of component performance and weight to achieve maximum overall system performance need more refined design and test data. In addition, the losses associated with mixing streams of differing energy levels, the effect of leakage, and, for the RVE, the effect on turbine efficiency of a periodic circumferential temperature variation need to be determined to validate the engine performance estimates.

The SCAR engine studies have resulted in unusual design and control concepts advocated by the engine manufacturers and verified by the airplane company's systems studies. Although they show significant range improvements as compared to the GE4, further improvements may be possible if the propulsion system concept is treated as an entity. To this end, future studies will involve the cooperative effort of the engine and airplane manufacturers to identify means of modifying the inlet, engine cycle, and nozzle by trading component performance and weight to either maximize range or minimize take-off gross weight.

7.0 CONCLUSIONS

An examination of the range potential of three candidate variable-cycle engines proposed for a second-generation supersonic cruise transport was undertaken to determine the possible improvements in performance as compared to that obtainable with 1969 technology represented by the U.S. SST GE4 engine. The three variable-cycle engines are descriptively designated as the Variable Stream Control Engine, the Rear Valve Engine, and the Double Bypass Engine. Comparison of the range performance of these engines was made with that of a fantasized completely variable propulsion system used to represent the potential upper bound of such engines. In addition, critical areas requiring new or improved technology for each of the subject variable-cycle engines are noted.

The aircraft configuration chosen for the study had an arrow-wing planform with four engines mounted in separate pods beneath the wing. The take-off gross weight and payload were fixed and the engine size and wing area were varied to achieve maximum range within certain operational restraints. The primary mission was a Mach number 2.62 hot day all-supersonic cruise; however, the effects of a 1111-km (600-n.-mi.)

subsonic cruise element at either the departure or arrival portion of the flight was considered. To determine the effects of noise regulations upon range, the maximum range was calculated for engines sized; first without any noise restraints, then to satisfy the noise criteria but without the use of any form of suppression, and, last, to satisfy the noise regulations using annular/coannular noise relief or mechanical suppression.

The best variable-cycle engine had about 25% greater range than the GE4 when no noise restrictions were imposed and about 40% greater range than the GE4 for engines sized to meet the noise criteria with suppression.

For engines sized to meet the noise limit with suppression, the best variable-cycle engine exhibited a range of 75% of that of the fantasized completely variable propulsion system while the GE4 had 53% of this range.

The subsonic cruise requirement reduced the range of the variable-cycle engines from 3 to 6% but reduced the range of the GE4 by approximately 19%.

The range of the candidate variable-cycle engine is reduced only 2 to 6% by the necessity to resize the engines in the event of failure to achieve any of the noise suppression assumed. For the same condition, a resized GE4 will lose 24% of its range.

Although major potential improvements in range appear possible with increased propulsion system variability, it cannot be realized if the system weight must increase significantly to achieve the flexibility required.

The performance of any of the variable-cycle engines depends critically upon the attainment of the predicted technology levels used in these studies. To provide the needed technology, programs are required to develop clean duct/afterburners with efficiencies very much better than present values; to develop light, low loss valving to divert and mix streams of differing energy levels; to develop rotating engine components which provide the flexibility to accept wide variations in flow with acceptable efficiency and surge margins; and, finally, to verify and understand the mechanism leading to the noise relief associated with the annular/coannular jet effect.

8.0 REFERENCES

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TABLE I. ENGINE SPECIFICATIONS
(UNINSTALLED 8° C HOT DAY)

	VSCE		RVE		DBE		GE4 (Standard day)	
Mass flow rate, kg/sec (lbm/sec)	408.2	(900)	408.2	(900)	435.4/362.9	(960/800)	287.1	(633)
Engine weight, kg (lbm)	6168.4	(13600)	6230.9	(13870)	7279.6	(16050)	6006.5	(13243)
Bypass ratio	1.3		2.5		0.35		0.0	
Fan pressure ratio	3.3		5.8		2.7/4.0		----	
Overall engine pressure ratio	16:1		21:1		17.3:1		12.5:1	
Max turbine inlet temperature, °K (°F)	1811	(2800)	1811	(2800)	1811	(2800)	1522	(2280)
Max secondary burner temperature, °K (°F)	1644	(2500)	1311	(1900)	1311	(1900)	1944	(3040)
Take-off max thrust, N (lbf)	286656	(64443)	287742	(64687)	299365	(67300)	284686	(64000)
Take-off SFC, kg/hr/N (lbm/hr/lbf)	.1482	(1.4546)	.11051	(1.0838)	.1224	(1.20)	.1896	(1.86)
Subsonic cruise								
Flight Mach number	0.9		0.9		0.9		0.9	
Flight altitude, m (ft)	10999	(36089)	10999	(36089)	10668	(35000)	11018	(36150)
Max net thrust, N (lbf)	111303	(25022)	100489	(22591)	84030	(18891)	108553	(24404)
SFC at max net thrust, kg/hr/N (lbm/hr/lbf)	.15719	(1.5416)	.1256	(1.2319) (TJ)	.1469	(1.441)	.187	(1.851)
Bypass ratio at max net thrust	1.2077		3.1692		0.36		0.0	
Net thrust at min SFC, N (lbf)	48899	(10993)	50567	(11368)	52155	(11725)	33930	(7290)
Min SFC, kg/hr/N (lbm/hr/lbf)	.0885	(.8681)	.08773	(.86039) (TF)	.11117	(1.003)	.108	(1.078)
Supersonic cruise								
Flight Mach number	2.62		2.62		2.62		2.62	
Flight altitude, m (ft)	19812	(65000)	19812	(65000)	19812	(65000)	19812	(65000)
Max net thrust, N (lbf)	128771	(28949)	79858	(17953)	74920	(16843)	99195	(22300)
SFC at max net thrust, kg/hr/N (lbm/hr/lbf)	.1793	(1.7584)	.15185	(1.4891)	.1588	(1.557)	.2008	(1.97)
Bypass ratio	1.5485		3.9927		0.74		0.0	
Net thrust at min SFC, N (lbf)	57124	(12842)	62221	(13988)	44810	(10074)	47151	(10600)
Min SFC, kg/hr/N (lbm/hr/lbf)	.1422	(1.3947)	.15020	(1.4730)	.1435	(1.408)	.1468	(1.44)

TABLE II. MISSION ELEMENTS — ALL SUPERSONIC

(a) No Noise Restraints

	GE4		VSCE		RVE		DBE		RE	
Aircraft description										
Nominal engine airflow, kg/sec (lbm/sec)	266.7	(588)	371.9	(820)	368.3	(812)	347.7	(768)	261.3	(576)
Propulsion system weight	.1014		.1026		.1035		.1193		.0998	
Thrust loading, N/kg (lbf/lbm)	3.24	(.33)	3.21	(.327)	3.19	(.325)	3.53	(.360)	3.18	(.324)
Wing loading, kg/m^2 (lbm/ft^2)	415.0	(85)	415.0	(85)	402.8	(82.5)	422.4	(86.5)	400.4	(82)
Operating weight empty	.443		.444		.448		.457		.447	
Payload weight	.085		.085		.085		.085		.085	
Mission description										
Weight at end of take-off	.982		.984		.986		.985		.991	
Climb Δ Fuel weight	.137		.105		.118		.143		.073	
Δ Relative range	.0611		.0427		.0921		.1103		.0608	
Start of cruise, Weight	.863		.895		.882		.857		.927	
Altitude, m (ft)	17983	(59000)	18092	(59360)	18565	(60910)	17983	(59000)	18368	(60265)
Lift-drag ratio	8.96		9.09		9.16		8.96		9.20	
SFC, kg/hr/N (lbm/hr/lbf)	.1677	(1.645)	.1552	(1.522)	.1592	(1.561)	.1613	(1.582)	.1414	(1.387)
Thrust ratio	.594		.518		.904		.828		.669	
End of cruise, Weight	.649		.636		.632		.643		.602	
Relative range	.5488		.6708		.6958		.6115		.9397	
End of descent, Weight	.641		.627		.626		.637		.598	
Relative range	.6052		.7308		.7565		.6712		1.000	
Burned fuel weight	.359		.373		.374		.363		.402	
Total reserve fuel weight	.113		.098		.093		.095		.066	
Total fuel weight	.472		.471		.467		.458		.468	
Reserve										
In route fuel reserve weight	.025		.026		.026		.025		.028	
Missed approach fuel weight	.017		.013		.010		.012		.004	
Cruise to alternate fuel weight	.045		.038		.036		.037		.021	
SFC, kg/hr/N (lbm/hr/lbf)	.1353	(1.327)	.1192	(1.169)	.1209	(1.186)	.1142	(1.120)	.0754	(.739)
Mach number	.65		.70		.65		.70		.65	
Altitude, m (ft)	4724	(15500)	4724	(15500)	4724	(15500)	4724	(15500)	4724	(15500)
Thrust ratio	.212		.173		.192		.352		.200	
Hold fuel weight	.026		.021		.021		.021		.013	

Note: All weights expressed as fractions of aircraft take-off gross weight.

TABLE II. MISSION ELEMENTS — ALL SUPERSONIC (Continued)

(b) 108 EPNdB, no Suppression

	GE4		VSCE		RVE		DBE	
Aircraft description								
Nominal engine airflow, kg/sec (lbm/sec)	401.9	(886.1)	388.7	(856.9)	417.0	(919.4)	377.4	(832.0)
Propulsion system weight	.1646		.1077		.1187		.1359	
Thrust loading, N/kg (lbf/lbm)	4.87	(.497)	3.35	(.342)	3.61	(.368)	3.83	(.390)
Wing loading, kg/m^2 (lbm/ft^2)	361.3	(74.0)	415.0	(85.0)	402.8	(82.5)	410.1	(84.0)
Operating weight empty	.525		.449		.463		.478	
Payload weight	.085		.085		.085		.085	
Mission description								
Weight at end of take-off	.981		.984		.986		.986	
Climb Δ Fuel weight	.108		.103		.103		.128	
Δ Relative range	.0310		.0397		.0683		.0881	
Start of cruise, Weight	.892		.897		.897		.872	
Altitude, m (ft)	18843	(61820)	18091	(59355)	18851	(61850)	17983	(59000)
Lift-drag ratio	8.904		9.073		9.099		8.981	
SFC, kg/hr/N (lbm/hr/lbf)	.1560	(1.530)	.1548	(1.518)	.1594	(1.563)	.1593	(1.562)
Thrust ratio	.472		.498		.856		.776	
End of cruise, Weight	.753		.643		.652		.666	
Relative range	.3409		.6541		.6428		.5670	
End of descent, Weight	.743		.634		.645		.659	
Relative range	.3969		.7143		.7037		.6269	
Burned fuel weight	.257		.366		.355		.341	
Total reserve fuel weight	.133		.100		.097		.096	
Total fuel weight	.390		.466		.452		.437	
Reserve								
In route fuel reserve weight	.018		.026		.025		.024	
Missed approach fuel weight	.028		.014		.011		.013	
Cruise to alternate fuel weight	.055		.038		.038		.038	
SFC, kg/hr/N (lbm/hr/lbf)	.1448	(1.420)	.1208	(1.185)	.1231	(1.207)	.1142	(1.120)
Mach number	.65		.70		.75		.70	
Altitude, m (ft)	4724	(15500)	4724	(15500)	4724	(22000)	4724	(15500)
Thrust ratio	.165		.167		.218		.352	
Hold fuel weight	.032		.022		.022		.021	

TABLE II. MISSION ELEMENTS — ALL SUPERSONIC (Concluded)

(c) 108 EPNdB, With Suppression

	GE4		VSCE		RVE		DBE	
Aircraft description								
Nominal engine airflow, kg/sec (lbm/sec)	316.7	(698.2)	371.9	(820.0)	375.6	(828.0)	352.3	(776.7)
Propulsion system weight	.1289		.1026		.1058		.1254	
Thrust loading, N/kg (lbf/lbm)	3.84	(.392)	3.21	(.327)	3.25	(.331)	3.57	(.364)
Wing loading, kg/m^2 (lbm/ft^2)	395.5	(81.0)	415.0	(85)	402.8	(82.5)	422.3	(86.5)
Operating weight empty	.477		.444		.450		.464	
Payload weight	.085		.085		.085		.085	
Mission description								
Weight at end of take-off	.982		.984		.986		.986	
Climb Δ Fuel weight	.120		.105		.115		.140	
Δ Relative range	.0444		.0427		.0869		.1066	
Start of cruise, Weight	.879		.895		.885		.860	
Altitude, m (ft)	17983	(59000)	18092	(59360)	18550	(60860)	17983	(59000)
Lift-drag ratio	8.816		9.09		9.15		8.96	
SFC, kg/hr/N (lbm/hr/lbf)	.1562	(1.532)	.1552	(1.522)	.1591	(1.5605)	.1610	(1.579)
Thrust ratio	.517		.518		.888		.821	
End of cruise, Weight	692		.636		.635		.650	
Relative range	.4685		.6708		.6882		.5950	
End of descent, Weight	.683		.627		.629		.644	
Relative range	.5243		.7308		.7488		.6547	
Burned fuel weight	.317		.373		.371		.356	
Total reserve fuel weight	.121		.098		.094		.095	
Total fuel weight	.438		.471		.465		.451	
Reserve								
In route fuel reserve weight	.022		.026		.026		.025	
Missed approach fuel weight	.021		.013		.010		.012	
Cruise to alternate fuel weight	.049		.038		.037		.037	
SFC, kg/hr/N (lbm/hr/lbf)	.1392	(1.365)	.1192	(1.1688)	.1215	(1.1919)	.1142	(1.120)
Mach number	.65		.70		.65		.70	
Altitude, m (ft)	4724	(15500)	4724	(15500)	4724	(15500)	4724	(15500)
Thrust ratio	.192		.173		.189		.352	
Hold fuel weight	.029		.021		.021		.021	

TABLE III. MISSION ELEMENTS
600 NAUTICAL MILE SUBSONIC CRUISE — 108 EPNdB WITH SUPPRESSION

(a) Subsonic Cruise Departure

		GE4		VSCE		RVE		DBE	
Aircraft description									
Nominal engine airflow, kg/sec (lbm/sec)		316.7	(698.2)	371.9	(820.0)	375.6	(828.0)	352.3	(776.7)
Propulsion system weight		.1289		.1026		.1058		.1254	
Thrust loading, N/kg (lbf/lbm)		3.84	(.392)	3.21	(.327)	3.25	(.331)	3.57	(.364)
Wing loading, kg/m^2 (lbm/ft^2)		395.5	(81.0)	415.0	(85.0)	402.8	(82.5)	422.3	(86.5)
Operating weight empty		.477		.444		.450		.464	
Mission description									
Weight at end of take-off		.981		.984		.986		.986	
Start of subsonic cruise,	Weight	.954		.959		.965		.960	
	Altitude, m (ft)	6096	(20000)	6096	(20000)	6096	(20000)	6096	(20000)
	Lift-drag ratio	13.90		14.31		14.37		14.23	
	SFC, kg/hr/N (lbm/hr/lbf)	.1580	(1.549)	.1102	(1.081)	.1136	(1.114)	.1222	(1.198)
	Thrust ratio	.2443		.2750		.3000		.3650	
End of subsonic cruise,	Weight	.847		.884		.889		.878	
Start of supersonic cruise,	Weight	.783		.829		.821		.797	
	Altitude	18649	(61185)	18589	(60990)	19042	(62475)	18346	(60190)
	Lift-drag ratio	8.514		9.041		9.105		8.888	
	SFC, kg/hr/N (lbm/hr/lbf)	.1563	(1.532)	.1553	(1.523)	.1592	(1.561)	.1607	(1.576)
	Thrust ratio	.5328		.5224		.8965		.8143	
End of supersonic cruise,	Weight	.692		.636		.635		.650	
	Relative range	.3765		.6458		.6608		.5642	
End of descent,	Weight	.683		.627		.629		.644	
Relative range		.4323		.7061		.7214		.6237	
Total reserve fuel weight		.121		.098		.094		.095	
Total fuel weight		.438		.471		.465		.451	
Reserves									
In route fuel reserve weight		.022		.026		.026		.025	
Missed approached fuel weight		.021		.013		.010		.012	
Cruise to alterante fuel weight		.049		.038		.037		.037	
	SFC, kb/hr/N (lbm/hr/lbf)	.1392	(1.365)	.1192	(1.196)	.1215	(1.192)	.1142	(1.120)
	Mach number	.65		.70		.65		.70	
	Altitude, m (ft)	4724	(15500)	4724	(15500)	4724	(15500)	4724	(15500)
Hold fuel weight		.029		.021		.021		.021	

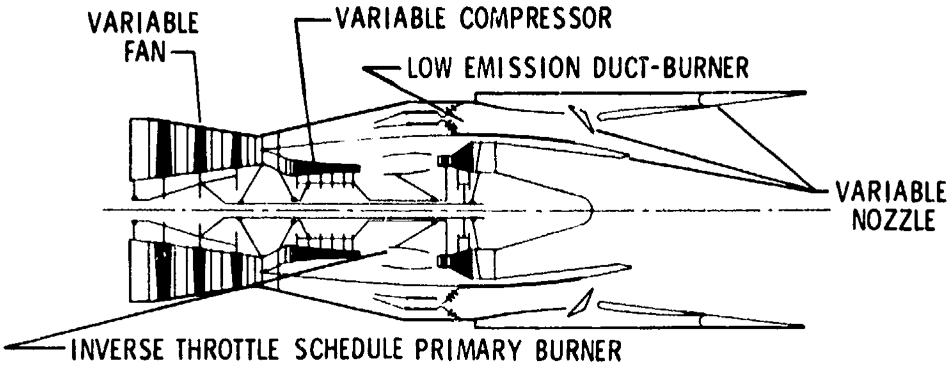
Note: All weights expressed as fractions of aircraft take-off gross weight.

TABLE III. MISSION ELEMENTS

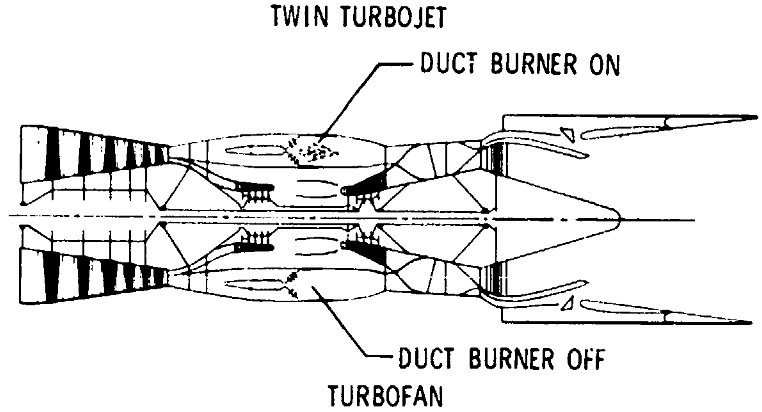
600 NAUTICAL MILE SUBSONIC CRUISE — 108 EPNdB WITH SUPPRESSION (Concluded)

(b) Subsonic Cruise Arrival

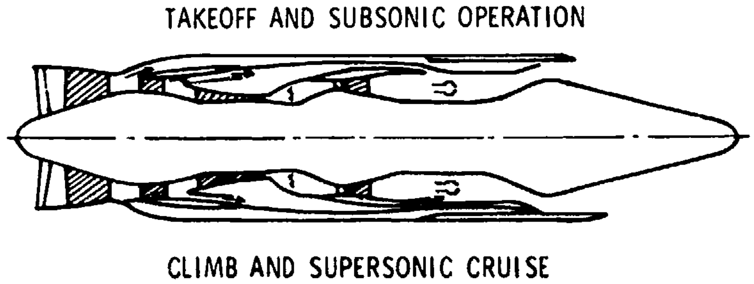
		GE4		VSCE		RVE		DBE	
Aircraft description									
Nominal engine airflow, kg/sec (lbm/sec)		317.0	(698.9)	371.9	(820.0)	375.1	(827.0)	352.2	(776.5)
Propulsion system weight		.1289		.1026		.1058		.1254	
Thrust loading, N/kg (lbf/lbm)		3.84	(.392)	3.21	(.327)	3.25	(.331)	3.57	(.364)
Wing loading, kg/m^2 (lbm/ft^2)		395.5	(81.0)	415.0	(85.0)	402.8	(82.5)	422.3	(86.5)
Operating weight empty		.477		.444		.450		.464	
Mission description									
Weight at end of take-off		.981		.984		.986		.986	
Start of supersonic cruise,	Weight	.879		.895		.885		.860	
	Altitude, m (ft)	17983	(59000)	18092	(59360)	18548	(60855)	17983	(59000)
	Lift-drag ratio	8.841		9.088		9.152		8.962	
	SFC, kg/hr/N (lbm/hr/lbf)	.1560	(1.530)	.1552	(1.522)	.1592	(1.561)	.1610	(1.579)
	Thrust ratio	.530		.518		.888		.821	
End of supersonic cruise,	Weight	.779		.691		.692		.714	
	Relative range	.2605		.5189		.5330		.4322	
Start of subsonic cruise,	Weight	.775		.688		.690		.712	
	Altitude, m (ft)	9144	(30000)	9144	(30000)	9144	(30000)	9144	(30000)
	Lift-drag ratio	13.980		14.16		14.24		14.06	
	SFC, kg/hr/N (lbm/hr/lbf)	.1511	(1.482)	.1070	(1.049)	.1095	(1.074)	.1178	(1.155)
	Thrust ratio	.297		.304		.332		.418	
End of subsonic cruise,	Weight	.689		.634		.634		.650	
End of descent,	Weight	.683		.627		.629		.644	
Relative range		.4381		.7009		.7240		.6137	
Total reserve fuel weight		.121		.098		.094		.095	
Total fuel weight		.438		.471		.465		.451	
Reserves									
In route fuel reserve weight		.020		.026		.026		.025	
Missed approach fuel weight		.023		.013		.010		.012	
Cruise to alternate fuel weight		.049		.038		.037		.037	
	SFC, kg/hr/N (lbm/hr/lbf)	.1393	(1.366)	.1192	(1.169)	.1215	(1.192)	.1142	(1.120)
	Mach number	.65		.70		.65		.70	
	Altitude, m (ft)	4724	(15500)	4724	(15500)	4724	(15500)	4724	(15500)
Hold fuel weight		.029		.021		.021		.021	



(a) Variable stream control engine (VSCE).



(b) Rear value engine (RVE).



(c) Double bypass engine (DBE).

Figure 1. Variable cycle engines.

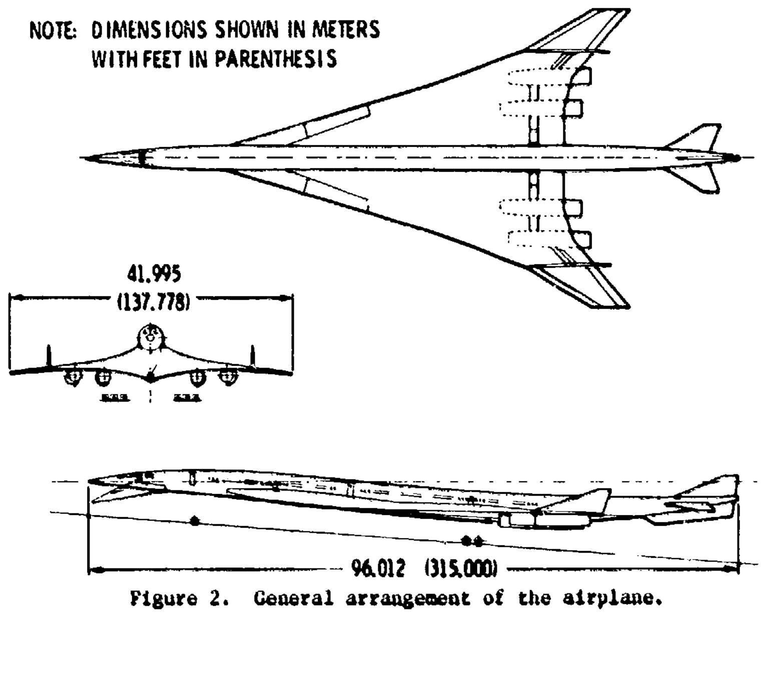


Figure 2. General arrangement of the airplane.

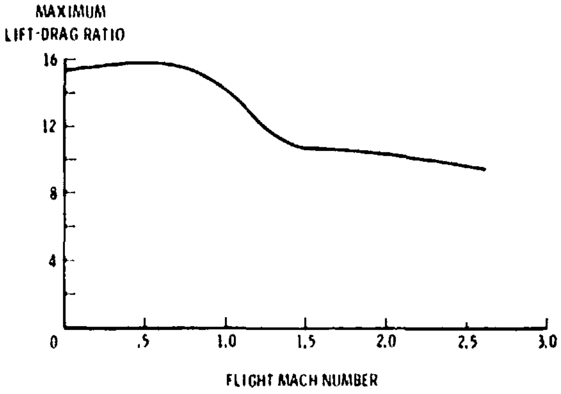


Figure 3. Maximum lift-drag ratio.

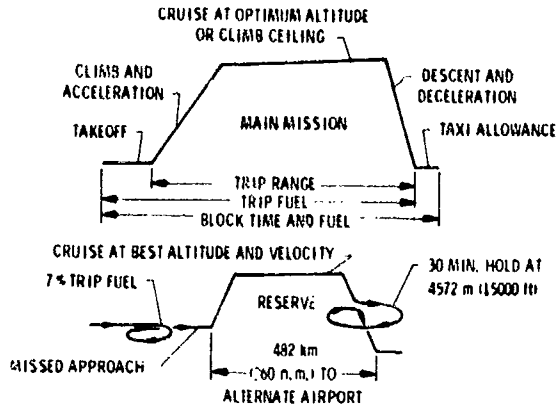


Figure 4. Mission profiles/reserves.

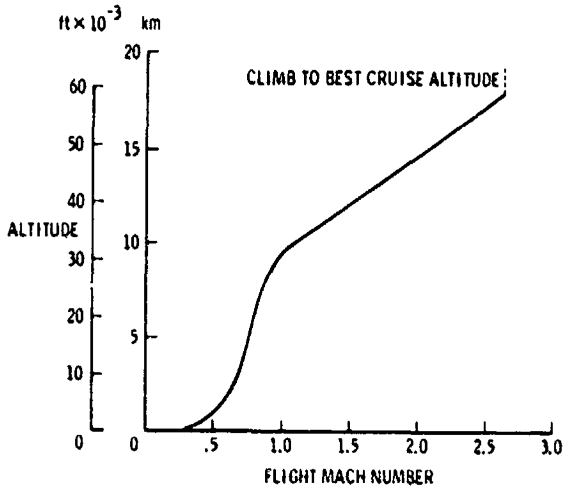


Figure 5. Climb schedule.

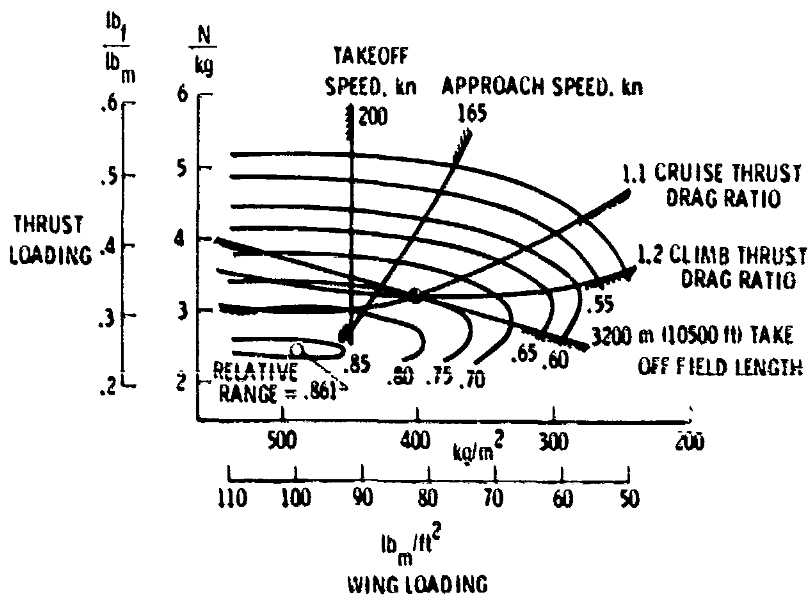
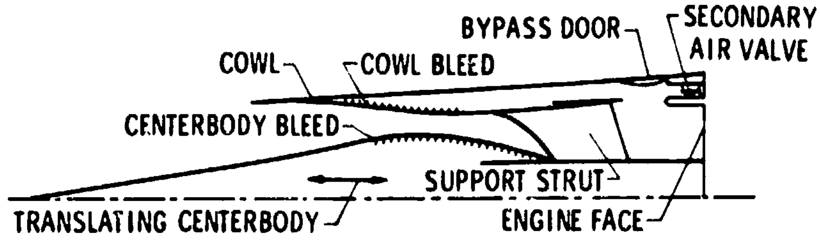
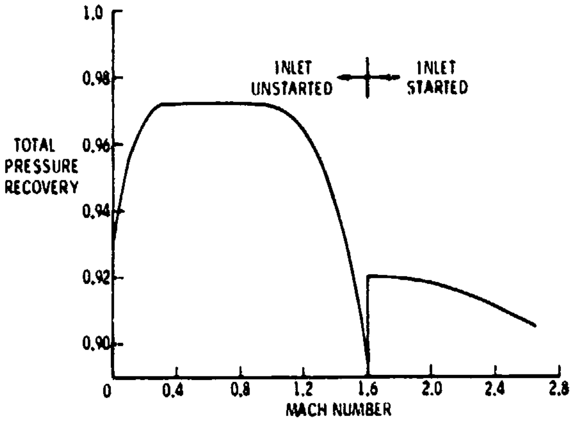
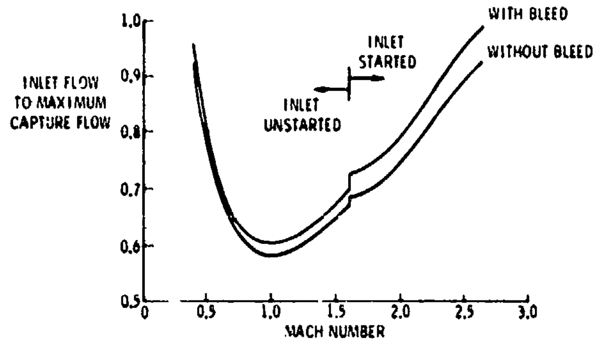


Figure 6. Engine airplane range diagram for the RVE.



(a) Inlet schematic.





(b) Inlet pressure recovery. (c) Inlet airflow schedule.

Figure 7. Inlet description.

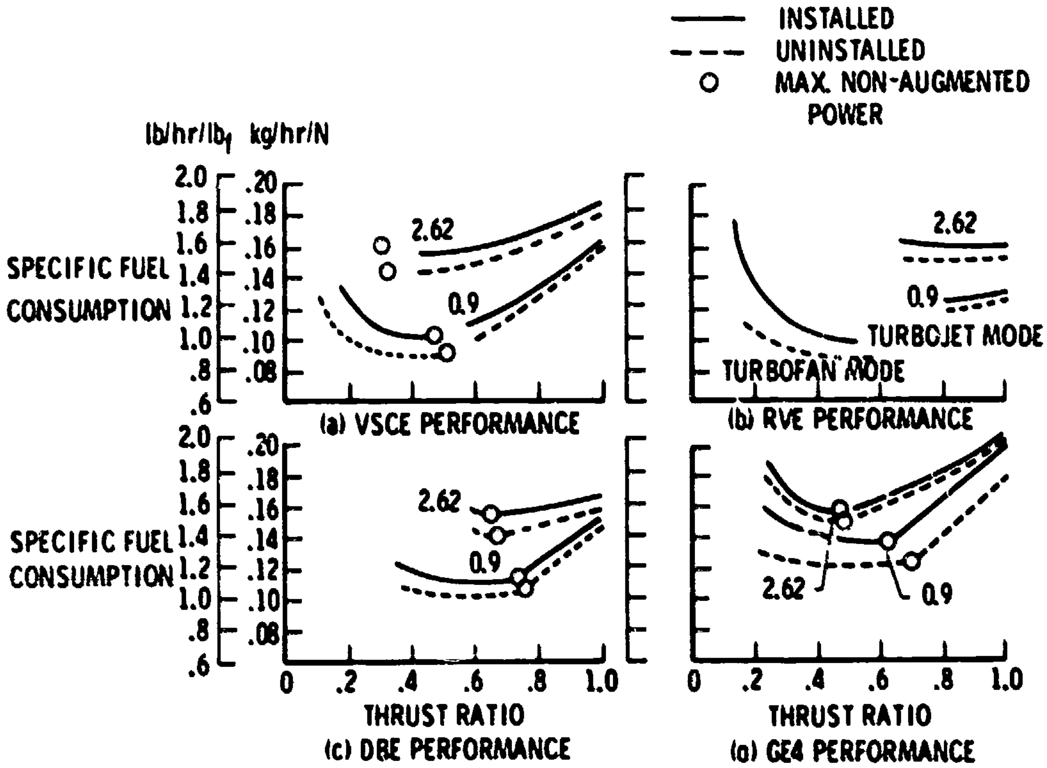


Figure 8. Installed and uninstalled engine performance.

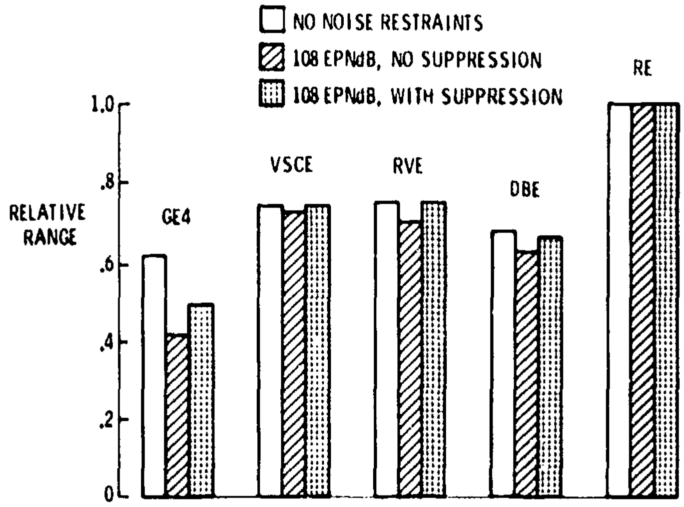


Figure 9. Maximum range comparison for all supersonic mission.

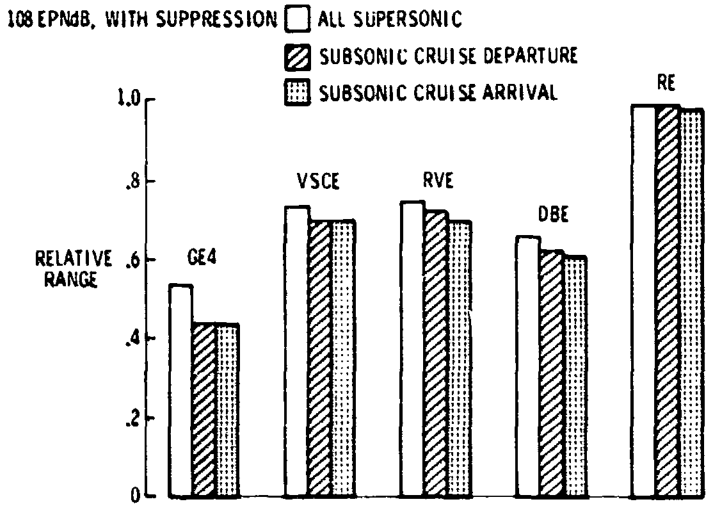


Figure 10. Effect of subsonic element.

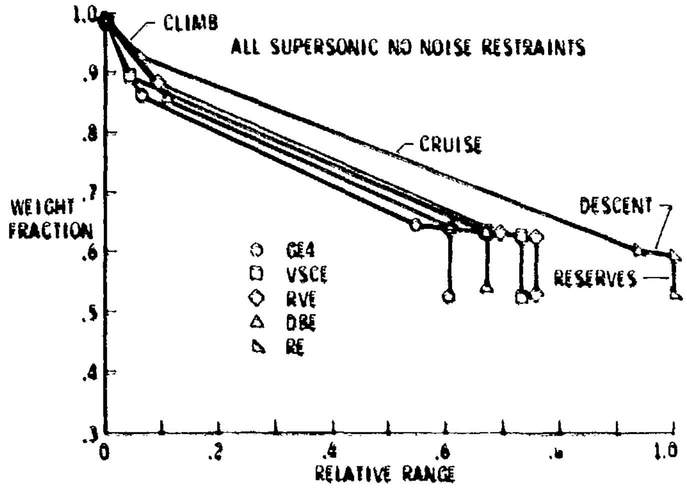
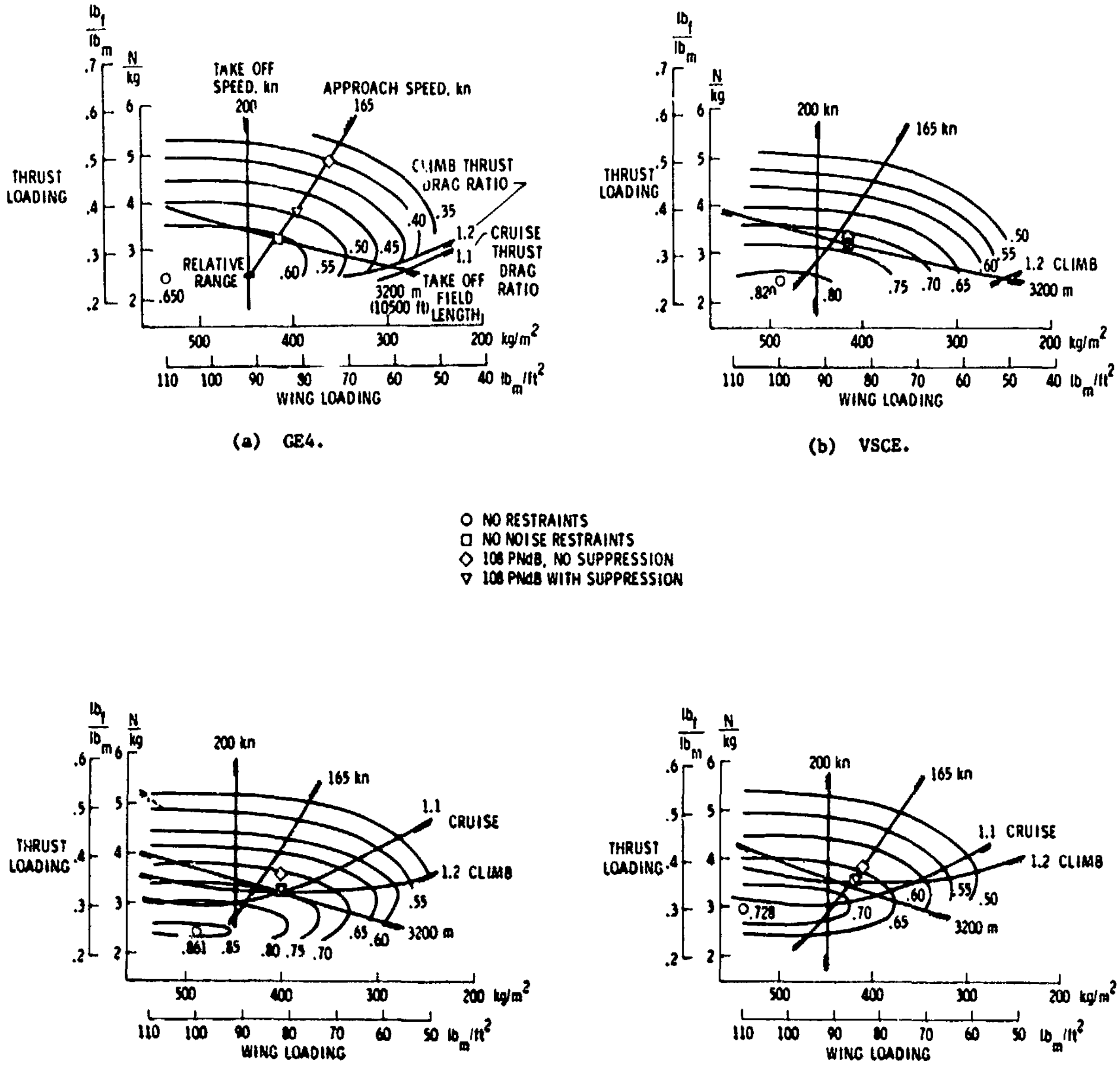


Figure 11. Fuel usage versus range.



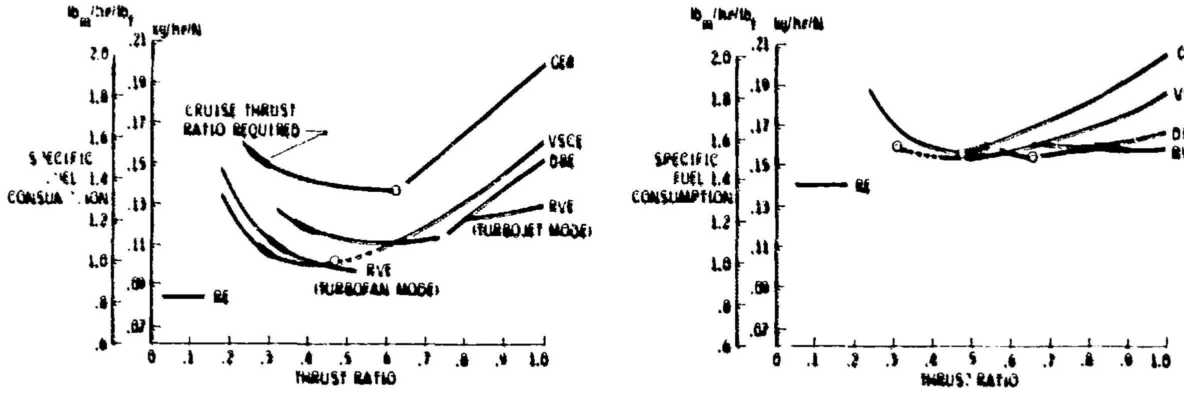
(a) GE4.

(b) VSCE.

(c) RVE.

(d) DBE.

Figure 12. Engine airplane range diagrams.



(a) Mach number = 0.9.

(b) Mach number = 2.62.

Figure 13. Installed engine performance.

DISCUSSION

H.A.Goldsmith

The effect of the co-annular silencing phenomenon on range at fixed noise appears to be remarkably small, particularly for the VSCE engine (Figures 9 and 12). Is this a correct interpretation?

Author's Reply

Yes. However, this is a result of a rather restrictive assumption of a takeoff field length of 3,200 m (10,500 ft.). The VSCE engine sized to meet this takeoff field length exhibits a noise level 4 EPNdB less than the allowable FAR value of 108 (untraded). Increasing the field length to 3,810 (12,500 ft.) would have permitted the co-annular effect to be more effectively used to meet the noise limit with a consequent increase in range of approximately five percent.

J.F.Chevalier

Dans le moteur RVE – Ces moteurs ont une température de réchauffe de 1311°C d'après votre papier, n'est-ce-pas? Dans le cas du moteur DBE ça doit correspondre à une élévation de température très faible dans le flux secondaire. On serait tenter, donc, d'essayer un double flux, sans chauffe du flux secondaire, et légèrement plus gros, de même technologie. Est-ce que l'on n'aurait pas un bon résultat également?

Author's Reply

In reply to the first question – The maximum outlet temperature of the secondary burner is indeed 1311°C. The secondary burner is not a reheater but a combustor which is used during the "twin turbojet" mode of operation to heat the airflow entering the rear turbine element.

To answer the second question – The use of an advanced turbofan engine without reheat for propelling an aircraft to a Mach number of 2.7 would lead to a prohibitively large engine. Although the good specific weight (weight per unit airflow) and specific fuel consumption are attractive, the low specific thrust requires large airflow rates to achieve the thrust required to accelerate and cruise at the design speed. Reheat applied to the bypassed airflow of such an engine would result in a cycle basically similar to the VSCE.

USE OF ENGINE VARIABLES

TO IMPROVE MILITARY PERFORMANCE

by

N.G. HATTON and B. SWANN

Rolls-Royce (1971) Limited, Derby England

SUMMARY

Military aircraft require superior engine performance over a wide range of flight conditions which in some cases involve operation far from the engine design point. If the cycle can be varied by geometric changes to allow operation nearer the design point then there is potential for significantly improving engine performance.

This paper covers several such cases, first examining the cycle changes which could be of value, then the consequent geometric changes that would be required. The effect of representative losses due to cycle and geometric changes are illustrated and with the inclusion of these not all the cases examined indicate success for variable cycles. However, with a theoretical investigation of this type it is considered to be worth including even those.

1. SELECTION OF CASES TO BE INVESTIGATED

This paper is restricted to conventional aircraft in that it does not include those with VTOL capability. The selection of cases worth investigating therefore resolves itself into looking for parts of the flight plan well removed from engine design conditions.

We have considered engines of moderate bypass ratio designed at sea level static conditions, which again is conventional. Figure 1 shows, typically, how far the airflow and turbine inlet temperature of these engines fall away together at low corrected speeds and relates these to the flight condition.

Using this diagram three flight conditions have been selected:-

1.1 Low Thrust Requirement, at Low Altitude and Low Mach No.

This condition covers combat aircraft with a long cruise or loiter requirement and maritime reconnaissance aircraft. For the cases in mind altitude would be in the range 0 to 1500 metres and Mach No. from 0.3 to 0.5. The thrust requirement could be as low as 12 to 30% of the maximum engine thrust at this condition and leads to the lowest corrected speeds at which performance is of importance. The fuel used here can be a very significant fraction of the total mission fuel requirement so that the target is to attain the lowest possible specific fuel consumption.

1.2 Maximum Thrust at Sea Level at Forward Speed

Even at maximum turbine inlet temperature the corrected speed of the engine falls with increase of forward speed due to increase in inlet temperature and by the time the Mach No. range 0.8 to 1.2 is reached the corrected airflow has fallen by 10% to 15%. Scope therefore exists for restoring the cycle to nearer the design point and so increasing thrust. Restoration by a system of variables would have the merit of increasing thrust without the overstressing and reduction of engine life involved in merely opening the throttle and could give lower fuel consumption.

1.3 Maximum Thrust at Altitude

At higher altitude similar considerations apply as at sea level when the maximum thrust is required. However, due to the lower ambient temperature the corrected speed of the engine does not fall severely until higher Mach No. At 6km the Mach No. range 1.2 to 1.6 is worth exploring and at these higher flight speeds a reheated engine is assumed.

1.4 Cases Involving High Corrected Speeds

A restriction on thrust could result from the engine running at higher corrected speeds than design, for example, in the stratosphere at subsonic speeds where the compressor flow may choke. However it is difficult to consider means of avoiding this which do not amount to designing the engine to pass more airflow from the outset with consequent increased size and weight. This aspect has therefore not been considered.

2. IMPROVEMENT IN SFC AT LOW THRUST, LOW ALTITUDE AND LOW MACH NO.

Figure 2 shows in simple terms what is potentially available. A typical fuel consumption against thrust curve is shown with the shape usually observed, namely an improvement in SFC on throttling from the maximum thrust to a minimum value followed by a deterioration as thrust is reduced further. Military aircraft operating subsonically commonly have to operate on the rising left hand side of the curve for long periods, and a saving in fuel could well be worth the cost involved.

The SFC can be expressed in terms of the thermal and propulsive efficiencies of the cycle.

$$SFC = \frac{K \qquad Vo}{\eta_{THERMAL} \times \eta_{PROPULSIVE}}$$

where Vo is the flight speed and K is a constant

The thermal and propulsive efficiencies are also plotted in Figure 2. With reduced thrust, thermal efficiency falls due to reduction of temperature and pressure but in principle thermal efficiency could be kept constant if temperature and pressure were maintained constant by geometric variables. Propulsive efficiency on the other hand increases because airflow does not fall so rapidly as thrust. However, if flow could be maintained constant the propulsive efficiency could be still higher.

It can be shown that if both these aims could be realised the SFC curve would follow a linear relation and would fall to astonishingly low levels at low thrust, offering a prize well worth investigating further.

The case which has been taken for further examination, from a number of possibilities, is a Maritime Reconnaissance mission at sea level with Mach No. 0.3. The operating point is in fact the point indicated on Figure 2. The thrust is assumed to be only 13% of the maximum available. This may seem exceptionally low, but the location on the SFC loop is not untypical of other military missions.

The design cycle is as follows:

Bypass ratio 1.5
Pressure ratio 20
Turbine entry temp. 1600°K (at 303°K compressor entry temp.)

The engine is assumed to have 2 shafts and to be of simple configuration in which the flows split downstream of the LP compressor. This means that with a separate jet engine there are four geometric variables which can be used actively to control the engine cycle:-

HP turbine capacity
LP turbine capacity
Primary nozzle area
Bypass nozzle area

With a mixed engine there is only one final nozzle but there are still four variables as a further variable is the ratio between hot and cold stream areas in the mixer.

2.1 Effect of Cycle Changes on SFC (Severely throttled operation)

Figure 3 shows the effect of gradually restoring the cycle operating parameters from the low values at the throttled point to design point values while keeping the thrust constant.

Separate jets are assumed and the first step is to correct the jet velocity balance. The primary velocity to bypass velocity ratio is too low and is increased by nozzle area changes to the optimum value. This is worth 2½% of sfc and this optimum is maintained for the remainder of figure 3.

Thermal efficiency is next restored by temperature and pressure increase. Together these reduce the fuel by no less than 40% and clearly improvement is obtained more rapidly by increasing temperature before increasing pressure ratio.

Restoration of the design corrected airflow improves the propulsive efficiency to give the total saving of fuel of 47%. Due to secondary effects like offtakes and duct losses this is not so great as predicted from the simplified treatment of figure 2 but still very well worth further investigation.

However the changes which are necessary to produce this effect are rather discouraging. As a measure of cycle change the bypass ratio increases from near 2 to over 20 for full restoration.

The area changes shown in figure 4 are very large. Of most importance, all three area restrictions controlling the core flow have to be decreased very severely and more or less in step. It might have been hoped that the first step in change of geometry would have given a disproportionate gain in sfc but this is not so. To gain the first 20% improvement in sfc a reduction in HP turbine capacity of over 50% (and corresponding changes in the other nozzles) would be required.

For a mixed engine the assumption is made that the total mixer area (hot + cold chutes) is constant. Fig.5 shows that the picture is quite similar to the separate jets case with the hot chute area replacing the primary nozzle.

2.2 Effect of Geometric Changes

The previous section has shown the effect of imposed cycle changes and given some idea of the geometric changes that would be needed to achieve the cycle. The next step is to take the given engine design and find the direct, individual, effects on sfc of changes in geometry. A seperate jets configuration is considered.

Assumptions must be made about the losses that are caused by the variability. In the turbines it is assumed that where losses occur they will be due to change of capacity caused by variation of the inlet nozzle guide vanes.

The variation of loss is shown in figure 6 and is taken from early Rolls-Royce tests. The level of loss no doubt can be improved and the effect of its total elimination is shown later.

Extra indirect loss may also occur in the turbines due to movement away from the normal operating points. This will also happen in the compressors with the additional problems of movement of the operating point towards the surge line or towards choking with possible structural hazards. Variability in the compressor therefore may also have to be invoked.

The magnitude of the penalties due to movement of component operating point clearly depend on the component characteristic assumed. The components used are representative of conventional engines without variables and it is possible that a rematching or re-design of the basic component to cope with the demands on working point movement would be necessary. The effect of totally eliminating these losses is shown later.

2.2.1 Results with Full Loss in Turbines and No Variability in Compressor

The most pessimistic assumptions are made initially that the full turbine losses discussed above are present and that the compressors have no built in, variability. The results in figure 7 reveal a conspicuous lack of success, as small gains (at best 1½%) are only obtained for reduction of primary nozzle area and reduction of HP turbine capacity. This is due mainly to component deterioration. In the turbines it was observed that HP turbine variability caused only HP turbine loss and LP turbine variability caused only LP turbine loss. There were no significant interaction effects. The loss of efficiency is plotted in figure 8 and indicates appreciable additional losses due to working point movement and suggests that the turbines could have been better matched to cope with the variability. On the LP compressor, figure 9 shows that only the bypass nozzle caused movement away from the working line, while on the HP compressor (Fig.10) reduction of HP turbine capacity causes severe movement towards the surge line and reduction of LP capacity a slight movement in that direction. The basic engine is well matched to the highest efficiency region so that any movement will lead to additional losses.

A summary of the effects of the four component changes is as follows:

VARIABLE	SFC IMPROVEMENT	COMMENT
Primary Nozzle Area	1.5% Max. improvement with 20% area reduction	Improvement due to reduction in jet velocity ratio soon overtaken by reducing HP compressor efficiency
Bypass Nozzle Area	The expected improvement - due to increased flow with increased area does not materialise	Improvement due to increased flow and reduced jet velocity ratio more than offset by reduced LP compressor efficiency and bypass duct loss increase
HP Turbine Capacity	1% maximum improvement with 5% capacity reduction	Improvement in cycle (increased temperatures and pressure) overtaken by rapidly increasing HP turbine and HP compressor efficiency loss
LP Turbine Capacity	No improvement with reduced LP turbine	Small net improvement in cycle (temperature up, pressure down) more than offset by LP turbine efficiency loss

The conclusion must be that gain in performance is being limited in each case by the additional losses which are being introduced.

2.2.2 Result with Losses due to Movement of Working Lines Eliminated but loss directly due to Turbine Capacity Changes Retained

A more optimistic assumption is now made that the introduction of variables on the compressors would eliminate any deterioration in efficiency (as well as avoiding surge). Similarly the loss of efficiency in the turbine due to working line movements is assumed eliminated due to a combination of rematching and redesign. The direct loss due to turbine variability itself is retained. The resulting gain in performance is however only marginally increased as shown by comparing fig.11 with fig.7 and requires a greater geometric change to take advantage of it.

2.2.3 Results with all Losses in Turbomachinery due to Variability Eliminated

This change can improve only turbine capacity effects. For the first time (as shown by fig.12) the reduction in SFC due to HP turbine capacity is quite large. However this is by far the most difficult variable to use for both mechanical and aerodynamic reasons.

It is worth stating what would be required to achieve say 5% sfc improvement due to the latter.

(a) No HP turbine loss with a 22% reduction in turbine capacity

(b) No HP compressor loss with a 15% reduction in flow at a pressure ratio.

This sets the designer a next to impossible target and the effort could be incommensurate with the possible gains.

2.2.4 Combinations of Variables

Particular combinations of variables however could be more successful than the sum of the individual effects. From the cycle investigation shown in figures 3 and 4 it appeared that all three geometric variables controlling core flow should be changed together. Applying this to the engine design being investigated and the most optimistic component deterioration assumption, all three capacities were reduced by 50%. The SFC reduction was 20% largely confirming the simple cycle calculations.

How a variation of this magnitude could be implemented practically is not too clear. One way of reproducing the cycle effects on a multi-engine situation would be to have the HP compressor of two engines feeding the combustion and turbine system of only one. This would involve elaborate ducting and cross shafting but would not need actual turbine variation.

However since a multi-engine situation is inherent in this scheme the simpler possibility exists of merely closing down one engine of a pair and opening the throttle of the other engine to double its net thrust. This takes advantage of the shape of the SFC loop and in the case considered the reduction in fuel can be seen to be about 20% also.

3. INCREASE THRUST AT FORWARD SPEED

To assess what scope exists for increasing thrust at forward speed the first step is to examine the operating conditions of a conventional engine against the assumed design point at sea level static.

The engine cycle considered is as follows at design point:-

Bypass Ratio	0.5
Overall Pressure Ratio	20
Turbine Entry Temperature	1600°K (at 303°K inlet temp.)

Reheat capability is assumed and therefore a mixed stream engine. Fig.13 covers both the 0.9 Mach No. sea level, and 1.6 Mach No, 6 km cases and shows that the corrected flow is well below design, as is pressure ratio particularly on the LP compressor. Also the actual LP shaft speed is below design so that the prospects look good for appreciable thrust gain by exploiting this underused capacity.

The simplest way to increase thrust is merely to open the throttle and it is against this action that all other possibilities have to be judged. It would be possible to open the throttle until the aerodynamics are restored to the design conditions but by this time the turbine entry temperature, the compressor delivery temperature and the speeds would be excessive.

The aim here is by means of variables to exploit the full aerodynamic capability of the compressors (together with LP compressor speed restoration) with minimum increase of operating temperatures.

The variables at our disposal can be listed initially as:

(a) Final Nozzle

In a reheated engine this is an inbuilt feature

(b) Mixer

Within a constant total mixer area the hot area can be increased at the expense of the cold.

(c) LP turbine capacity

(HP Turbine capacity change has not been considered. It is the most difficult to achieve and preliminary work showed no merit in its use).

To help in understanding which way the cycle should be changed to give most thrust the results of simple cycle calculations with constant efficiencies are given in Fig.14. The total corrected airflow is assumed to be restored to its design value and the turbine entry temperature (and reheat temperature) is held constant. With constant temperature and constant HP nozzle guide vanes it can be shown that the core flow is directly proportional to the overall pressure ratio. It is seen that on this simple basis at 0.9 Mach No, 18% thrust increase is predicted (mainly through increase of core flow) and at 1.6 Mach No, reheated, 41% thrust increase is predicted (mainly through increase of the total engine flow).

3.1 Application of Variable Geometry

It appears from the foregoing that the key to increase of thrust is in the LP compressor either via restoration of airflow or restoration of pressure ratio or both. The working line on the compressor has therefore been shown in Fig.15.

A working line with throttle opening is shown compared with the effect of increasing the three selected geometric variables. Two variables produce movement in more or less the right direction on the LP compressor (while the LP turbine capacity effect is largely confined to increasing the pressure of the HP compressor). The diagram suggests that a judicious combination of final nozzle area and mixer hot area increase would give a large thrust increase and could leave the operating point on the normal working line. This would eliminate the need for any special variable on the LP compressor.

This diagram also raises the question of limits. How far should one push the flow and speed increases involved? It is difficult to fix hard and fast limits since one parameter can be traded against another. However the following tentative list was drawn up:-

(a) LP compressor corrected flow not to exceed design + 5% (with LP pressure fixed as the corresponding point on the normal working line).

(b) HP compressor corrected flow, not to exceed design +10%.

(c) Shaft speeds. Precise limits are not assigned to these as use of variables in the compressor could allow the speed at a flow to be decreased somewhat. Cases to be considered on merits.

(d) Compressor delivery temperature. No limit is set for this, as it must be regarded as a main parameter in assessing the success of the operation. With sophisticated turbine cooling systems it is at least as valuable in terms of turbine blade life to cool the compressor delivery cooling air as it is to cool the turbine entry temperature by the same amount.

3.2 Thrust Increase at Mach No. 0.9 Sea Level, Dry

In assessing the merits of the alternative ways of thrust increase a target of 10% increase is set. Results in Fig.16 are given both with constant component efficiency (the most optimistic) and the most pessimistic standard with full losses in the turbines and fixed compressor characteristics (the latter given in parenthesis).

Considering the optimistic standard first the best variation would be a combination of increased final nozzle and mixer primary chute.With a thrust increase of 10%, the SFC is slightly better and the compressor delivery temperature is only 7°K in excess of the throttle opening case to set against saving the 53°K increase of turbine entry temperature that this would involve. This would be quite attractive. However with the full loss standard only 5% thrust increase can be achieved with this combination of variables and the advantage in terms of SFC improvement and engine operating temperatures relative to throttle opening has been lost.

The thrust gain found here could have been larger if it were not for the penalty carried for the reheat capability of this engine. The pressure loss is assumed to be 5% at design but with opening of the nozzle this increases severely. However if in the context of this paper we can be allowed the luxury of a variable reheat system with gutters and possibly fuel pipes stowed away when not in use then the pressure loss can be largely eliminated (to say a nominal 1%). The performance even without the other variables is improved by nearly 2% of thrust and SFC and this is included with the gain due to the other variables in Fig.17. The max. gain allowing for a practical loss standard in the other components is then 8% at which there is a better SFC and overall lower operating temperature relative to a fixed engine with throttle opening.

3.3 Increase in Thrust at Mach No.1.6, 6 Km, Reheated

At this condition the use of variables is more successful than the non-reheated case just considered. The competition is not so severe since to get 10% more thrust at a given reheat temperature by increased turbine temperature now requires 82°C rather than 53°C (Fig.18). A combination of final nozzle and mixer chute area is still the most promising and at this flight condition is much less subject to deterioration by the introduction of a practical loss standard. Even assuming this, a gain of 10% of thrust is predicted with no increase in compressor delivery temperature over opening the throttle for an SFC penalty of 3%. However reheat operation could be rather difficult in practice, because the ratio of total pressure in the mixer (cold to hot) is high.

4. CONCLUDING REMARKS

One important aspect the authors suspected initially and have become increasingly aware of during the preparation of this paper is the extent to which assumptions about the cycle, the way the engine is matched, and how different parts are sized could influence the results, although s far as possible representative assumptions have been made. In particular it had been intended to cover a range of bypass ratios, choice of which may exert a major influence on the conclusions, but this has proved beyond the scope of the current paper.

The engine layout too should not be disregarded. For example LP shafts which have some of the core compressor stages rotating with them have been excluded from this paper but in fact may well prove to be important in that they allow the introduction of a further active variable changing the cycle.

Subject to these reservations it is possible to draw the following conclusions for the flight conditions considered:-

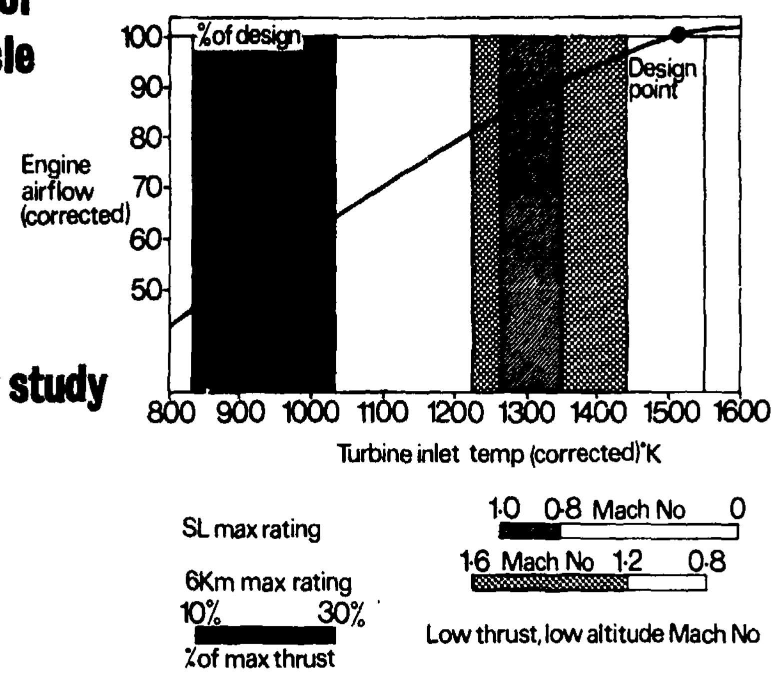
(a) It is only possible to improve the throttled specific fuel consumption significantly by variables if the HP turbine capacity, the most difficult, is varied. Even then the losses introduced must be kept small.

(b) Good prospects exist for increase of thrust at high forward speed of up to 10% by varying the final nozzle and mixer areas.

5. ACKNOWLEDGEMENT

The authors wish to thank their colleagues at Rolls-Royce, Derby, for help in the preparation of this paper.

Application of variable cycle engines



Selection of cases for study

FIG. 1

The potential improvement of specific fuel consumption. Throttled engine, low altitude, low Mach. No

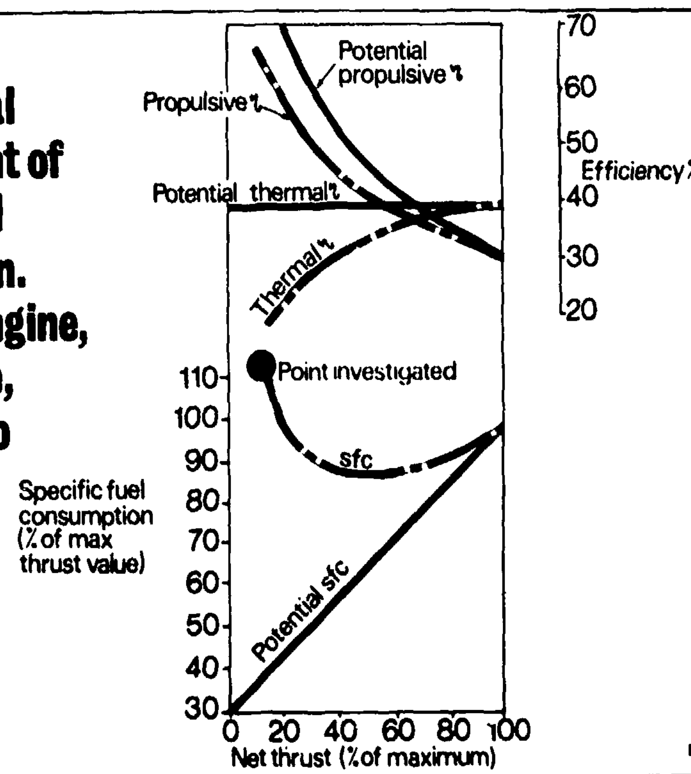


FIG. 2

Effect of cycle on sfc-thrust constant. Engine throttled to 13% max. thrust-sea level, Mach. 0·3, ISA

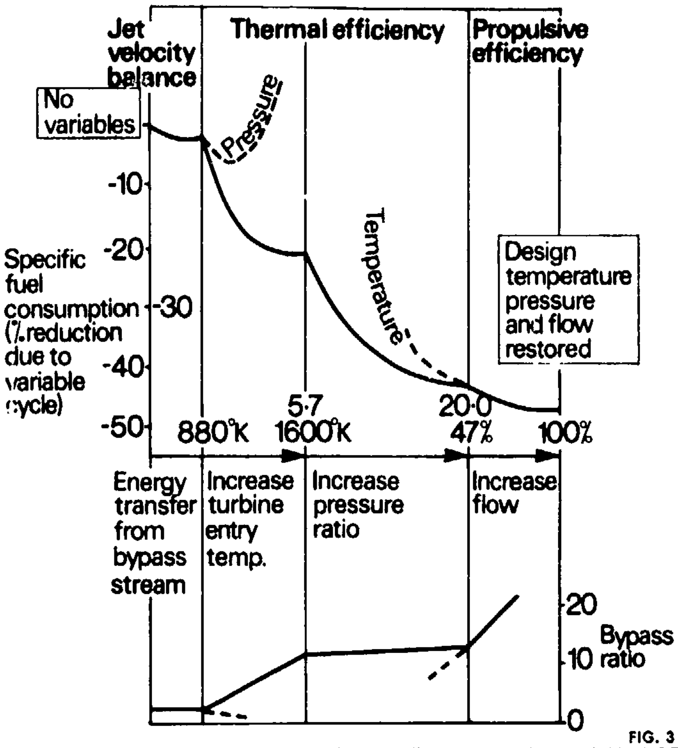


FIG. 3

Nozzle area and turbine capacity changes required

Separate jets

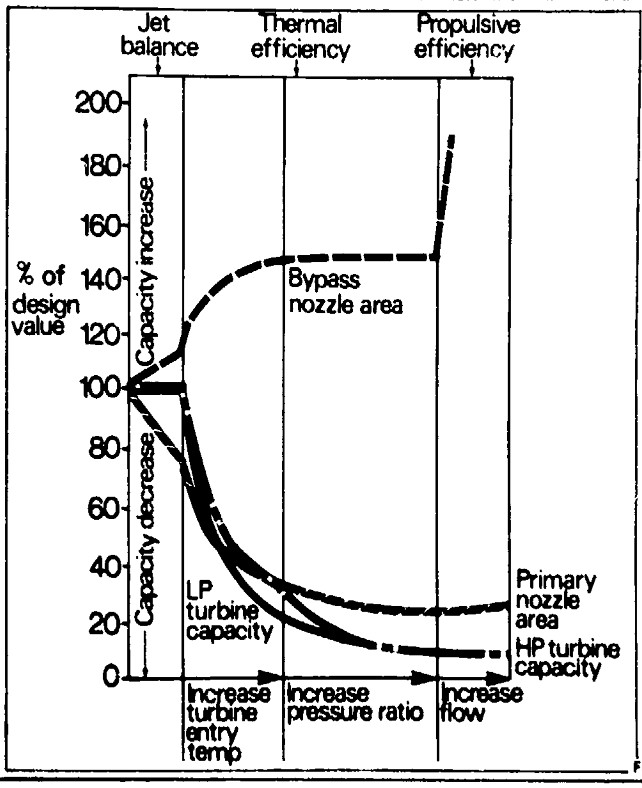


FIG. 4

Nozzle area and turbine capacity changes required. Mixed jet

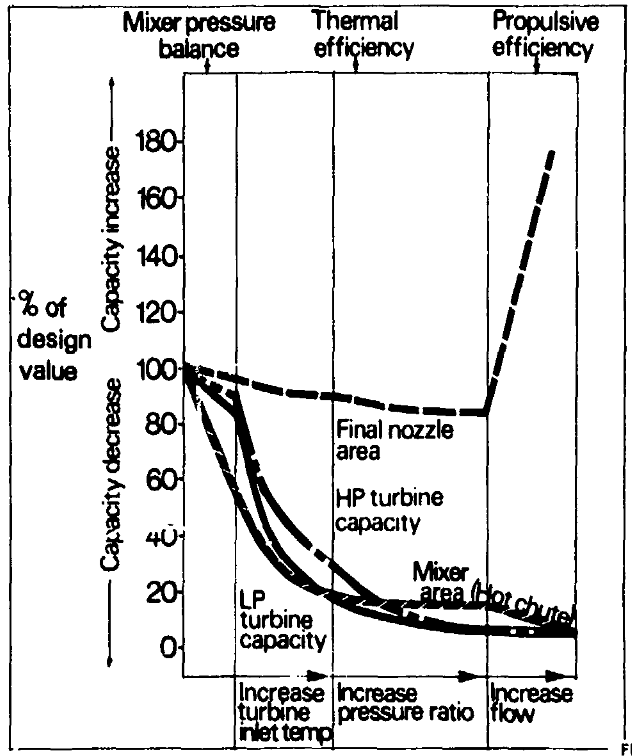


FIG. 5

Assumed direct loss of turbine efficiency due to capacity change

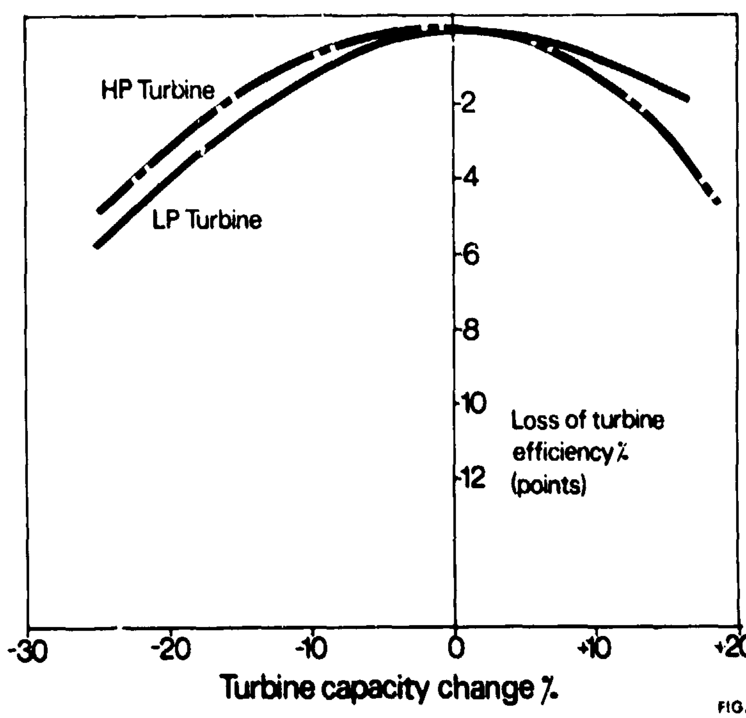


FIG. 6

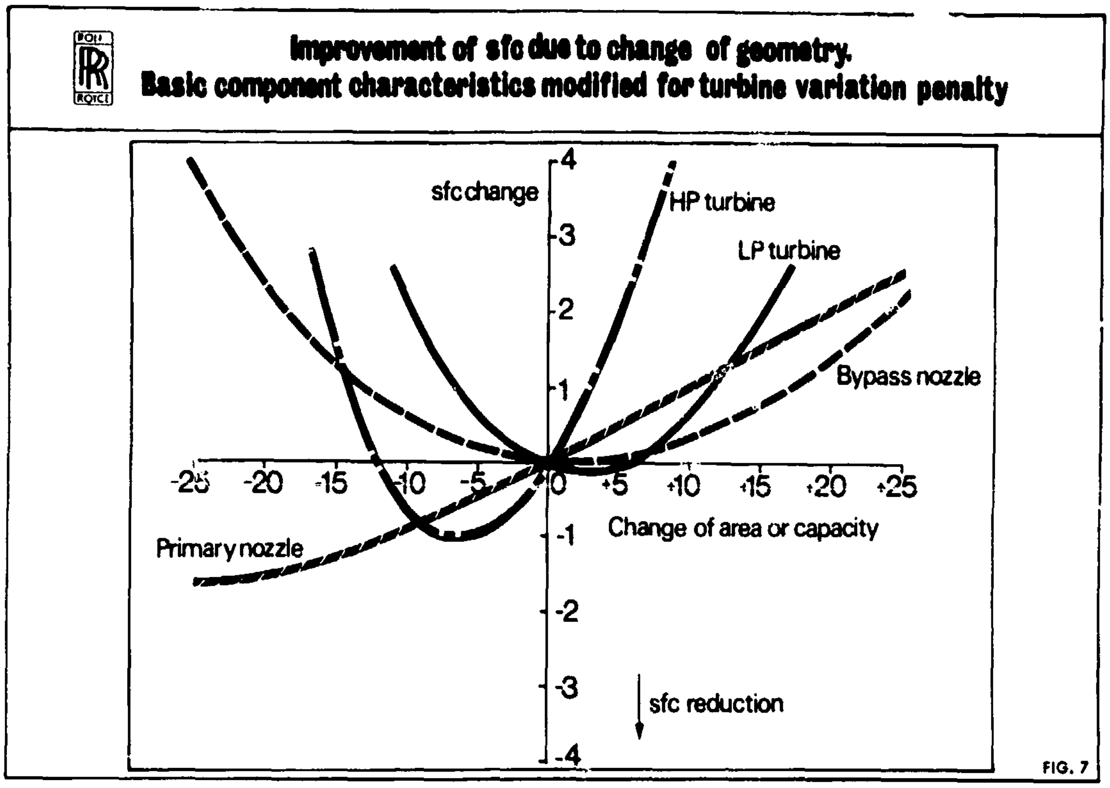

Improvement of sfc due to change of geometry.
Basic component characteristics modified for turbine variation penalty
sfc change
HP turbine
LP turbine
Bypass nozzle
Primary nozzle
Change of area or capacity
sfc reduction
4
3
2
1
-1
-2
-3
-4
-25
-20
-15
-10
-5
0
+5
+10
+15
+20
+25

FIG. 7

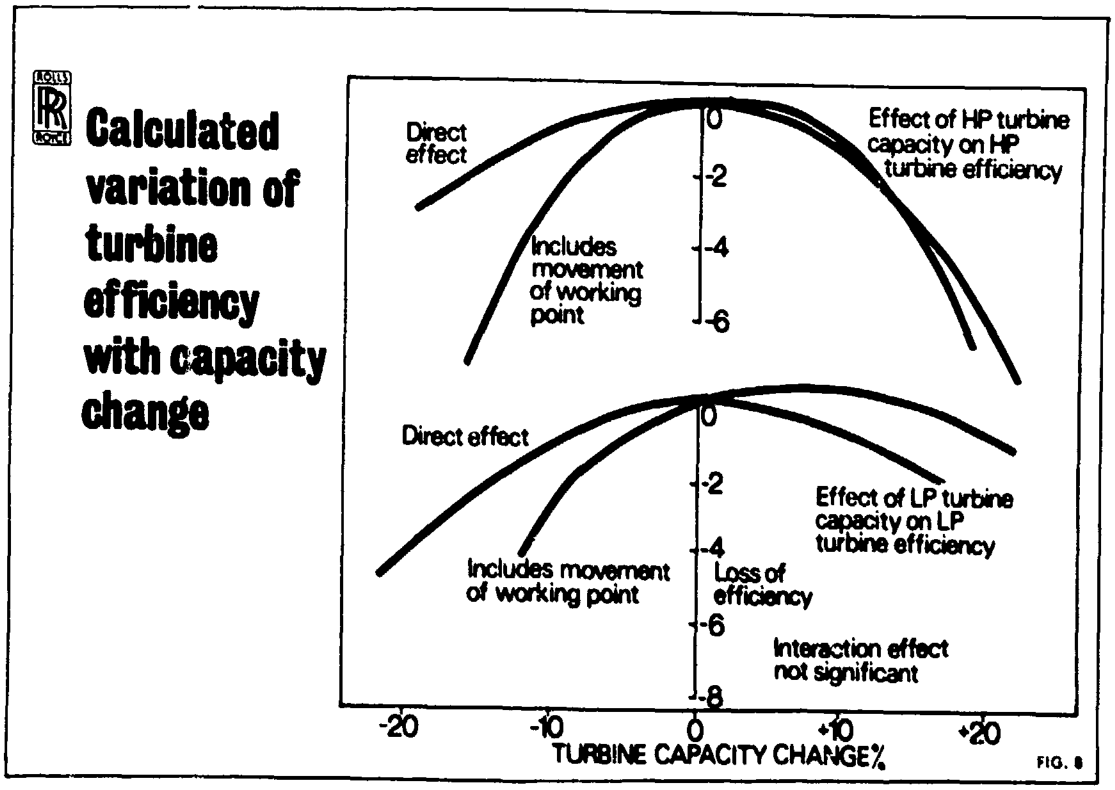

Calculated variation of turbine efficiency with capacity change
Direct effect
Includes movement of working point
Effect of HP turbine capacity on HP turbine efficiency
Direct effect
Includes movement of working point
Effect of LP turbine capacity on LP turbine efficiency
Loss of efficiency
Interaction effect not significant
0
-2
-4
-6
-8
-20
-10
0
+10
+20
TURBINE CAPACITY CHANGE%

FIG. 8

LP compressor characteristics

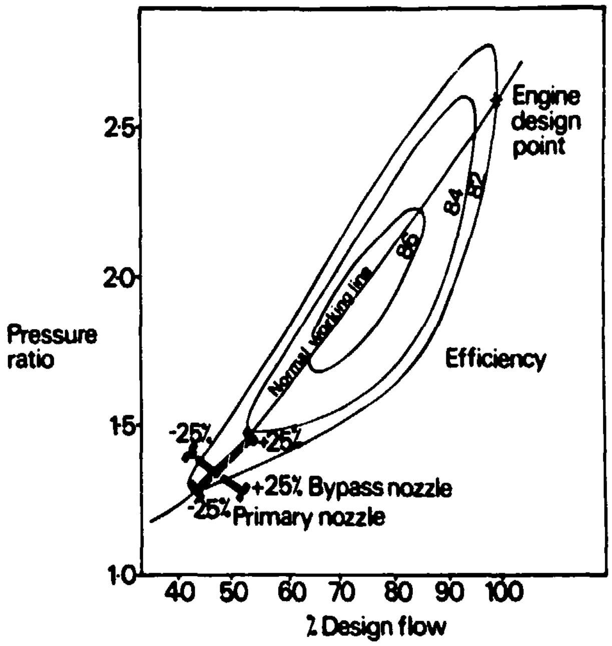


FIG. 9

HP compressor characteristics

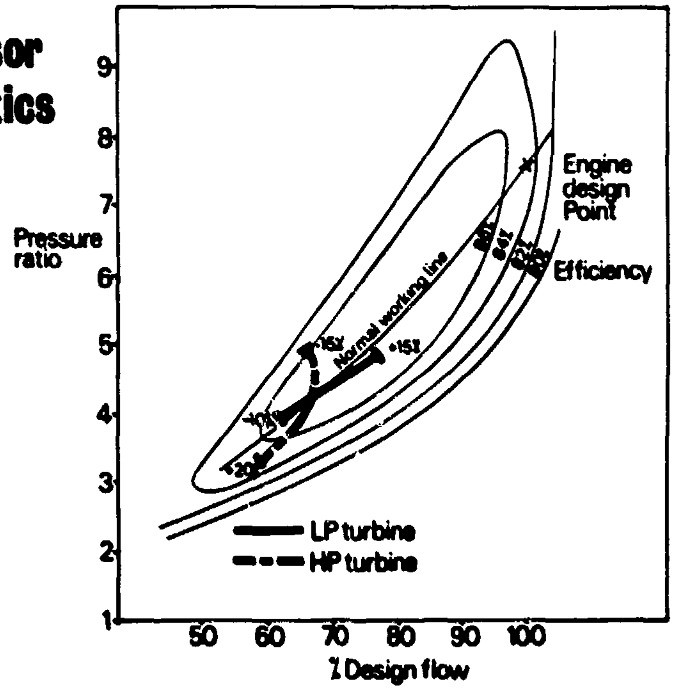
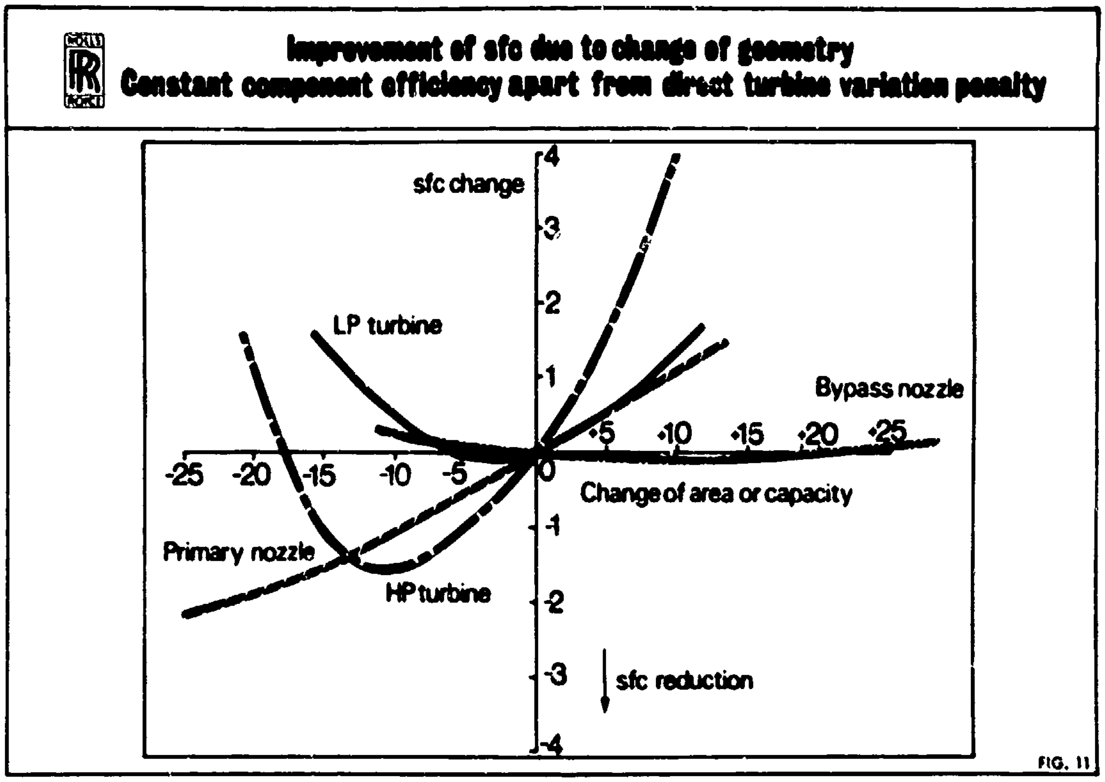
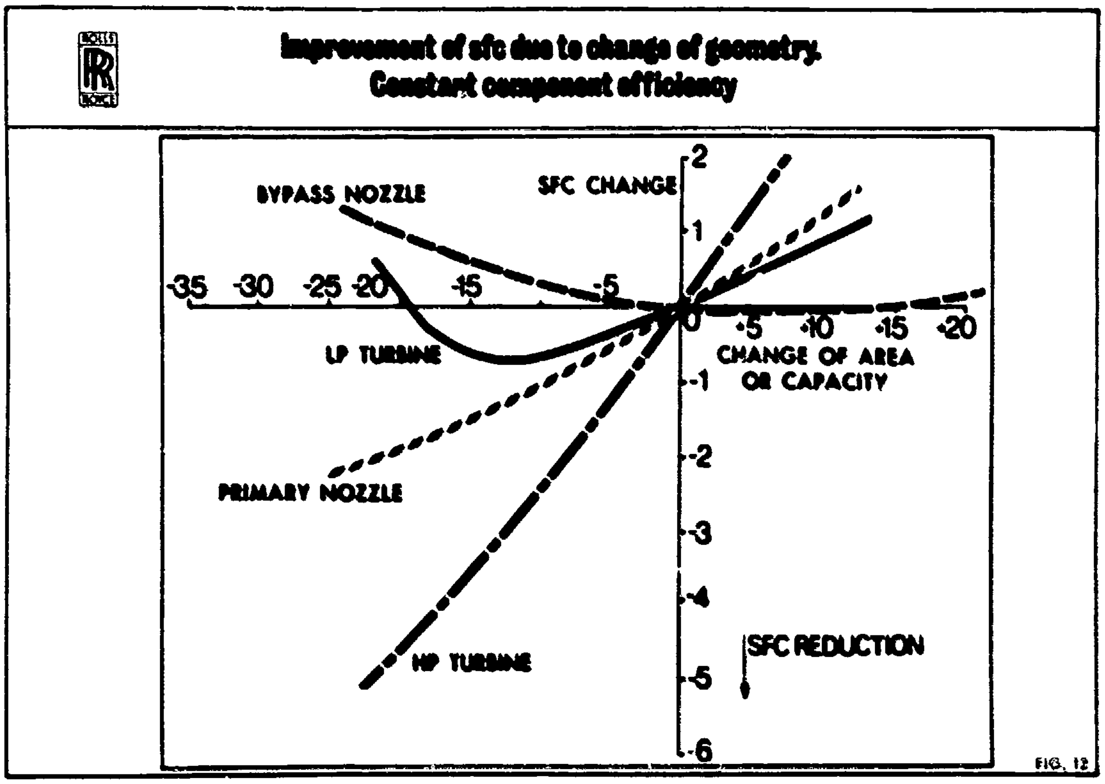


FIG. 10



Improvement of sfc due to change of geometry
Constant component efficiency apart from direct turbine variation penalty
sfc change
4
3
2
1
-1
-2
-3
-4
LP turbine
Bypass nozzle
-25 -20 -15 -10 -5 0 +5 +10 +15 +20 +25
Change of area or capacity
Primary nozzle
HP turbine
sfc reduction

FIG. 11



Improvement of sfc due to change of geometry.
Constant component efficiency
BYPASS NOZZLE
SFC CHANGE
2
1
-35 -30 -25 -20 -15 -5 0 +5 +10 +15 +20
LP TURBINE
CHANGE OF AREA OR CAPACITY
-1
-2
-3
-4
-5
-6
PRIMARY NOZZLE
HP TURBINE
SFC REDUCTION

FIG. 12

Cycle change at forward speed

Turbine entry temperature 1600°K

	Airflow (corr)	Pressure ratio LP compressor	Pressure ratio HP compressor	Shaft speed LP	Shaft speed HP
Design	Datum	Datum	Datum	Datum	Datum
M0.9, sea level	-14%	-17%	-6%	-10%	-1%
M1.6, 6km (20,000 ft)	-26%	-31%	-13%	-14%	-1%

FIG. 13

Thrust increase due to cycle changes

	0·9 Mach. No. sea level dry		1·6 Mach. No. 6 km reheated	
	Total airflow	Thrust	Total airflow	Thrust
Engine without variables	100	100	100	100
Total flow restored (to corrected design value)	116	104	136	129
Core flow restored (pressure ratio, design value)	116	118	136	141
Turbine entry temp. Reheat temp. HP turbine capacity	Constant			

FIG. 14

LP compressor operating points

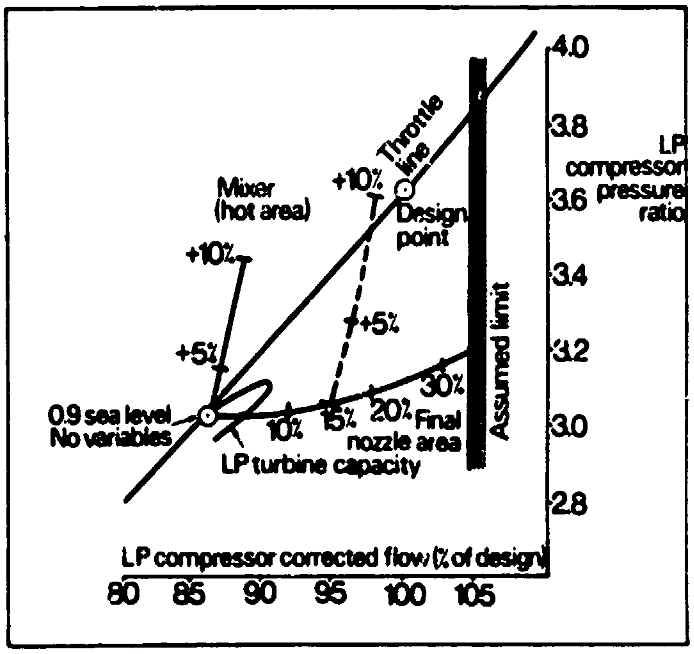


FIG. 15

Thrust increase by use of variables M=0·9 sea level reheat off

Aim for 10% thrust increase
Jet pipe pressure loss (cold) 5%

	Thrust increase %	SFC increase %	Compressor delivery temp °C (rel.normal max.)	Comments
Open throttle	+10	+1	+19	Turbine entry temp +53 °C
Open final nozzle +35%	+7(−ve)	−2(—)	+14(—)	Limited by LP compressor corrected flow
Open mixer primary chute +13%	+10(+4)	+1(+1)	+32(+13)	
Combination of final nozzle 15% (18%) & mix primary chute +9% (15%)	+10(+5)	−1(+1)	+26(+31)	Operates on LP compressor normal working line, therefore needs no compressor variables
Increase LP turbine capacity +25%	+8(−ve)	+1(—)	+25(—)	HP compressor corr. flow on 10% limit

FIG. 16

BASIC FIGURES – CONSTANT EFFICIENCY   FIGURES IN BRACKETS–FIXED COMPONENTS

Thrust increase by use of variables
M=0·9 sea level Reheat off

Aim for 10% thrust increase
Jet pipe pressure loss cold, 1% (reheat systems retracted)

	Thrust increase %	SFC increase %	Compressor delivery temp °C (rel normal max.)	Comments
Open throttle	+10	+1	+19	Turbine entry temp +53C
Open final nozzle +28%	+10(-ve)	-6(–)	+12(–)	Limited by LP compressor corrected flow
Open mixer primary chute +13%	+10(+6)	+1(0)	+28(+15)	
Combination of final nozzle 11%(9%) & mixer primary chute +8%(15%)	+10(+8)	-3(-2)	+20(+29)	Operates on LP compressor normal working line, therefore, no compressor variables

BASIC FIGURES–CONSTANT EFFICIENCY FIGURES IN BRACKETS–FIXED COMPONENTS FIG. 17

Thrust increase by use of variables
M-1·6 6km(20,000ft) max. reheat in all cases

Aim for 10% thrust increase

	Thrust increase %	SFC increase %	Compressor delivery temp K (rel. normal max)	Comments	Pressure ratio at mixer Bypass primary
Open throttle (increase turbine entry temperature at fixed reheat temperature)	+10	-2	+28	Turbine entry temperature +82C	1·07
Open final nozzle +30%	+10(+7)	+4(+6)	+11(+9)	Mixer pressure ratio high	1·22
Open mixer primary chute +17%	+10(+2)	-4(-1)	+63(+25)		1·47
Combination of final nozzle and mixer primary chute Final +16% mixer +7% (+12%)	+10(+10)	+1(+1)	+25(+28)		1·29

BASIC FIGURES - CONSTANT EFFICIENCY FIGURES IN BRACKETS-FIXED COMPONENTS FIG. 18

DISCUSSION

J.C.Ripoll

Figure 2 shows a linear decrease of SFC down to zero thrust as a potential possibility. Does not thermodynamics impose an increase towards infinite values near zero thrust?

Vous avez montré une figure où vous indiquez que la consommation spécifique peut être ramenée à zéro pour des très faibles poussées. Je me demande si les lois de la thermodynamique n'imposent pas, au contraire, qu'il y ait toujours une asymptote vers l'infini pour une poussée nulle, ce qui impliquerait qu'on ne puisse pas obtenir des résultats aussi bons que ceux que vous avez indiqués comme potentiels. Est-ce que vous pourriez commenter cette idée.

Author's Reply

'The linear decrease of sfc with reduction of thrust shown in figure 2 is consistent with the assumption that with variable cycle the thermal efficiency can be maintained constant at reducing thrust. This requires that the combustion temperature, the compressor pressure ratio and all the component efficiencies are maintained constant. At the same time flow through the core is reduced, (in the limiting case to zero at zero thrust) and the compressor pressure ratio in the bypass stream is also reduced. No losses are assumed other than in the rotating components. Thus in the limiting case of zero thrust all losses reduce to zero and this is consistent with the linear relation plotted in figure 2.

Of course this is a quite impractical state of affairs. If the airflow is finite (in fact assumed to be the design value) then clearly there will be finite parasitic losses, for example in the bypass duct, so that in practice there must be infinite sfc at the limiting zero thrust condition.

In the work following figure 2 therefore more realistic assumptions are made. Figure 3 is a step in this direction, in that, as the cycle is varied at a thrust, duct losses are maintained as a fixed percentage of pressure. At the specified thrust (13% of maximum) the potential reduction in fuel consumption is 47% compared with a reduction of approximately 65% indicated from figure 2.

J.P.Vleghert

You have talked about steady state performance of the engine. Could you say something about accelerating performance from idle as influenced by nozzle variation. Would that be improved with open nozzle especially when you have intake distortion in your engine?

Author's Reply

I think it must be improved. We have not investigated this in the context of this paper but if one has an extra degree of variability then clearly the possibility of controlling the working line of the compressor is improved.

H.Grieb

In the case of the mixed flow turbofan with variable HP- and LP-turbines and a variable mixer, you reduce dry thrust by the reduction of TET at a high level of mass flow and overall pressure ratio. This results in a higher pressure level of the duct flow compared with that of the primary flow behind the LP-turbine. The duct flow expands through the mixer to the pressure level of the primary flow. From our own studies we expect SFC-penalties at dry partload due to the energy loss of the duct flow. Did you investigate this point?

Author's Reply

'The dry low thrust cases we looked at were applicable to a separate jets engine. There is exactly the same problem in that the jet velocity ratio may move away from the optimum value. This tends to happen if a single controlling area is varied in isolation but can usually be prevented if suitable combinations of variables are employed.'

POSSIBILITIES OF ADAPTING BYPASS-ENGINES
TO THE REQUIREMENTS OF HIGHER SUPERSONIC FLIGHT

by
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SUMMARY

This paper investigates some methods of improving the adaptability of the reheated bypass engine to supersonic flight missions by varying the thermodynamic cycle. In particular, the following possibilities have been studied:

- fuel-rich primary-flow heating during reheat operation
- fuel-rich total-flow reheating during high-thrust mission segments
- precooling of turbine coolant by heat exchange with reheat fuel
- intercooling of primary air flow by heat exchange with reheat fuel.

SYMBOLS

A	area
a	specific work
AB	afterburner
C	compressor
c	velocity, specific heat capacity
D	drag
DP	design point
F	thrust
H	altitude
HPC	high pressure compressor
h	specific enthalpy
IPC	intermediate pressure compressor
k	specific heat ratio
LPC	low pressure compressor
M	Mach number
MP	matching point
$\dot{m}$	mass flow
N	rotor speed, RPM
p	pressure
T	temperature
TF	turbo-fan
TJ	turbo-jet
β	fuel/air ratio
γ	cooling-air/air ratio
η	efficiency
μ	bypass ratio
π	pressure ratio
τ	corrected air flow

SUBSCRIPTS

a	air
By	bypass
C	compressor
ca	cooling air
E	entrance
f	fuel
MB	main burner
s	isentropic
st	stoichiometric
t	total
∞	undisturbed flow
0	inlet face
1	compressor inlet
2	compressor outlet
3	turbine inlet
4	turbine outlet
4a	afterburner inlet
4b	afterburner outlet
5	nozzle outlet
I	primary flow
II	secondary flow

Most investigation of this paper were performed during the author's affiliation to the Institute of Jet Propulsion and Turbomachines, Technical University Aachen.

1. INTRODUCTION

For a long time, there has been a demand for aircraft which are supposed to fly missions both, in the subsonic range and - after passing through the transonic range - for extended durations in the supersonic range. This combination of flying missions requires adaptable engine systems allowing to accomplish different tasks by applying simple design principles and within the acceptable limits of fuel consumption.

In order to show some of the problems connected with the development of such dual-purpose engines, the following paragraphs will discuss how technological innovations in engine construction, and the increased requirements aircraft designers are trying to meet, have resulted in the initial development of the reheated turbo-fan. This will be done by comparing the suitability of the reheated turbo-jet and that of the reheated turbo-fan for current requirements.

Subsequently, possibilities will be presented and discussed for improving the thrust and fuel-flow characteristics of the engine type considered under typical loads. The chance for increasing the adaptability of the engines so much that it will cover even excessive Mach numbers will also be discussed.

2. THE CONVENTIONAL REHEATED TURBO-FAN

2.1 THE DEVELOPMENT AND FUNCTIONAL PRINCIPLES OF THE TURBO-FAN

Initially, the turbo-jet fulfilled the dual-mission requirements of supersonic aircraft. The lay-out and dimensions of the engine allowed to generate, under only slightly reduced unaugmented power, the thrust the aircraft needed for flying in the upper subsonic range and this with a relatively favorable specific fuel consumption. So, the amount of thrust produced by this type of engine, using maximum afterburning power, was sufficient for take-off, rapidly passing through the transonic range, and the compensation of the aircraft drag for the intended maximum Mach numbers between $M_\infty = 1.5$ and $M_\infty = 2$.

The degree of thrust augmentation depends on the degree of possible reheat. However, the specific fuel consumption normally also increases with increasing reheat (Fig. 1 top).

The achievable temperature ratio of the reheat has gradually decreased in the course of the development of engine technology. The considerable increase of the permissible turbine inlet temperature has been accompanied by a corresponding increase of temperature at the afterburner entry. Since this rendered the cooling of the afterburner more difficult, the maximum permissible reheat temperature at the afterburner exit could only be slightly increased.

For lowering the total temperature at the afterburner entry, the obvious solution seemed to be to substitute a jet-mixing turbo-fan for the turbo-jet. In this case, the more expanded exhaust gas of the hot primary flow is mixed with the relatively cold secondary air flow at the afterburner entry, which results in an additional effective lowering of the temperature. This allows a more effective cooling of the afterburner and, consequently, a higher reheat temperature. Thereby a heating temperature ratio can be achieved which is about twice as high as that of the reheated turbo-jet (Fig. 1 bottom).

Besides this typical reheated turbo-fan, engine lay-outs having extremely low bypass ratios have been developed in recent years (e.g. SNECMA - M 53; GE- J 101). Their small bypass airflow is primarily used as a cooling layer for the afterburner. This type of engine, which may be considered a compromise between the turbo-fan and turbo-jet, has been designed to combine the high reheat temperatures achievable in the turbo-fan with the simple and less expensive turbo-jet concept. Obviously, the result still remains a compromise.

Because of the turbo-fan thrust behavior, the thrust augmentation resulting from the reheat grows rapidly with the growth of the Mach number. Whenever, e.g. the afterburner is ignited before the transonic range, the thrust ratio is between 2 and 3. This high thrust increase allows to adapt the engine to the varied thrust requirements of a supersonic aircraft, without the power, i.e. turbine inlet temperature, having to be unduly reduced (which would adversely affect the specific fuel consumption).

When operated without reheat, the turbo-fan will, of course, have a better specific fuel consumption anyway than the corresponding turbo-jet. When applying the reheat, however, the turbo-jet will show a lower fuel consumption since it can always achieve a higher afterburner pressure ratio than a turbo-fan operating in a conventional cycle.

2.2 OBJECTIVES AND CRITERIA OF DESIGN

The basic objectives of the thermodynamic layout of the engine are: minimum specific fuel consumption, for operation without reheat, and maximum specific thrust, for operation with reheat. For the sake of clarity, secondary aspects - although affecting the configuration in many ways - will be neglected in this section.

An essential prerequisite for the optimum thermodynamic layout of the reheated turbo-fan is the adequate consideration of the interrelationships among the following parameters:

- Turbine Inlet Temperature T_{t3}
- Primary Flow Compression Ratio $\pi_{tC,I}$
- Bypass Ratio μ
- Secondary Flow Compression Ratio $\pi_{tC,II}$

For high specific thrust and low specific fuel consumption, the turbine inlet temperature T_{t3} should be close to the technological limits, both in reheat and non-reheat operation. From a purely thermodynamic point of view, the optimum inlet temperature is limited because of the growing cooling-air demand.

The corresponding optimum primary flow compression ratio $\pi_{tC,I}$ - for non-reheat operation - should be chosen - as a first approximation - with respect to the maximum internal efficiency of the primay flow cycle. In reheat operation, it should result in a maximum expansion pressure ratio at the afterburner exit. In the latter case, it is lower than the compression ratio of a corresponding reheated turbo-jet.

The bypass ratio μ is bound to be the result of a compromise. In non-reheat operation, a high μ value will result in a low specific fuel consumption. For reheat operation, the optimum bypass ratio would be as low as possible, even down to zero (corresponding to the plain turbo-jet). Moreover, aerodynamic and design restrictions prevent high bypass ratios from being achieved in a reheated turbo-fan. Last but not least, the choice of the bypass ratio is determined by the rate of the thrust augmentation-level required by the flight mission which in turn increases with increasing μ.

Based on the foregoing, the final layout parameter, namely corresponding optimum thermodynamic secondary flow compression ratio, can be found by the following simplified calculation:

$$\boxed{(\pi_{tC,II})_{opt} = \pi_{tC,I} \cdot \frac{\pi_{tMB}}{\pi_{tBy}} \cdot \left[1 - \frac{1}{1+\beta-\gamma} \cdot \frac{\bar{c}_{pC} / \bar{c}_{pT}}{\eta_m \cdot \eta_{Cs} \cdot \eta_{Ts}} \cdot \frac{1 + \frac{\bar{k}_E - 1}{2} \cdot M_\infty^2}{T_{t3}/T_\infty} \cdot \left\{ \left(\pi_{tC,I}^{\frac{\bar{k}_C - 1}{\bar{k}_C}} - 1 \right) + \mu \cdot \left(\pi_{tC,II}^{\frac{\bar{k}_C - 1}{\bar{k}_C}} - 1 \right) \right\} \right]^{\frac{\bar{k}_T}{\bar{k}_T - 1}}}$$

simplifications: $\bar{c}_{pC,I} \approx \bar{c}_{pC,II} = \bar{c}_{pC}$ *und* $\eta_{Cs,I} \approx \eta_{Cs,II} = \eta_{Cs}$

Figure 2 shows an evaluation of this calculation.

In order to fulfill the optimum conditions, and after meeting the needs of primary flow compression, only a gradually diminishing portion of the energy flow available for thrust generation may be extracted, as the Mach number increases, from the hot primary flow and transferred to the secondary one. Consequently, the optimum bypass pressure ratio will drop rapidly, if the bypass ratio μ is kept constant.

For a real jet-mixing turbo-fan, the resulting curve is bound to approximately follow the course shown. Normally however, the bypass ratio μ increases with increasing flight Mach numbers so that the energy-extraction rises far beyond the optimum value. Ideally, the bypass ratio should continually decrease down to zero with increasing flight Mach numbers.

2.3 ASPECTS OF STEADY-STATE BEHAVIOR

Since an air-breathing turbo-engine can generally not operate continually at its design point, its steady-state behavior must be already taken into consideration at the design stage.

The steady-state behavior of a turbo-fan depends largely on the design concept. In a non-jet-mixing turbo-fan designed for high subsonic missions with a high bypass ratio, there are many reasons for a three-spool concept. For the reheated turbo-fan designed for the flight mission described before, this concept shows only minor advantages over the twin-spool concept because of the low bypass ratio of the engine. The steady-state behavior of the latter, under changed inlet flow conditions, will be shown in the following sections, using the comparison of a suggested twin-spool alternative (Ref. 1) with the three-spool concept of the RB-199 as an example.

Fig. 3 shows the schematics of this type of engine and the corresponding compressor characteristics. The compressor consists of a low-, intermediate-, and high-pressure part, the low and intermediate ones being installed in one combined spool. Because the bypass ratio $\mu \approx 1.25$ is low, this design still allows adequate tip speeds in the intermediate pressure compressor.

For the basic control of the engine, a great variety of control laws are conceivable. The operating lines for the control with a constant nozzle area A_5, and constant turbine inlet temperature T_{t3}, are entered in the three characteristics (the control with constant A_5 and constant HP-rotor speed provides similar curves).

The curves shown apply to reheat operation, with the reheat temperature T_{t4b} kept constant. Almost the same operating lines result from non-reheat operation.

Fig. 4 shows how the values for the effective rotor speeds N_{LP} and N_{HP}, and for the by-pass ratio μ, change as the flight Mach numbers increase.

2.4 NATURAL ADAPTABILITY OF THE CONVENTIONAL REHEATED TURBO-FAN

In this section, the layout of a conventional reheated turbo-fan for a supersonic aircraft will be used as an example to show the natural adaptability of this turbo-fan to different thrust requirements. Its performance will be compared with that of a reheated turbo-jet designed for the same mission. For the sake of simplification, the two parameters essential for assessing the suitability of the engines were assumed to be as follows: high supersonic cruise M_{∞}=0.85 to 0.95 and acceleration to maximum design Mach number M_{∞} = 2.5.

In order to obtain clear and smooth curves in the following graphs, the demonstration is based on the altitude /Mach-number schedule shown in Fig. 5. This schedule, which, for an optimum three-shock inlet, has a constant value of p_{t1}/T_{t1}, allocates a suitable flight altitude to each flight Mach number so that it leads to more realistic results than an investigation conducted for a constant flight altitude would.

In Fig. 6, the drag distribution of the aircraft considered (e.g. a supersonic transport) is entered over the flight Mach number. In this, each of the values are related to the design flight Mach number drag. The course of the curve is characterized by the steep rise of the drag at the beginning of the transonic range.

The twin-spool reheated turbo-fan described before will show the thrust course entered, if its maximum available thrust is matched to the aircraft drag at the design Mach number. Taken into account as an additional restriction for the calculation was the fact that, for reasons of cooling the turbine and afterburner, the turbine inlet temperature and the reheat temperature must be reduced starting at M_{∞} = 1.75 and M_{∞} = 2.2, respectively.

As the curve for the maximum thrust provided by the non-reheated turbo-fan shows, the power setting need hardly be reduced to adapt the engine thrust to the aircraft drag, even under stationary subsonic-cruise conditions. For comparison, the thrust curve for a single spool turbo-jet designed for the same flight mission has been entered for both, reheat and non -reheat operation (dashed curves). Because this type of engine allows much less thrust augmentation, the turbo-jet has to be run at greatly reduced and uneconomical partial-power for stationary subsonic cruise.

Fig. 7 shows for both engine concepts the specific fuel consumptions, and reciprocal values $F/\dot{m}_f$, which result from maximum power operation. While the turbo-fan markedly surpasses the turbo-jet engine in non-reheat operation, it shows a high specific fuel consumption when used in reheat operation. Yet this can only be partly attributed to the reasons mentioned before. Besides, the high nozzle pressure ratio the turbo-jet engine has the "Thermodynamic advantage" of a lower possible reheat temperature which, with T_{t4b} = 1800 K, is about 300 degrees below the value for the turbo-fan engine.

3. POSSIBLE TRENDS IN THE DEVELOPMENT OF THE REHEATED TURBO-FAN

In the following sections possibilities for further increasing the adaptability of the reheated turbo-fan engine will be investigated.

3.1 LIMITS AND OBJECTIVES IN DEVELOPING CONVENTIONAL REHEATED TURBO-FAN ENGINES

Before setting the objectives for desired improvements, it seems practicable to consider the essential limits to improving the adaptability of the conventional reheated turbo-fan engine. These limits are found in the heating temperatures achievable in the main combustor and afterburner.

The limitations of the turbine inlet temperature and, consequently, the compression ratio and airflow resulting from admissible limits of the main combustor pressure in ground-level high-speed flight are irrelevant for civil air traffic. This also applies to the temperature limit resulting from the compressor aerodynamics in high-altitude low-speed flight.

The reduction of the turbine inlet temperature necessary in the upper flight-Mach-number range, however, results in a marked thrust restriction. The reduction of the rotor speeds - especially those of the fan - not only reduces the inlet airflow but also leads to an undesired rise of the bypass ratio. As mentioned before, the latter increases anyway with increased flight Mach numbers,an increase that defies the objective of optimum adaptability.

The reduction of the turbine inlet temperature becomes necessary as the efficiency of turbine cooling decreases with the increasing flight Mach number. This is related to the temperature rise of the cooling air extracted at the compressor exit.

The thrust is further restricted because the reheat temperature in the afterburner must also be reduced in the high flight-Mach-number range. The temperatures of the secondary and outer bypass airflows in the engine tunnel, which cool the inside and outside of the afterburner, also increase with increasing flight Mach numbers. Relatively irrelevant, however, is the limitation of the reheat temperature T_{t4b} for extremely low combustor entry temperatures (high-altitude low-speed flight). It is caused by the thermodynamic blockage of the afterburner with its negative effects on the turbo engine.

Based on the foregoing, the following objectives can be formulated for the improvement of the adaptability of the turbo-fan engine:

- Increase of the thrust augmentation by reheating (for both, the entire reheat-thrust phase and short term operation)

- Improvement of the thrust decline gradient with increasing flight Mach number M_∞ .

- Achieving a high bypass ratio μ in unreheated operation and a lower μ with increasing M_∞ and increasing reheat.

- Achieving a general improvement of the specific fuel consumption and thrust/weight ratio of the engine.

The following sections will critically investigate a number of unconventional approaches to, and the feasibility of, influencing the aerothermodynamic cycle of the type of engine considered, and whether they offer a chance of improvement in the desired direction.

3.2 FUEL-RICH COMBUSTION

Fuel-rich reheating is a fairly simple method to achieve an effective augmentation of the afterburner thrust. With unchanged reheat temperature and, depending on the flight Mach number, fuel-rich combustion will increase the thrust by about 1o to 2o per cent (Fig. 8; one-dimensional calculation). The specific fuel consumption which simultaneously increases by about 5o per cent, makes this method of thrust augmentation only suitable for short -time operation. In other words it is an interesting concept for fighter aircraft only. It would require the installation of additional burner manifolds and the modification of the cooling air distribution at the afterburner heat shield, as well as the re-adaptation of the thrust nozzle and afterburner controls.

Another way to increase the afterburner thrust, and simultaneously improve the specific fuel consumption, would be to use fuel-rich combustion in the main combustor:
It has been shown that, for reasons of thermodynamics and reliability (LPC-surge), a bypass ratio exceeding values of $\mu = 1.5 \div 2$ is unsuitable for conventional afterburner operation. On the other hand, higher bypass ratios would allow the entire fuel flow needed for reheating to be fed already into the primary cycle air flow in the main combustor before the turbine. Thus, fuel-rich combustion can be used to limit the turbine inlet temperature to its technologically admissible maximum value. The fuel flow required for this purpose will be available from bypass ratios of $\mu = 2$ on, if present-day values are assumed for the turbine inlet and reheat temperatures.

The supply of the additional fuel, compressed in its liquid state, upstream of the turbine causes more than 15 per cent of increase ($\mu = 3$) in mass flow. This has the effect that the specific enthalpy difference in the turbine can decrease while the compression effort remains the same. Since in addition, in fuel-rich combustion, the specific heat of the combustion gas shows markedly higher values than normally, the temperature and pressure in the turbine will drop much less.

This allows to take a greater share of energy from the hot primary flow: The bypass ratio can be increased if the compression ratio in the secondary air flow is increased simultaneously. The higher secondary flow compression ratio will result in a higher expansion pressure ratio in the thrust nozzle and, consequently, in an increase of both, the specific and actual thrusts, as well as in an improved specific fuel consumption. The higher bypass ratio allows an increased thrust augmentation by reheating.

Fig. 9 shows the schematics of this engine modification and its corresponding compressor characteristics. The intermediate-pressure compressor attached to the LP-spool had to be omitted in this case. Steady-state behavior greatly changes in transition from fuel-rich combustion in the main combustor to the usual under-stoichiometrich combustion which continues being required in the non-reheated case. For this reason, it is of importance to separate the aerodynamic connection - existing via the attached IPC - between the high -pressure compressor and the low-pressure rotor.

If the fact that, in this engine concept the optimum compression ratio for the primary flow is much higher than usual, were taken into account, this would consequently result in a three-spool design with separate spools for the LPC and the IPC. To allow a comparison, however, the results for the twin-spool design were also determined. For this simple engine concept, the compression ratio in the primary flow had to be limited to low design values ($\mu = 3$; $\pi_{tC,II} = 3.8$; $\pi_{tC,I} = 23$; $T_{t3} = 165o$ K). A comparison with Fig. 3 will indicate the weight advantage of this concept which, because of its higher bypass ratio, has a comparatively small core engine. Since the fan-pressure ratio is higher, the afterburner will also have smaller dimensions.

Again it is assumed that the engine is controlled in accordance with the "A_5=constant and T_{t3}=constant"-control law. As the flight Mach number range covered by this engine concept is very large, its operating lines extend far down into the intermediate RPM-range. The maximum flight Mach number has been set at M_∞ = 3.5.

The short operating lines have been entered for non-reheat operation as required for subsonic flight.

Fig. 10 shows versus flight Mach number the aircraft-drag, the turbo-fan thrust characteristics and, for comparison, those of the turbo-jet. Each of the values entered is related to the value at the matching-point Mach number M_∞ = 3.5. Also indicated for comparison is the thrust curve for the conventional reheated turbo-fan that will result if both turbo-fans receive the same total air flow in their design points (dot-dash curve).

Although the reheat temperatures for both, the turbo-jet and turbo-fan, are equal at the matching-point Mach number M_∞ = 3.5, the turbo-jet engine had to be dimensioned with a frontal area about 25% larger than that of the turbo-fan in order to provide the required thrust at this flight Mach number. This requires accepting a further considerable weight increase in excess of a normal turbo-jet weight over that of a turbo-fan engine at the same airflow. In stationary subsonic flight, the turbo-jet engine has to be run at low and uneconomical partial power.

Fig. 11 finally, shows the fuel consumption behavior of the engines versus the flight Mach number. Despite its higher bypass ratio, the fuel consumption of the reheated turbo-fan with fuel-rich combustion in the main combustor approaches the more favorable values of the turbo-jet. The curves of the two turbo-fan engines would match that of the turbo-jet even better, if their reheat temperatures remained limited to the lower admissible value of that of the turbo-jet.

3.3 UNCONVENTIONAL TURBINE COOLING

If the chemical reaction in the main combustor is achieved by fuel-rich combustion and if the turbine blading is conventionally cooled with compressor air, this will have a favorable thermodynamic effect: After the cooling air has flowed out from the blading, e.g. through the trailing edge of the guide vane, the oxygen contained in it will react with the fuel-rich combustion gas. The heat freed by this reaction increases the flow energy of the fluid with a better thermodynamic efficiency than the later reaction occurring in the afterburner at a lower pressure level.

The expansion-plus-heat-supply in the turbine decisively influences the steady-state behavior of the turbo engine. New technological and design problems also have to be faced. Since, for all practical purposes, fuel-rich combustion is only practiced at the high temperature level of maximum power operation, it ought indeed to be possible to stabilize a relatively uniform combustion within the turbine.

The following considerations indicate that the effectiveness of the turbine cooling may be fairly easily improved with a simultaneous saving of cooling air:
As the temperature T_{t1} rises at the compressor inlet, e.g. with an increasing flight Mach number, the temperature at the compressor exit (T_{t2}) also rises and with it that of the cooling air (Fig. 12). On the other hand, the cooling air ratio $\gamma = \dot{m}_{ca}/\dot{m}_{a,I}$ - most of which is extracted at the compressor exit - retains an approximately constant value. The cooling effectiveness is reduced. If the turbine inlet temperature remains constant, the material temperatures of the turbine blading and discs will increase. For this reason, the cooling air ratio must be designed for the maximum cooling air temperature occurring, or the heating temperature T_{t3} before the turbine must be reduced starting at a selected value T_{t2}. The latter is the accepted practice.

In Fig. 13 the cooling air requirements have been entered over the respective flight Mach numbers. The dashed line shows a cooling air ratio established as a compromise. At all low cooling air temperatures, and for the low heating temperature T_{t3} of partial power operation, this cooling air ratio is too high. This unnecessarily burdens the thermodynamic cycle and restricts the possible energy supply in non-reheat operation.

It is conceivable to adapt the cooling air ratio to the amount of air required during the main flight mode within the planned mixed mission. At all higher heating temperatures T_{t3} or compressor exit temperatures T_{t2}, as much water is injected into the cooling air flow as is needed for the desired cooling effect in each particular case. The high air temperature T_{t2}, which amounts to about 800 K in modern turbo-fan engines, causes the water to evaporate immediately after injection so that the cooling air flow loses heat. At the same time, the heat transfer on the cooling side is improved by the higher humidity of the air.

By adequately optimizing the cooling air ratio, the total quantity of water needed for a mission can be so defined that it is compensated by the fuel savings achieved. The advantage of this cooling method is that it allows to operate the engine even at high flight Mach numbers with full power RPM (HP rotor), as far as the compressor and main combustor permit this. The increased air flow and afterburner pressure ratio, which results from the decreasing bypass ratio μ , provide an increased afterburner thrust and improved specific fuel consumption.

The concept of direct cooling - corresponding to a kind of water cooling - of the turbine blading by the fuel flow or by a very rich fuel-air mixture (as may result, e.g. from the vaporization of the afterburner fuel in a small carrier air flow) must be abandoned because of the danger of explosion involved. Yet it would be consequent to transfer the cooling capacity of the fuel flow to the cooling air flow indirectly and without any mixing of the flows:
In a high-pressure heat exchanger, the turbine cooling air extracted at the compressor exit is intercooled, by ' exchange, with the reheat fuel flow. Within the fuel cycle, the heat exchanger (whose dimensions have been kept small because of a deliberately low exchange effectiveness) is installed only beyond the afterburner controls. The latter is practically not affected at all, because it is unnecessary to evaporate the fuel in the heat exchanger. If such cooling-air intercooling is applied to a reheated turbo-fan engine with a medium bypass ratio, e.g. RB-199, the cooling and leakage air flow needed for the turbine may be reduced by some 35 ÷ 40 per cent. At the same time, it will be no longer necessary to reduce the turbine inlet temperature in the higher flight-Mach-number range. The consequences of such a modification for the entire engine will be:

- In non-reheat operation both, the turbine inlet temperature and - to a lower degree - the rotor speeds will have to be reduced. This will also reduce the maximum dry thrust depending on the compressor inlet temperature - by about 5 per cent on the average, while the specific fuel consumption will increase but little.

- Despite the higher turbine inlet temperatures in reheat operation, the blading temperatures in the lower and intermediate T_{t1} range will remain below the wall temperatures of the corresponding maximum unaugmented power operation. In the upper flight-Mach-number range, T_{t3} need no longer be reduced, or may be reduced much less, depending on the engine layout.

- Whereas for the RB-199, the bypass ratios μ are the same for unaugmented power and afterburner operation, μ is reduced as the afterburner is ignited (higher rotor speeds resulting from higher turbine inlet temperatures). This effect is desirable in principle for the reasons mentioned in Section 2.2.

- The marked increase of μ, resulting from T_{t3} reduction in the upper flight-Mach-number range and involving thermodynamic disadvantages and fluidmechanical problems (splitter flow, increased diffusion in the hubregion before the intermediate pressure compressor) will be effectively reduced. Since the superimposed reduction of the fan speed connected with the T_{t3} reduction no longer exists, an increased inlet air flow is caused in many important flight modes, which increases the maximum thrust. Beyond $M_{\infty} = 2$, the thrust increase climbs to considerable values ($>20\%$).

- The higher temperature of the fuel injected and the somewhat lower Mach number level in the afterburner assures reliable afterburner operation at a higher altitude limit. (The areas in the afterburner are the same as in the RB-199.)

- Over the entire flight Mach number range a slight improvement of the specific fuel consumption is achieved. This is due to a higher afterburner pressure ratio and an improved combustion chamber efficiency.

3.4 INTERCOOLING BEFORE ENTRY INTO THE INTERMEDIATE PRESSURE COMPRESSOR

Another practicable way to make use of the cooling capacity of the afterburner fuel flow in pursuit of the objectives defined under 3.1, would be the intercooling of the primary air flow before it enters the intermediate pressure compressor. The effects achieved resemble those described for turbine-cooling-air intercooling. They increase with increasing flight Mach numbers.

Fig. 14 shows the construction schematics of the three-spool RB-199 with its generalized compressor characteristics. Inbetween the low- and intermediate pressure compressors, a heat exchanger has been installed. It has deliberately low exchanger effectiveness to keep its dimensions and pressure losses small. In order to generate a temperature distortion, the afterburner fuel flows through the exchanger in a cross stream radially from inside to outside. The major direct consequences of this engine modification are:

- The non-reheat operation of the engine is adversely affected by the pressure loss of the heat exchanger. Some of the resulting thrust reduction might be compensated, if the fuel flow to the main combuster were conducted through the heat exchanger. However, this would cause problems in controlling the engine and heat exchanger.

- As the afterburner is ignited, fuel flows through the heat exchanger. The temperature in front of the intermediate-pressure compressor drops (Fig. 15) which results in an increase of the reduced RPM's, the mass flows, and the pressure ratios at the IPC and HPC. The bypass ratio decreases significantly while the fan speed and pressure ratio increase. In order not to exceed the design value of reduced RPM for the IPC the heat exchanger will be fed only with a partial fuel flow in the range of lowest flight Mach numbers. The growth of the air flow supplied to the engine (Fig. 16), and of the turboengine pressure ratio, increase the thrust while the specific fuel consumption is improved (Fig. 17).

- Since the compressor exit temperature and, consequently, that of the cooling air drops, a greater cooling effect is achieved in the turbine by the same cooling air flow. In the upper flight Mach number range the turbine inlet temperature and rotor speeds will have to be reduced less. For this reason, too, the bypass ratio increases less in this case, and the thrust drop is considerably modified. Another effect of the temperature reduction at the compressor exit - worth mentioning - is the extension of the flight Mach number range covered by the engine.

- The higher mass flow density at the afterburner entry pushes the afterburning limit to higher flight altitudes and improves the combustion chamber efficiency.

- The insertion of the heat exchanger makes the intermediate pressure compressor less sensitive to distortion. Pressure distortion, locally limited and/or dynamic coming from the inlet, or directly from the low-pressure compressor to the heat exchanger result in a locally limited or temporary increase of the exchanger intercooling. Superimposed on the pressure distortions are "temperature distortions", represented by greater than average temperature drops, which support the reduction of the pressure distortions in the intermediate pressure compressor. In a similar way, the deliberate radial temperature distortion takes into account the greater sensitivity against distortion of the hubregion air flow.

In the Figures 14 to 17 the example of the basically unmodified RB-199 was used to demonstrate, quantitatively, how steady-state behavior and performance parameters were influenced by the intercooling. The possible combination of turbine-cooling-air intercooling as mentioned before, was emitted. It goes without saying that an engine redesigned to incorporate these modifications would promise added advantages.

3.5 UNCONVENTIONAL FUELS

Because of their higher calorific values the use of liquid methane or hydrogen instead of the usual hydrocarbons would result in a considerable reduction of the specific fuel consumption. Additionally, their use could essentially contribute toward increasing the positive effects of the possible improvements mentioned before or toward facilitating their implementation. For fuel-rich combustion, the following specific properties of these fuels are of major importance:

- wider ignition limits,
- faster chemical reactions,
- lower molecular masses.

For cycles with intercooling, they decisively surpass the conventional hydrocarbons (Ref.2) especially because of their high usable heat capacity.

Despite their difficult handling, their use as aircraft fuels seems an extremely promising idea.

4. CONCLUDING REMARKS

It has been shown for what reasons and within what scope the reheated turbo-fan engine is suitable, in principle, to be closely adapted to the various requirements of supersonic aircraft.

After establishing the functional limits and development objectives of the engine, a number of possibilities have been discussed to give the reheated turbo-fan engine the added flexibility that would adapt it even to the demands of future high supersonic flight:

1) Fuel-rich Combustion
   - in the afterburner (short-time operation)
   - in the main-combustor
2) Unconventional Turbine Cooling
   - water injection into cooling air flow (short-time operation)
   - intercooling of turbine cooling air
3) Intercooling in the Primary Air Flow
4) Unconventional Fuels.

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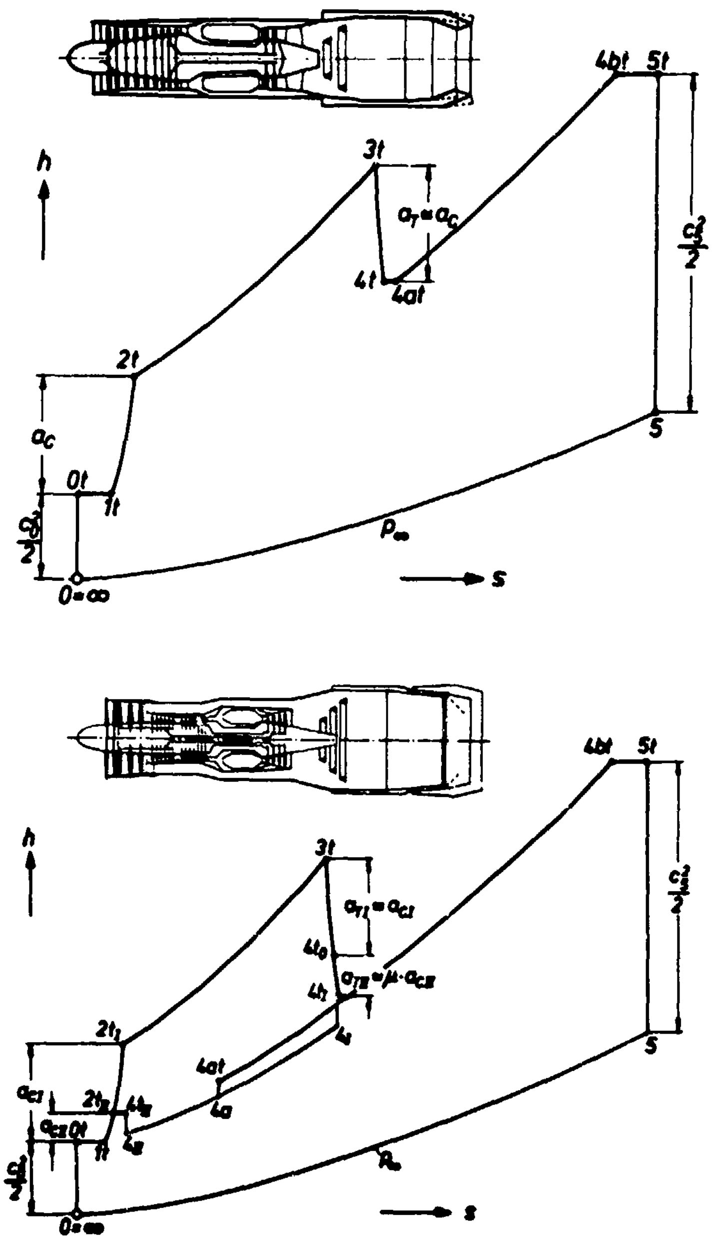

Fig. 1 Cycles of the Reheated Turbo-jet and Turbo-fan

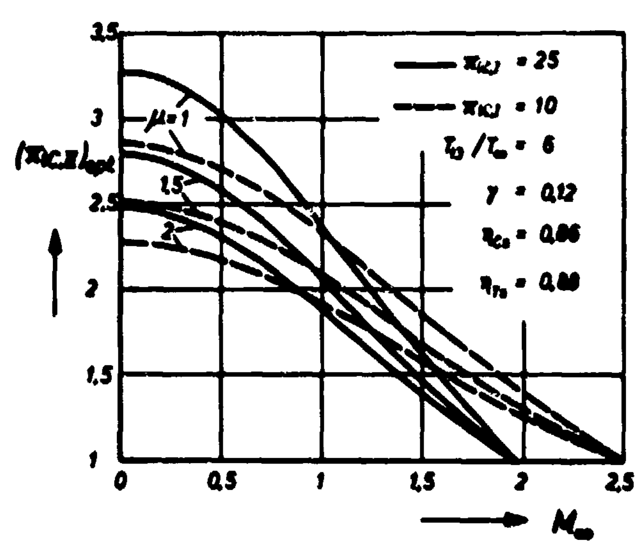

Fig. 2 Optimum Bypass Pressure Ratio

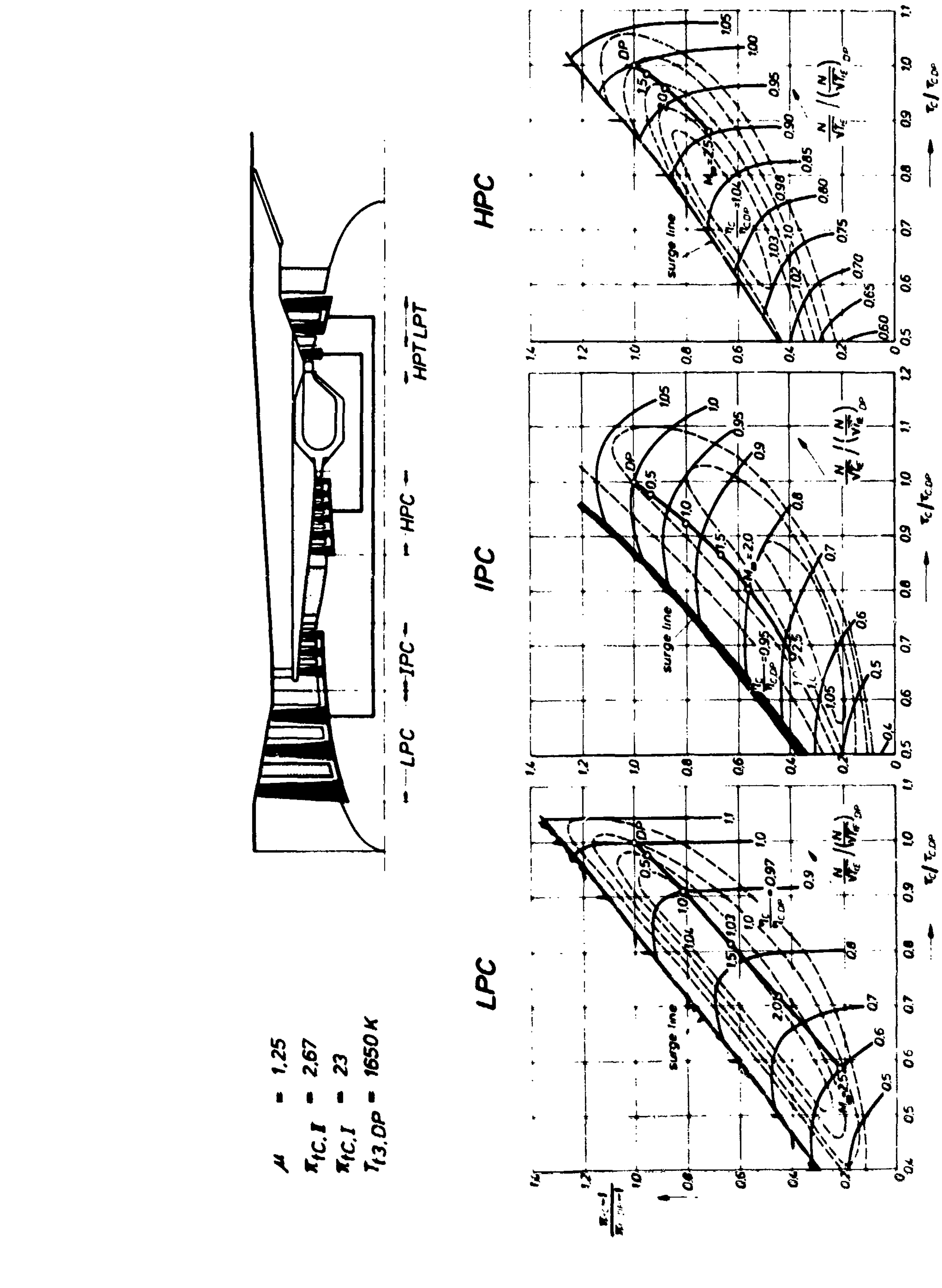

Fig. 3 Schematics and Steady-state Behavior of the Conventional Twin-spool Engine

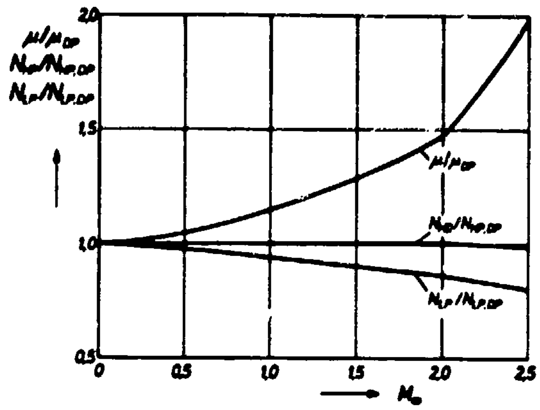

Fig. 4 Bypass Ratio and Rotor Speeds vs. Flight Mach Number

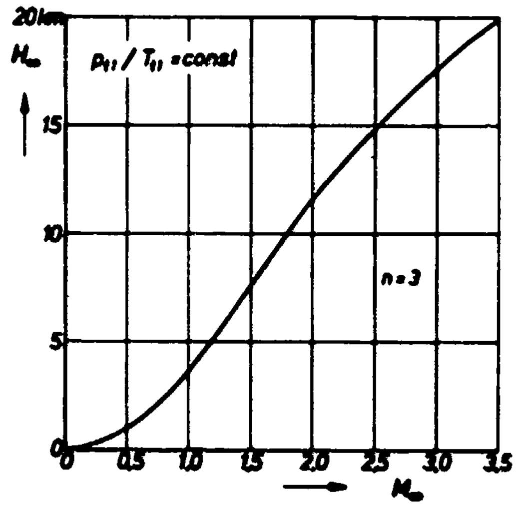

Fig. 5 Altitude/Flight Mach Number Schedule

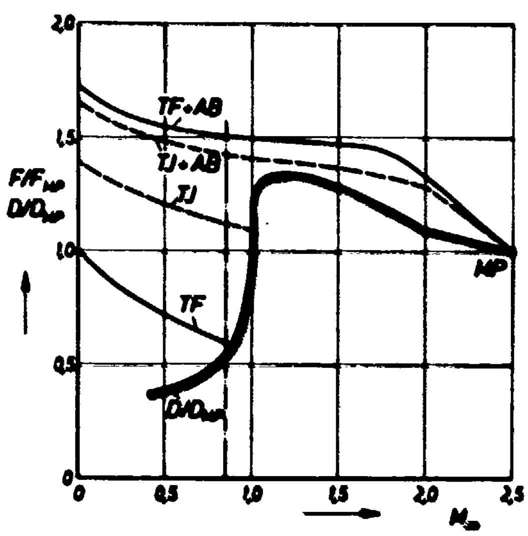

Fig. 6 Distribution of Engine Thrust and Aircraft Drag vs. Flight Machnumber Range

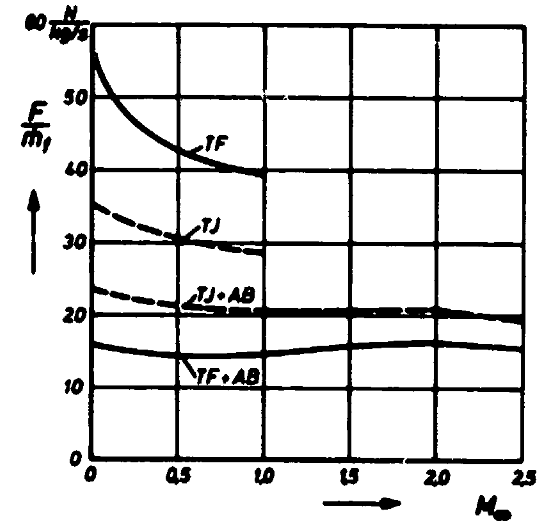

Fig. 7 Comparative Fuel Economy

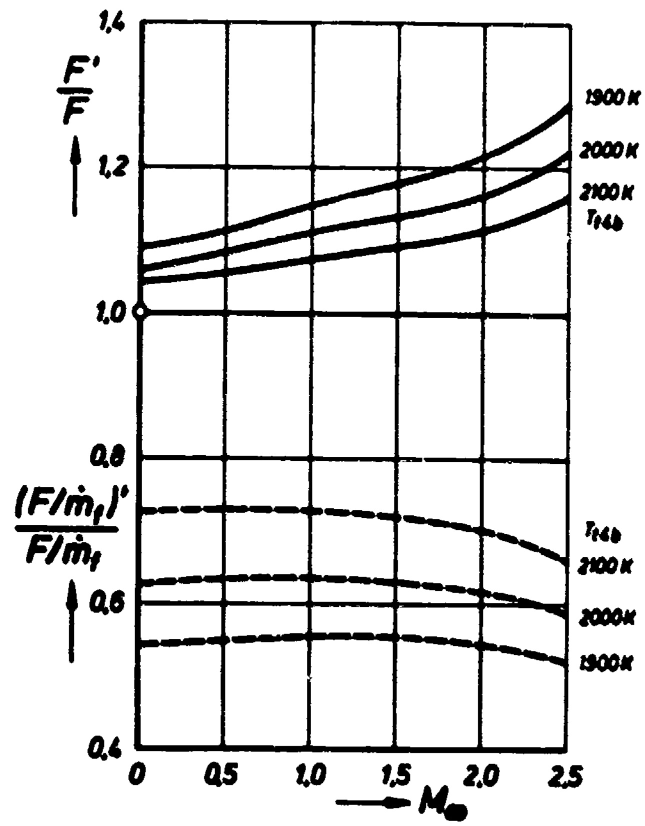

Fig. 8 Fuel-rich Reheat: Effect on Thrust and Fuel Economy

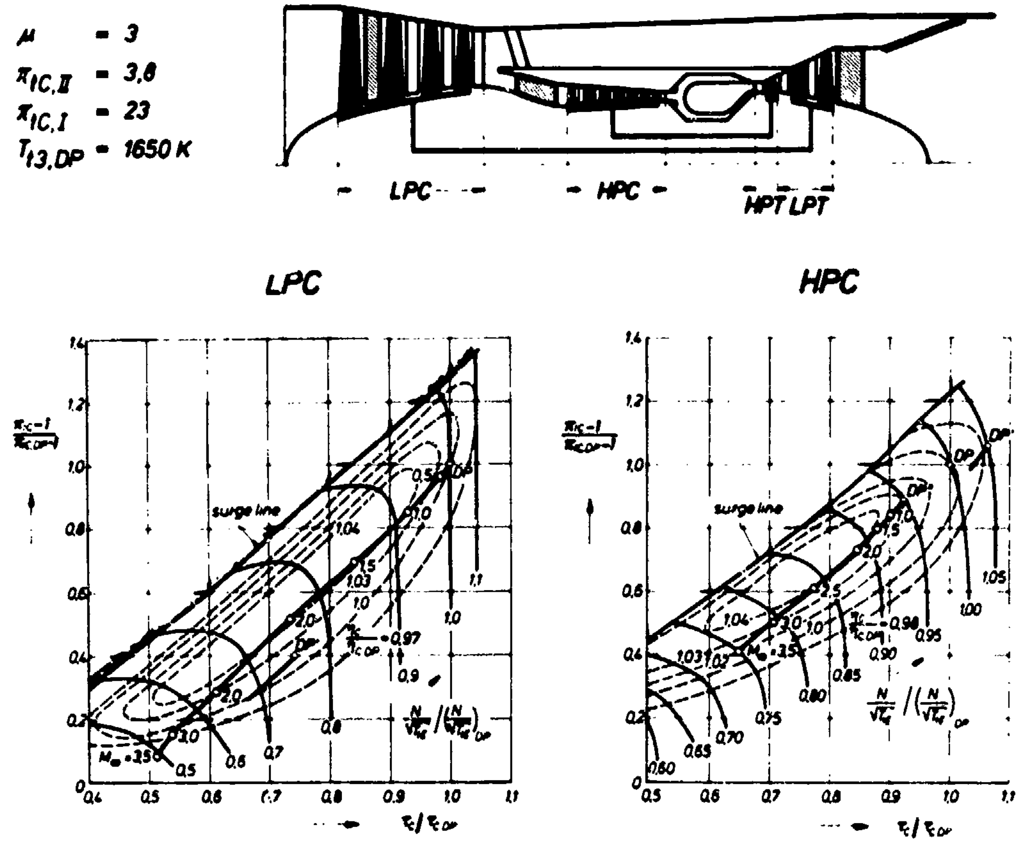

Fig. 9 Schematics and Steady-state Behavior of the Engine with Fuel-rich Combustion

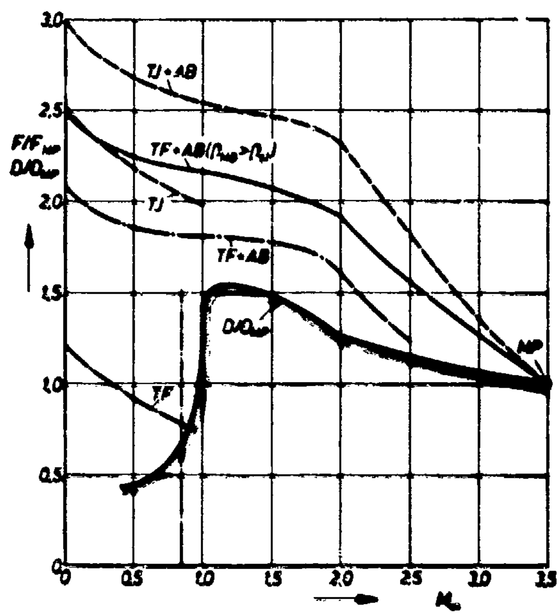

Fig. 10 Comparative Thrust Distribution

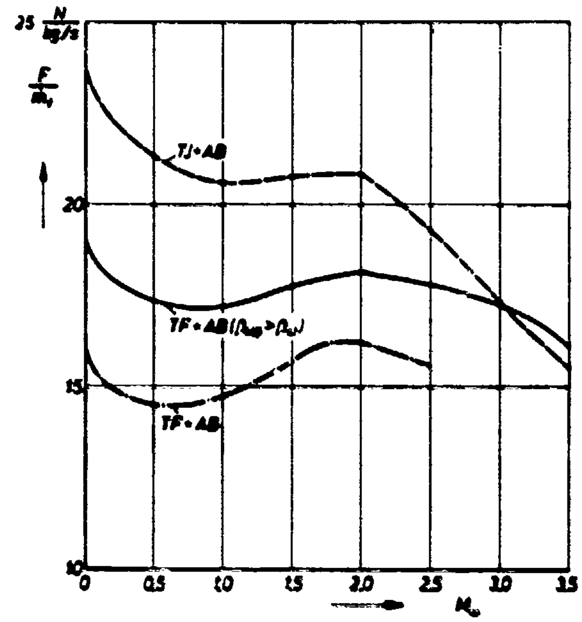

Fig. 11 Comparative Fuel Economy

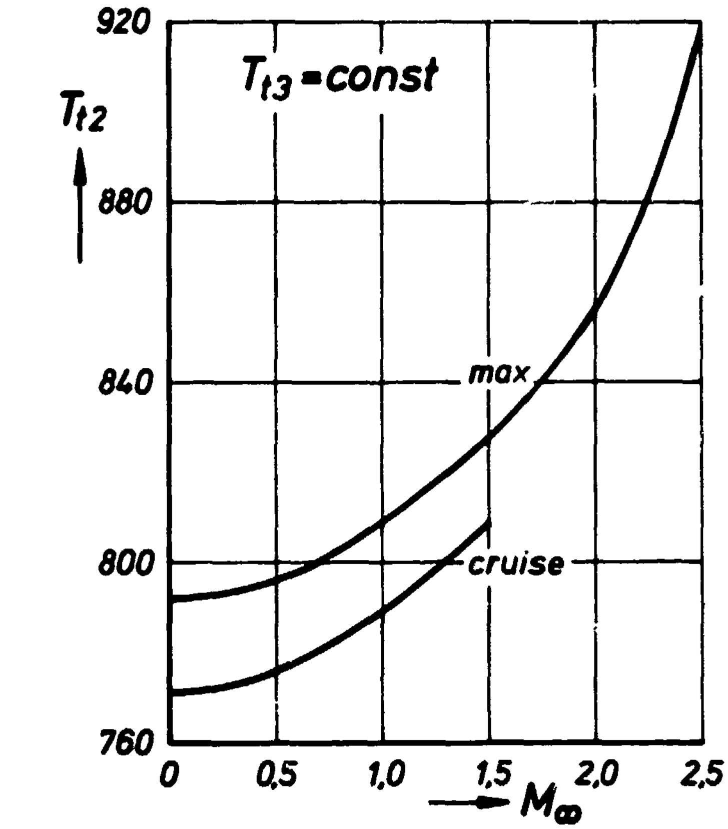

Fig. 12 Compressor Exit Temperature
vs. Flight Mach Number Range

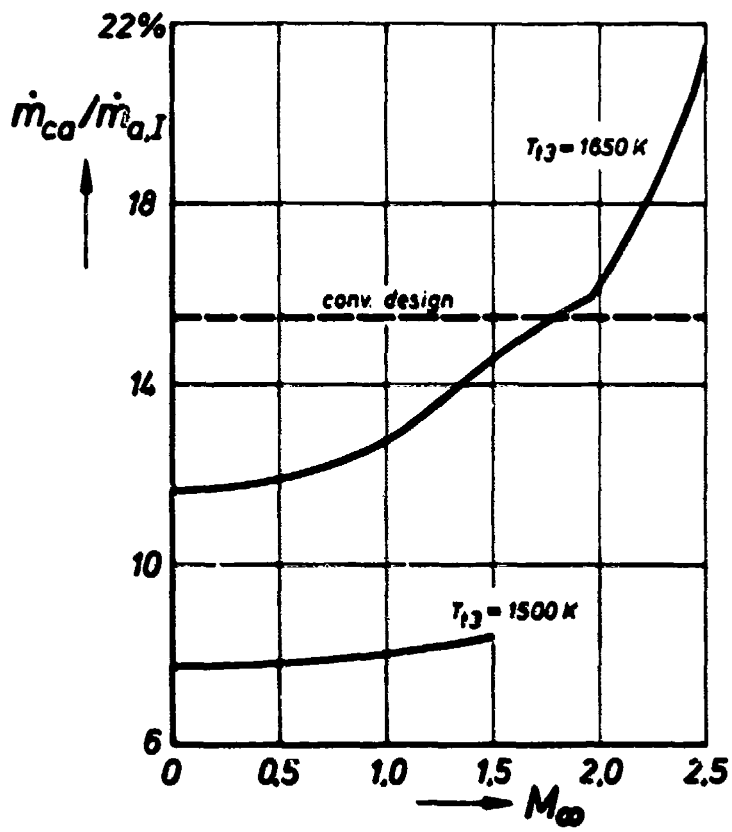

Fig. 13 Minimum Required Cooling-air Ratio

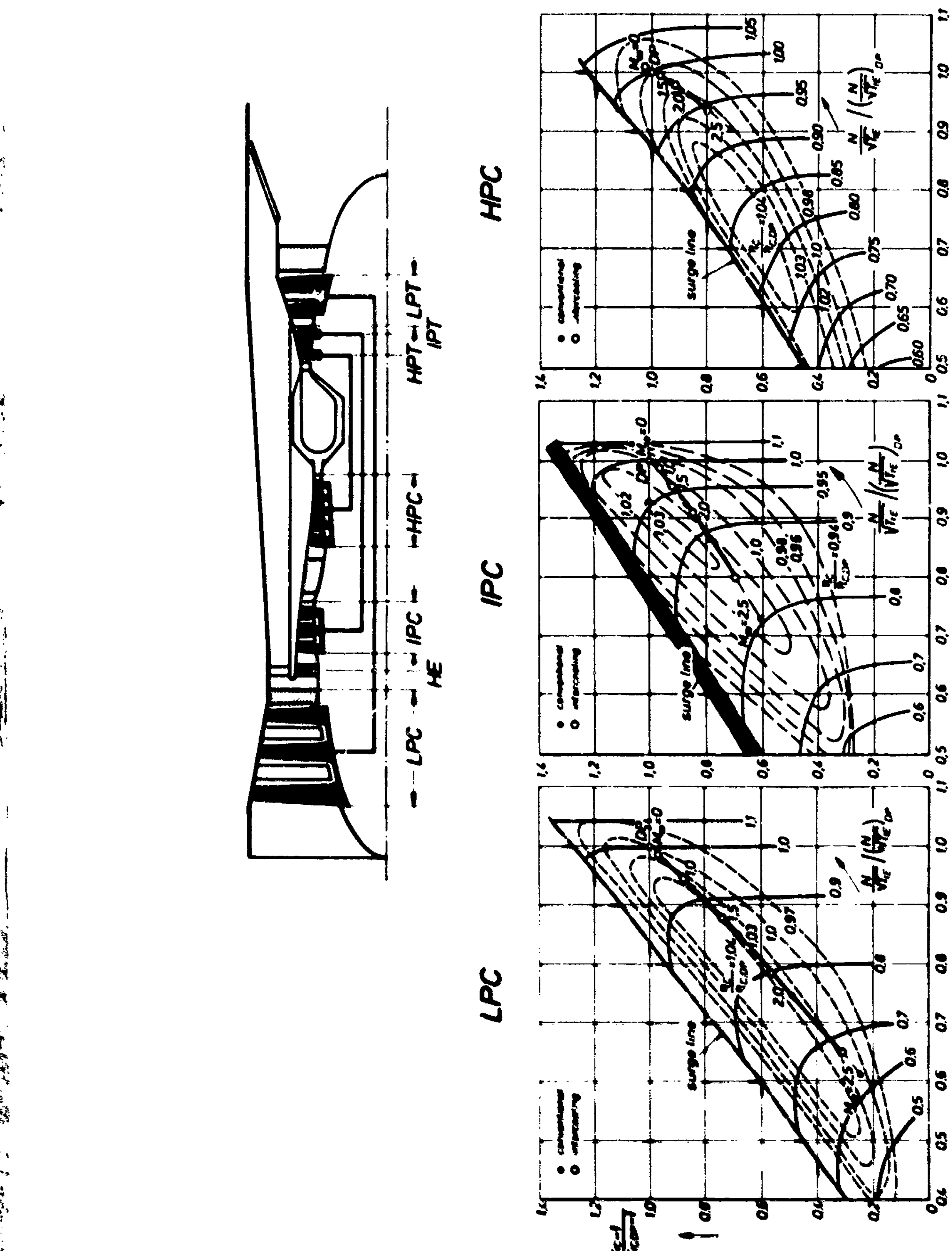

Fig. 14 Schematics and Steady-state Behavior of the Engine with intercooling of the Primary Air Flow

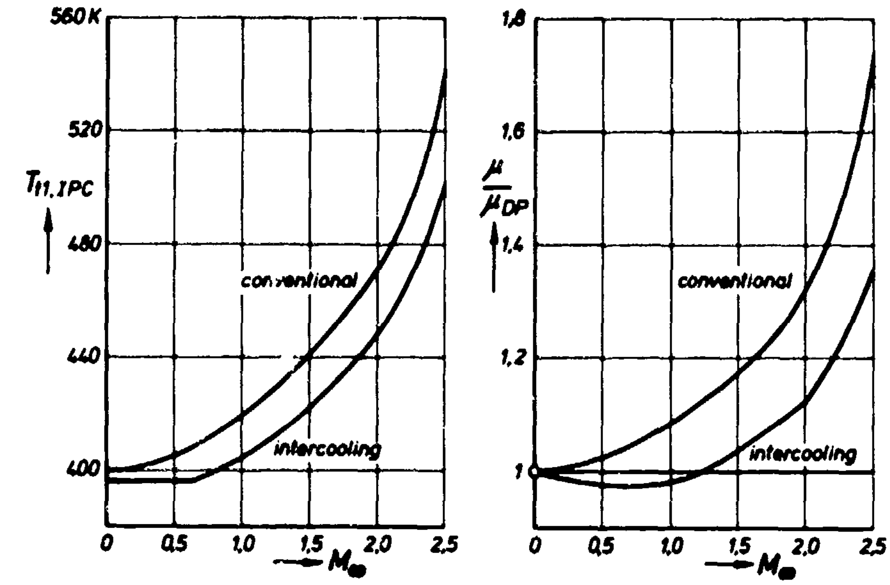

Fig. 15 Intercooling: Effect on IPC-Entry Temperature and Bypass Ratio

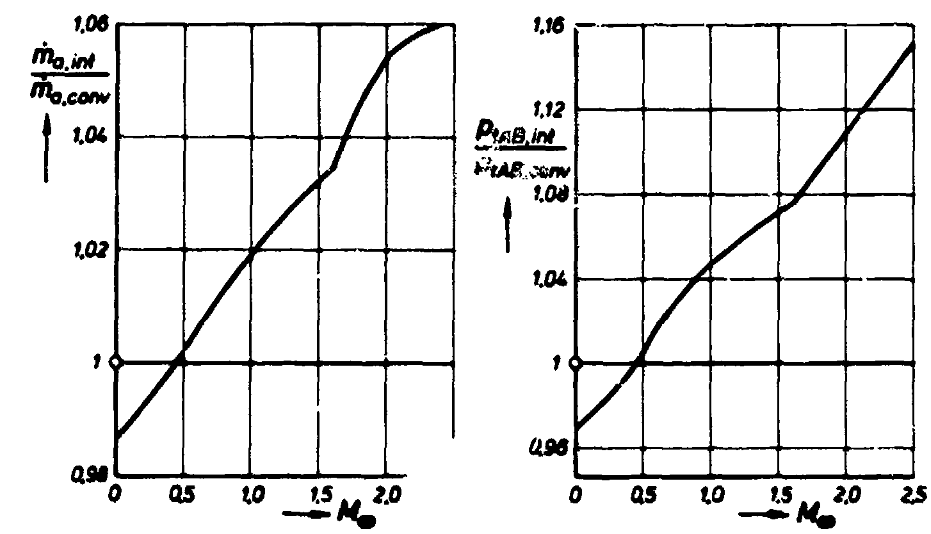

Fig. 16 Intercooling: Effect on Engine Air Flow and Afterburner Pressure

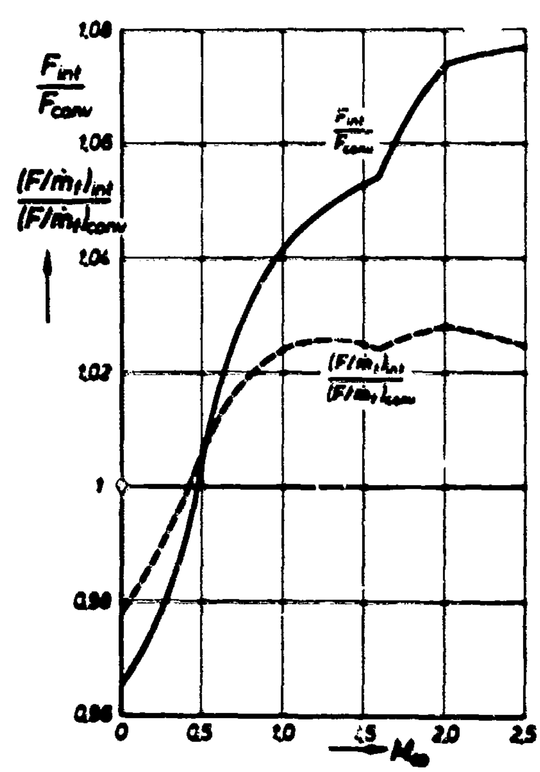

Fig. 17 Intercooling: Effect on Thrust and Fuel Economy

DISCUSSION

J.J.Kurzak

What reheat fuel temperature did you assume for the inter-cooling and what was the heat up of this fuel during cooling?

Author's Reply

The fuel temperature at the heat exchanger exit depends on the flight Mach number. In this case it will be about 100°C at Mach one and about 145°C at Mach two.

J.J.Kurzak

So you assumed somewhat like 60–70° fuel temperature supply?

Author's Reply

The fuel temperature at the heat exchange. . ıtrance was assumed to be about 50°C.

J.J.Kurzak

Do you think that is realistic in view of the MRCA?

Author's Reply

The temperaıure of the main combustor fuel will be somewhat higher because this has been used previously for inter-cooling of the oil, etc. But it is not necessary to use the reheat fuel for this purpose too.

H.Ahrendt

Can you comment how the fuel rich reheat works when the after burner fuel is introduced in the main combustor, Have you a practical solution in mind?

Author's Reply

In the main combustor there is usually a nearly stoichimetric mixture ratio in the primary combustion zone. Normally you supply surplus air to lower the turbine inlet temperature. In the case of fuel rich combustion however you must add supplementary fuel for the same purpose, thus reaching mixture ratios well beyond the stoichiometric

J.Dunham

In your performance calculations, did you account for the spillage drag of the supersonic intake, and the base drag of the supersonic nozzles, under conditions of reduced engine airflow? So is your "thrust" F *installed* thrust or engine *thrust*?

Author's Reply

All thrust curves are computed with respect to the spillage drag, not with respect to the exhaust system drag.

AUGMENTED DEFLECTOR EXHAUST NOZZLE (ADEN) DESIGN FOR HIGH PERFORMANCE FIGHTERS

by

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Advanced Development Division
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ABSTRACT

A two-dimensional Augmented Deflector Exhaust Nozzle (ADEN) has been designed, fabricated and tested for use on future advanced multimission V/STOL aircraft. The ADEN program was part of a Navy Advanced V/STOL Propulsion Component Development Program. This program included a comprehensive series of aerodynamic and mechanical design studies, aircraft system studies, scale model experimental test programs, and a full size ADEN demonstration test on a General Electric YJ101 engine. This paper presents the aerodynamic and mechanical features of the ADEN and describes the full scale demonstrator program.

1. INTRODUCTION

The United States Navy, recognizing the advanced technology requirements to develop a superior multimission V/STOL fighter aircraft, initiated an Advanced V/STOL Propulsion Component Development Program. In mid-1972 the General Electric Company was awarded the exhaust nozzle/deflector portion of the program.

The Nozzle/Deflector Component Development Program consisted of three parts: (1) was a comprehensive evaluation of the performance, design and installation parameters required for the identification of a nozzle/deflector system having the greatest potential for application to a Navy V/STOL fighter; (2) included the detail design and component testing of the selected nozzle/deflector; (3) was the design fabrication and test of the full size engine demonstrator assembly of the ADEN.

2. V/STOL PROPULSION SYSTEM REQUIREMENTS

Future V/STOL fighters require high specific thrust (afterburning)engines to best achieve multimission objectives. Results of aircraft system studies conducted in Part (1) indicate that the combat and P_S mission points size the main propulsion system such that all the required vertical thrust is potentially available from the cruise engines. From a total system weight standpoint, it is desirable to maximize the amount of onboard cruise engine thrust and minimize direct lift engine/devices size and thrust requirements during vertical operation. The capability to operate the cruise engine at afterburning power settings in the vertical and cruise mode is therefore a desirable feature of an advanced nozzle/deflector system.

Advanced V/STOL aircraft configured to utilize the maximum amount of cruise engine thrust during vertical operation tend towards mid-fuselage propulsion system installation to meet pitch balance constraints. Studies have indicated that two-dimensional exhaust systems generally have superior cruise installation characteristics relative to conventional axisymmetric exhaust systems in this type aircraft configuration.

Advanced V/STOL fighter aircraft normally include multimission objectives for transonic/supersonic cruise and combat and subsonic loiter. The nozzle/deflector exhaust system internal flowpath design is critically important to a multimission V/STOL fighter because of dual requirements for both low speed and high Mach capabilities with high installed performance. As subsonic and supersonic combat specific power (excess thrust) levels increase, variable geometry exhaust systems are required to provide nozzle throat area modulation and expansion area control to provide high nozzle thrust coefficients for a wide range of cruise/acceleration and maneuverability points. It is therefore a critical requirement that the thrust vectoring device does not compromise the exhaust system aerodynamic flowpath during forward mode operation.

3. AUGMENTED DEFLECTOR EXHAUST NOZZLE (ADEN)

Exhaust systems capable of deflecting or turning the exhaust gas of jet engines to achieve vertical and short take off and landing have been considered for many years. A wide variety of thrust vectoring nozzles has evolved. In general these nozzles have had one or more of the following limitations.

- o Low internal performance due to internal flowpath compromises required to accommodate the deflector system.
- o Limited to non-augmented temperatures in the vectored mode of operation.
- o Excessive downward projection of the nozzle/deflector in the lift mode resulting in ground clearance problems.
- o Slow vector angle and nozzle area rate of variation.
- o Discontinuous vectoring performance between cruise mode and lift mode.
- o Airframe doors required to accommodate the nozzle in the vectored mode.

- o Installed performance problems due to excessive base area or nozzle boattail angles.
- o Low lift thrust available when compared to the required weight addition to the basic cruise engine.
- o Excessive complexity with consequent reliability limitations.

The Augmented Deflector Exhaust Nozzle (ADEN) is a simple, reliable, compact, high performance V/STOL nozzle which resolves all of the above listed limitations.

The ADEN (Figure 1) is a two-dimensional, variable area external expansion exhaust system. Basic components consist of: (1) a transition casing from a round cross section at the tail pipe connect flange to a two-dimensional cross section at the nozzle throat station; (2) a two-dimensional variable geometry convergent-divergent flap assembly; (3) a two-dimensional variable ventral flap; (4) a two-dimensional external expansion ramp which can be fixed or variable depending on specific installation requirements and; (5) a rotating deflector for thrust vectoring. In the stowed (cruise mode) position the deflector is located outside the casing so that it does not compromise the required internal flowpath contours.

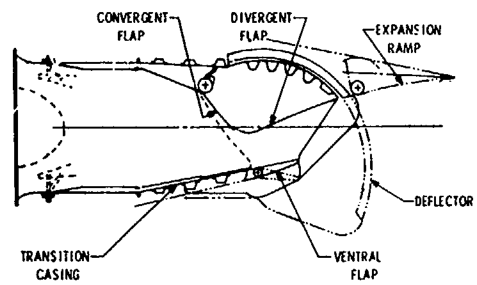


Fig. 1 ADEN Flowpath

ADEN operation is illustrated by a variable geometry model in Figure 2. In the cruise mode, nozzle area control is achieved by the variable convergent-divergent flap assembly. The variable ventral flap located downstream of the throat controls the nozzle expansion area ratio as required over the range of operating pressure ratios. (Figures 2a and 2b). Note that the throat is forward of the ventral flap such that nozzle area is independent of the ventral flap position.

For V/STOL operation, the rotating deflector diverts the jet downward (Figure 2c). The nozzle flap assembly is rotated to the maximum open position to substantially reduce the flow Mach number approaching the turn. The throat is established between the tip of the ventral flap and the deflector bonnet and rotates with the deflector such that flow turning is accomplished subsonically at all deflector settings. The low pressure region at the inside of the turn is freely vented to ambient air to: (1) minimize the secondary flowfield from the outside (high pressure) to the inside of the turn, and (2) provide a free jet expansion supersonically. These features; low velocity approaching the turn, subsonic turning and free jet expansion combine to provide a maximum efficiency thrust vectoring system.

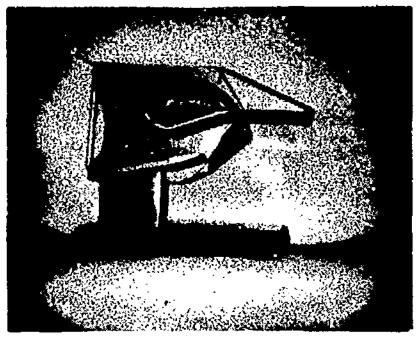
An additional feature of the ADEN design is the capability to provide inflight thrust vector control by utilizing a variable aft expansion ramp (Figure 2d). Rotation of the expansion ramp will provide an upward or downward vertical thrust component as desired.

Thrust vector control can be achieved without increasing the propulsion system frontal projected area or requiring external bomb bay doors, etc. to accommodate thrust vectoring such that benefits of inflight thrust vectoring can be realized without increasing aircraft drag.

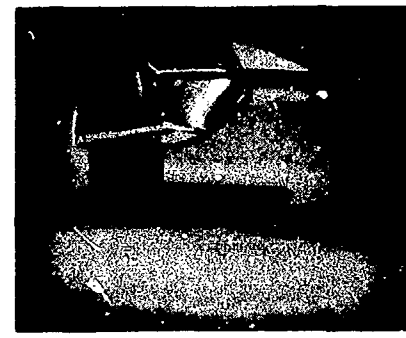
The impact of the ADEN inflight thrust vectoring capability on aircraft system performance is strongly configuration dependent. For example, in aft fuselage installations, small variations in the ADEN resultant thrust vector can significantly contribute to the aircraft pitch control authority resulting in downsizing the horizontal control surfaces and minimizing trim drag. On wing pod mounted installations, the aircraft can be designed such that the nozzle exit coincides with the wing trailing edge. In this type installation the nozzle expansion ramp can be integrated with the wing flap system and the ADEN inflight thrust vectoring capability can be utilized to produce wing lift enhancement/supercirculation effects. Figure 3 is a photograph of a twin engine/ADEN VTOL aircraft during testing in the NASA Langley low speed wind tunnel. A schematic of the ADEN installation in both cruise and lift mode is shown in Figure 4. Note that the ADEN aspect ratio (dry cruise throat width/height ratio) and the arrangement of the nozzle flaps, deflector, expansion flap, actuators and structural elements have been carefully selected to allow high performance afterbody contours. The two-dimensional nozzle shape blends well with airframe contours without drag producing base regions. The selected aspect ratio allows the ADEN to be installed without increasing frontal projected area.

3.1 Performance

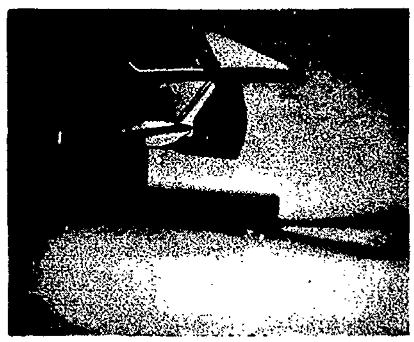
Exhaust system installed performance during forward mode operation is highly dependent on the particular aircraft/engine installation. Studies of both twin and wing pod mounted installations show that ADEN installation characteristics can provide a significant reduction of exhaust system base area, especially at dry power operation. ADEN installed drag reductions of 20 to 30% relative to conventional exhaust systems can be achieved. Wind tunnel test programs to investigate ADEN installed performance characteristics and inflight vectoring performance on an advanced multimission V/STOL aircraft were conducted at the NASA Langley, low speed V/STOL and 16-foot, high speed, windtunnels.



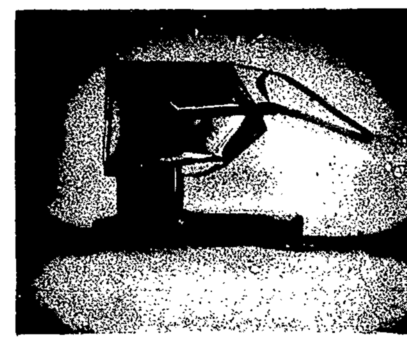
a. dry cruise



b. max A/B



c. VTOL



d. inflight thrust vectoring

Fig. 2 ADEN operation

Fig. 3 ADEN low speed wind tunnel test

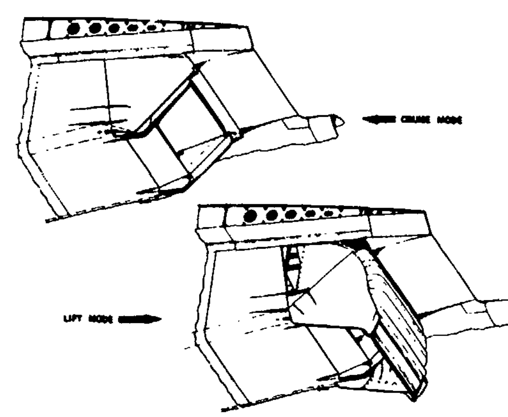


Fig. 4 ADEN installation

The cruise and vectored mode performance characteristics were verified through a series of cold flow static internal performance tests. Tests simulated forward and vectored modes over the range of operating pressure ratios and nozzle deflection angles. A comparison of the static internal performance characteristics of the ADEN is presented in Figure 5.

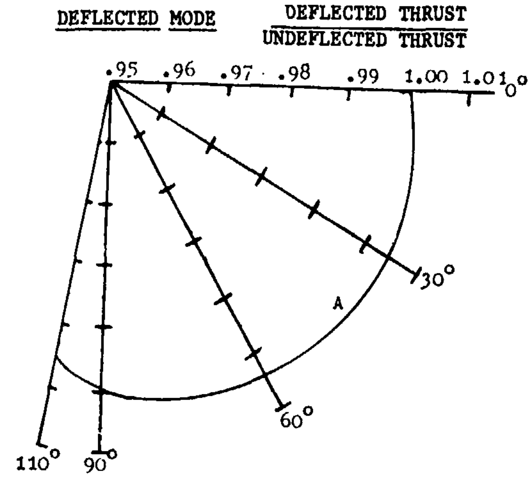


Fig. 5 Nozzle/Deflector Internal Performance Comparison

3.2 Nozzle/Deflector Installation Tests

Hover and low speed wind tunnel testing of two-dimensional and axisymmetric lift/cruise nozzle/deflectors installed in a 1/10 scale V/STOL fighter were conducted in the NASA Langley V/STOL tunnel (Figure 3). Variables tested included nozzle exit shape (round and 2-D), lift/cruise and direct lift engine jet deflection angle, ground proximity and lift/cruise engine spacing. The effects of these variables on aircraft performance (lift, drag, and pitching moment) were determined over a range of free stream to jet effective velocity ratios and angle of attack to simulate hover through transition flight conditions. The test results provided a data base to estimate in ground effect (IGE) and out of ground effect (OGE) installation characteristics of the various nozzle/deflector designs. The test program concluded that:

(1) Installation characteristics of the two-dimensional nozzle/deflector design is superior to the axisymmetric nozzle at transition flight conditions.

(2) Two-dimensional aft deflector outboard installation (approximately one nozzle diameter) is superior to the inboard installation. The pod position effect was evident both in simulated transition flight conditions and during hover. In hover, interference lift is the net result of suckdown due to jet entrainment of ambient air around the aircraft and jet/ground plane interaction producing jet fountain lift. Reduced interference lift loss at landing gear height directly impacts (reduces) the direct lift engine sizing requirements.

Diagnostic flow visualization tests were also conducted to provide a qualitative internal flow field description at various operating conditions.

3.3 Design Criteria and Requirements

The ADEN demonstrator was designed with life objectives and strength criteria similar to recent production exhaust system components. Such important considerations as thermal cyclic fatigue, acoustic fatigue, stress concentration effects, thermal stress and distortion and plastic creep were included in the analytical design along with stress and deflection analysis.

The materials and processes used are much the same as those used on current exhaust nozzles and augmentors, Figure 6. As an example, chemical milling is used to provide weight effective material distribution throughout the structure. This method of varying thickness in sheet metal structures is preferable to the use of doublers since stress concentrations at doubler attachment welds can lead to fatigue cracking. Conservative design criteria have been chosen to ensure realistic ADEN design life and weight objectives.

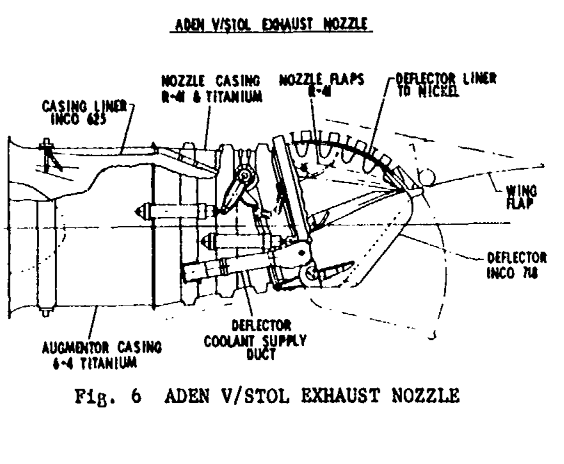


Fig. 6 ADEN V/STOL EXHAUST NOZZLE

The ADEN was designed for a low cycle fatigue life of twice the cycles anticipated based on the duty cycle. Elevated temperature parts are designed based on creep distortion limits and the objective rupture life values.

Wear and fatigue life is based on the duty cycle which includes 2667 missions of 1.5 hours each in 4000 operating hours. The vectoring actuation and augmentor ignition cycles given correspond to an average of 2.7 cycles per mission. Three nozzle actuation cycles per mission are assumed.

Maneuver load capability of 10g (downward in VTOL operation) was specified. Acoustic fatigue life analysis is based on excitation data from production engine augmentor testing.

3.4 Structural Considerations

One of the major concerns relative to two-dimensional nozzle designs has been the potential deflections due to the effects of pressure loading and thermal gradients in the flat casing and deflector sidewalls. ADEN deflections were carefully considered relative to sealing and leakage as well as general fit up and operating clearance variation of the flaps, deflector and casing. The ADEN demonstrator design has been tailored to minimize deflection without exceeding practical weight considerations. A safe level of operating clearances is maintained under all conditions.

A specialized structural analysis procedure was developed for non-axisymmetric nozzle structures. This procedure is based upon MASS (Mechanical Analysis of Space Structures) which is a three-dimensional finite element computer program.

The flat internal surfaces of a two-dimensional exhaust nozzle are subject to acoustic excitation. The ADEN design makes use of contoured and corrugated panels wherever possible to provide local stiffness and resistance to panel vibration. The root mean square alternating stress was calculated based on recent augmentor acoustic excitation test data.

The ADEN system is primarily fabricated from sheet material with formed ribs and stiffeners. Resistance welding, rivets and bolts are used for joining the sheet and ribs. This construction was selected over sandwich construction for three reasons:

(1) The sheet and rib construction is utilized extensively to distribute cooling air.

(2) Tests have shown this construction to be more thermal fatigue resistant than sandwich construction.

(3) Lower cost.

3.5 Control and Actuation

Simplicity and reliability have been emphasized throughout the ADEN control system design. The motion of the three nozzle flaps is scheduled by cam and link mechanisms such that they are operated by a single nozzle area control system. The deflector, used during V/STOL only, requires a second control system.

Highly reliable digital electronic controls will be used to control the nozzle functions. The control components, such as a hydraulic pump and actuators, are of proven design with conventional 3000 to 4000 psi hydraulic pressures.

Nozzle area control in the cruise mode is provided by varying the convergent and divergent (upper) flaps by means of hydraulic actuators.

The ventral flap positioned by a dual cam mechanism has two functions:

- o Expansion area control in the cruise mode.
- o Nozzle throat area control in the V/STOL mode.

The ventral flap must be varied during cruise mode operation to provide efficient expansion of the nozzle flow. This variation is accomplished by a single cam drive mechanism.

The ventral flap must also be varied with the deflector angle. This is necessary to control nozzle throat area as the deflector is rotated through the thrust vectoring range. A second cam mechanism provides this motion. Inflight thrust vector control is provided by a movable external expansion flap. This flap is best suited to integration with the aircraft control surface actuation system, but can be engine controlled if required.

3.6 Seals

Sealing is a critical area in a high performance exhaust nozzle. The seals must be conformable and capable of accommodating load deflections, thermal expansion and manufacturing tolerances. Experience gained with effective sealing techniques in previous exhaust nozzles and thrust reversers has been factored into the ADEN design. High excursion seals are used on the flap edges and deflector to accommodate the flap panel deflections. In areas where gap variation is small, such as at the ventral flap hinge, simple elastic leaf seals are used. The reduced number of moving parts in the ADEN relative to a conventional flap and seal nozzle results in a 30 to 50% reduction in the gap length to be sealed.

3.7 Cooling

To achieve high lift thrust levels, high temperature augmentation is required and the key to success with such a V/STOL exhaust system is the cooling scheme. Particularly difficult cooling problems are characteristic of V/STOL exhaust systems for augmented operation.

A conventional augmented exhaust system has only minor changes in pressure and velocity along the flowpath. This means that efficient film cooling can be achieved by use of constant area slots or holes for injecting the cooling film with only minor changes in coolan w resulting as the operating conditions are varied. By comparison, the cooling system of ADE xhaust systems are subject to large gas stream pressure gradients during transit on from cruise to d mode. This could disturb or reverse the normal flow of cooling air provided by a conventional ng system design.

The ADEN cooling system provides effective, reliable cooling of the hot exhaust system parts with the cooling flow available from the fan air stream. The flow is ducted around the augmentor liner, distributed through structural ribs, and metered to provide the varying amounts of cooling flow as required during cruise and vectored mode operation.

During vectored operation the nozzle throat is rotated with the deflector so that the gas flow is turned upstream of the throat at velocities substantially lower than sonic velocity. Pressure losses minimized and efficient performance achieved. This feature, while benefitting performance, results in difficult coolant flow control since the changes in flowpath geometry result in wide variation of velocity and pressure of the hot gas along the flowpath.

A variable valving system is used to regulate the ADEN cooling flow under the varying pressure conditions. Adequate cooling flow in the lift mode is provided when flowpath cross section areas are large, velocities low and main stream static pressure high. Adequate but not excessive coolant flow is also provided when operating in the cruise mode with smaller flowpath cross section areas, high velocities and correspondingly low stream pressures. In the cruise mode, the coolant flow to the stowed deflector is shut off.

Cooling of the deflector surface is difficult since, due to the centrifugal forces which result from turning the flow, the gas stream static pressure is as great or greater than the fan air pressure which is the only available source of coolant. This condition precludes use of film cooling and in general prevents coolant from flowing from the cool side of the deflector liner to the hot side. An impingement/convection system is utilized for cooling the critical flow deflecting surface. Figures 7 and 8 show the coolant flow distribution in the forward and V/STOL modes respectively.

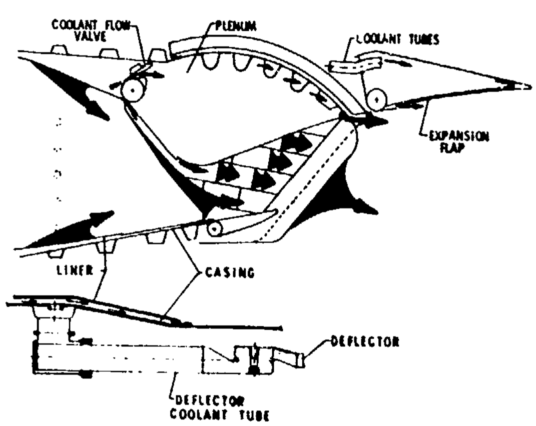


Fig. 7 ADEN Forward Mode Cooling Arrangement

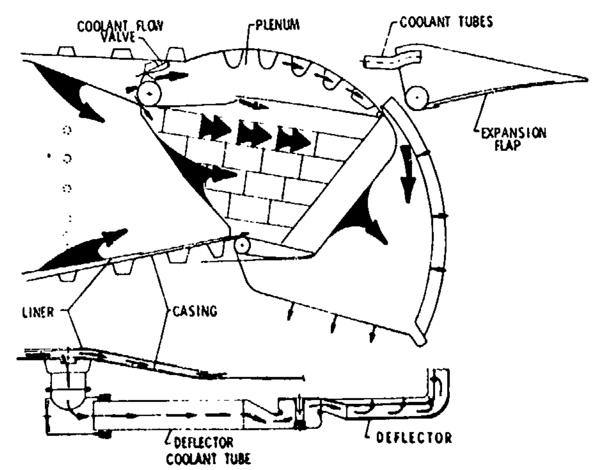


Fig. 8 ADEN Vectored Mode Cooling Arrangement

3.8 Reliability Considerations

Reliability considerations in a V/STOL exhaust system are more critical than in a conventional cruise exhaust nozzle, especially during take off and landing. Reliability has been emphasized in the design of the ADEN exhaust system. Proven reliable components are used wherever possible and in general, risk is held to a minimum consistent with a light weight high performance exhaust nozzle.

Loss of hydraulic pressure is a serious concern. The ADEN control system makes use of a proven, reliable hydraulic pump. In addition, a fail-fix servo valve feature is incorporated which, in event of loss of hydraulic pressure, will lock the actuator piston into the position at the time of failure.

In the event of an electrical control failure, either "hard over" or "loss of power", the controlled variable is maintained in the last command position by the action of the fail-fix servo valve. This valve is designed to block flow to and from the actuators at either end of the valve stroke or at mid-stroke. The ADEN makes use of the most reliable sensors and feedback devices available.

Quarter Scale Test

Since cooling of the ADEN was a more complex problem than cooling of conventional exhaust systems, a one-quarter scale ADEN (relative to YJ101 size) component test was conducted to substantiate the cooling design and provide data in support of the design effort.

The cooling test set-up (see Figure 9) consisted of an existing jet fuel burner facility modified by adding cooling air lines and flow metering sections to simulate the quantity, temperature and pressure level of cooling air supplied by the YJ101 engine. The design also simulated the cooling distribution design of the ADEN.

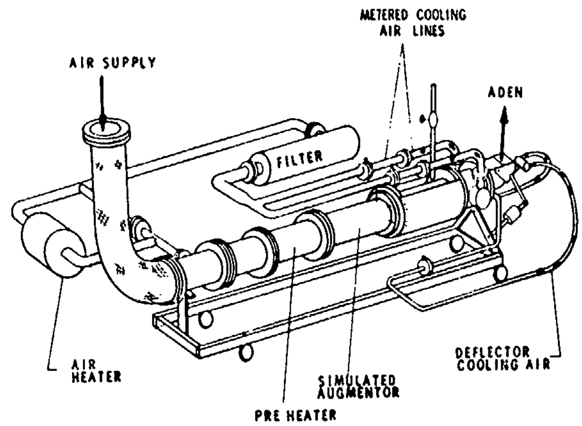


Fig. 9 ADEN Cooling Test Set Up

Since the augmentor burning length could not be scaled down with diameter, the forward position was water-cooled and only the geometrically scaled ADEN section was cooled by the metered air supply. Portions of the ADEN model not critical to the cooling system evaluation were of boiler plate design for ease and economy of hardware fabrication. However, all hot parts simulated the flight weight design relative to material, thickness, construction and cooling design.

4. Full Scale ADEN Demonstrator Test

The ADEN was statically tested during August 1976 on a YJ101 engine. Total test time was 40.8 hours of which 13.8 hours were conducted at augmented power settings. Jet deflection angles tested included 0 (Cruise), 45, 60, 70 and 90 degree vectoring. All mechanical systems performed as designed and no structural problems were encountered during the test. Figure 10 presents the nozzle installation during the test. Data analysis is presently in progress and preliminary results indicate conformance with the model test data shown in Figure 5.

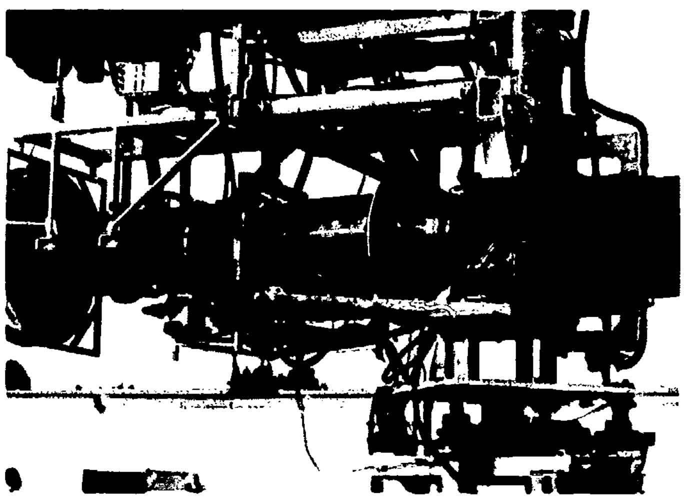


Fig. 10

DISCUSSION

E.Hienz

What temperature was the deflector run at and what weight penalty must be considered for such a nozzle?

Author's Reply

The temperatures were in the range of 1600 to 2700°F. There is a weight penalty – it is heavy -- but I do not know exactly the numbers, it may be 15 to 20%.

J.M.Hardy

Quelle masse d'air utilisez-vous pour le refroidissement de la tuyère?

Author's Reply

Fan discharged air was used for cooling the nozzle.

J.M.Hardy

Oui, mais quel est son pourcentage par rapport à un débit de masse/moteur?

Author's Reply

I don't know the exact answer to that question. Not included in the paper was how the air flows are distributed in different flight regimes. During the vector mode the deflector is cool but during the horizontal mode it is not. So a different amount of cooling air is used in different parts of the flight regime.

N.G.Hatton

Is the performance loss relative to a normal axisymmetric convergent nozzle acceptable in the normal cruise mode?

Author's Reply

To date we haven't shown off the performance loss. It is very little, if any.

The data on the full scale test have not been reduced completely. It's very recent, but it looks very good. The paper shows that the deflected mode is much better than any other vectoring nozzle we have seen so far.

J.Kurzak

What thrust loss is expected for vertical as compared to horizontal thrust? (Pressure loss due to turn plus exhaust coefficients).

Author's Reply

There is 100% thrust available when the afterburner is used in both cases. The nozzle thrust coefficients are about the same in horizontal and vertical.

VARIABLE CYCLE ENGINE APPLICATIONS AND CONSTRAINTS

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ABSTRACT

This paper will discuss the potential applications for Variable Cycle Engines. It will also discuss the payoffs for variable cycle engines and the constraints on these engines both from a totally installed aspect, as well as the internal engine restrictions.

POTENTIAL APPLICATIONS OF VARIABLE CYCLE ENGINES

There are several potential applications of Variable Cycle Engines in military and commercial missions. We will discuss the Military applications first. VCE's will show an advantage for applications which have some of the following characteristics.

- The basic design mission will have engine requirements at both supersonic and subsonic flight conditions.
- A requirement for operation flexibility by variation of the flight speed in the design mission.
- Alternative, non-design mission with emphasis on fuel savings.

There are always alternate non-design missions for any military aircraft. With the current emphasis on the reduction of fuel usage and the military logistics problem of fuel availability, greater emphasis is expected to occur on the fuel used in the prime and alternate mission of future aircraft. Four possible applications of VCE's are discussed for the following aircraft types:

- Air to ground penetration with supersonic and subsonic missions.
- Supersonic air-to-air combat fighter.
- Ground Support/Attack Fighter/Bombers
- VTOL Fighter applications.

The best design mission for a military air-to-ground penetration will include significant supersonic and subsonic mission legs. This type of aircraft should have extreme flexibility in the penetration mode in order to give the battlefield commander the operational flexibility to vary his penetration mode and increase the survivability of his force. To the engine designer this will place a premium on variable cycle features during the design mission. The alternate missions for such a vehicle will contain many subsonic flight conditions and will, consequently, show additional payoff for VCE's.

Air-to-air combat fighters with supersonic capability have been in the inventory for the last twenty years, but these designs are not capable of efficient supersonic operation resulting in the fact that they are not utilized in the supersonic arena. It is possible to design an aircraft with high supersonic cruise efficiency which would then permit improved combat persistence in the supersonic mode. Such a fighter would, of necessity, also have requirements for efficient supersonic and subsonic engine operation in terms of maneuverability and persistence. Against these requirements the variable cycle engine could show an advantage.

There is always an ideal requirement to have an aircraft loiter on station near the area of action from which it could be called for combat. This type of aircraft places a large emphasis on low power fuel consumption and engines optimized for that condition do not have good combat performance. A variable cycle engine might have a strong payoff in such type mission and aircraft.

The design of a VTOL fighter is of necessity a significant compromise procedure. The powered lifting requirements and resulting configuration, in general, conflict with the requirements for a fighter or attack aircraft. The flow modulation and versatility inherent in possible VCE designs could have significant payoffs in improved performance and simplifying the overall design.

In commercial applications the best candidate is the supersonic transport which has mixed mission requirements. It is desirable to fly supersonically as much as possible, but due to the multitude of route segments much or all of the flight could be subsonic. The take-off and approach noise also plays a factor in the SST system design, and this requirement plus the supersonic/subsonic flight requirements would be a natural application for a variable cycle engine.

It is obvious that there are many possible applications for variable cycle engines, however, there are also many constraints in which the variable cycle engine must fit. Although ideally it would be nice to have wide swings in engine characteristics in the practical case, this is not possible. The instal-lation effects, sizing effects and weight and complexity all will limit the arena of the variable cycle engine.

INSTALLATION EFFECTS

The cycle characteristics of the mixed flow bypass engine are strongly influenced by the require-ment that the static pressures be balanced at the point of confluency between the two streams. A low design fan pressure ratio (FPR) demands a low turbine exit pressure with a large amount of energy available to the turbine. Concurrently, the energy per unit fan flow is lower for the low FPR design. These two phenomena work together to drive the bypass ratio (BPR) up for a low ıan pressure ratio design and down for a high FPR design. Figure 1 illustrates this characteristic for a range of designs consistent in overall cycle pressure ratio, turbine temperature, and component performance charac-teristics.

BYPASS RATIO VARIATION WITH FAN PRESSURE RATIO FOR ADVANCED TECHNOLOGY TURBOFAN ENGINES

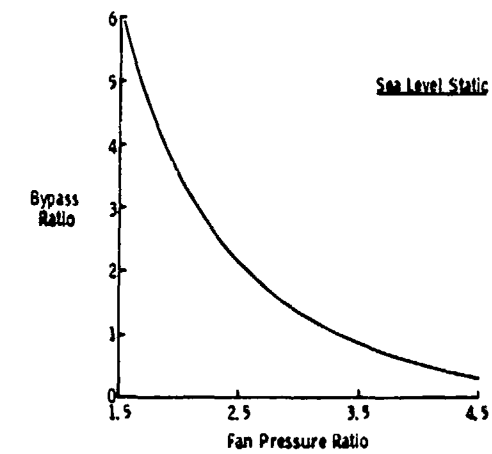


Figure 1

In the supersonic flight regime, bypass cycles having high fan pressure ratio tend to have lower values of specific fuel consumption (SFC) both in the unaugmented and in the augmented modes when compared with lower FPR cycles, and in the unaugmented mode, significantly more thrust. Although augmentation will tend to bring th.ı thrust levels of the low and high FPR designs more closely together, the low FPR engine falls short of catching up (for engines having equal overall engine airflow). Figure 2 illustrates this characteristic for augmented engines designed for the Mach 2 speed range. The FPR 4.5 design not only has a larger maximum thrust per unit airflow but also displays a substantially lower SFC level over the full thrust range. For the levels of turbine inlet temperature and component performance levels used a 4.5 FPR design will be limited in BPR to a value less than 0.5 giving this engine cycle characteristics similar to a typical augmented turbojet cycle.

COMPARISON OF SPECIFIC FUEL CONSUMPTION VARIATION WITH THRUST FOR VARIOUS FAN PRESSURE RATIO TURBOFANS

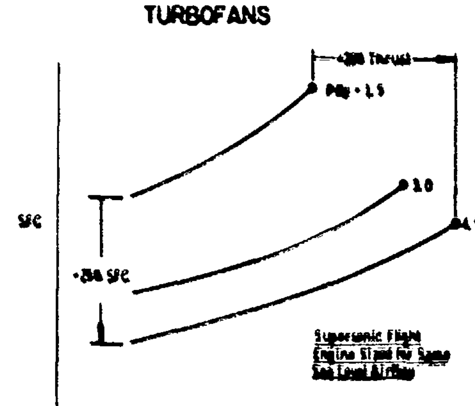


Figure 2

In the subsonic flight regime the low FPR design has substantially lower minimum SFC levels but still falls short of the specific thrust capabilities of the high FPR designs. These characteristics are illustrated on Figure 3.

COMPARISON OF SPECIFIC FUEL CONSUMPTION VARIATION WITH THRUST FOR VARIOUS FAN PRESSURE RATIO TURBOFANS

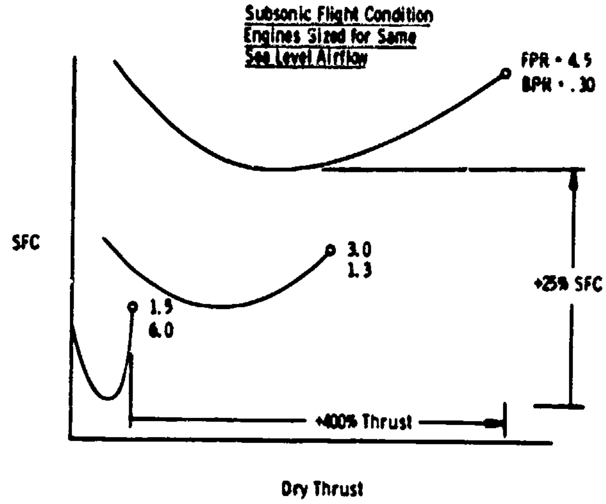


Figure 3

When designing a powerplant for a mission application having substantial flight time in both the subsonic and the supersonic regimes, it would be desirable to display the characteristics of the low FPR design subsonically and those of the high FPR design in the supersonic conditions. The desire to develop engines having such flexibility has led to substantial effort directed toward variable cycle engine (VCE) configurations.

The configurations currently under study do have the ability to adjust their basic cycles in this manner but mechanical constraints tend to limit the range of FPR variation to a much lower degree than that illustrated in the previous charts. With some of the designs the range of effective FPR could be extended to the levels illustrated; however, such a cycle would tend to display thrust mismatch in a real aircraft due to the large changes of specific thrust levels associated with this range of FPR.

A less apparent problem which limits the useful range of FPR values in specific applications stems from the conservation laws governing the creation of propulsion force by reaction. The mechanical energy required to accelerate the working fluid increases as the square of the velocity while the thrust produced increases directly with velocity. Thus the energy required to produce a given level of thrust tends to be proportional to the velocity of the exit jet and consequently tends to increase with FPR. Thus, in the ideal (no loss) case, the optimum cycle would tend to have a very low FPR with large flow rates (low specific thrust) and low energy addition. This can be clearly illustrated by inspection of the algebraic equations describing the laws of kinetic energy and momentum:

$$\frac{\text{Thrust}}{\text{Flow}} = \frac{V_{jet} - V_o}{g}$$

$$\frac{\text{Jet Power}}{\text{Flow}} = \frac{V_{jet}^2 - V_o^2}{2g}$$

$$\frac{\text{Thrust/Flow}}{\text{Jet Power/Flow}} = \frac{2}{V_{jet} + V_o}$$

Obviously, the most thrust per unit jet kinetic power is achieved when the jet velocity, V_{jet} approaches the flight velocity, V_o.

When the effects of the various loss mechanisms at work in an engine are accounted for, a new dimension is added to these equations. The mechanical power (Shaft Power) must equal the jet power plus the power dissipated by internal losses. Defining Δh loss in the usual manner as a loss in thermodynamic enthalpy per mass unit, with J converting thermodynamic units of work to mechanical units, as used in the above equations:

$$\frac{\text{Power Losses}}{\text{Flow}} = J\Delta h \text{ Loss}$$

and

$$\frac{\text{Shaft Power}}{\text{Flow}} = \frac{V_{Jet}^2 - V_o^2}{2g} + J\Delta h_{Loss}$$

Installation drags associated with the nozzle and inlet of a propulsion system must be deducted from the momentum thrust. The remaining installed net thrust is then:

$$\frac{\text{Installed Thrust}}{\text{Flow}} \quad \frac{V_{jet} - V_o}{g} - \frac{\text{Drag}}{\text{Flow}}$$

It is convenient to describe this installation drag in terms of a conventional aerodynamic drag coefficent referenced to the free stream capture area.

$$\frac{\text{Drag}}{\text{Flow}} = \frac{C_D A_o q_o}{\text{Flow}}$$

but $q_o = 1/2 \rho_o V_o^2$ by definition of q,

and $\text{Flow} = \rho_o g A_o V_o$ from continuity

and $\frac{\text{Drag}}{\text{Flow}} = 1/2 \frac{C_D V_o}{g}$

Dividing the installed thrust by the shaft power:

$$\frac{\text{Installed Thrust/Flow}}{\text{Shaft Power/Flow}} = \frac{2 - \frac{C_D V_o}{V_{jet} - V_o}}{(V_{jet} + V_o) + \frac{2Jg\, \Delta h_{Loss}}{V_{jet} - V_o}}$$

Inspection of this equation reveals that the beneficial effects of reducing the jet velocity are still at work in the first term of the denominator. However, the two major loss phenomena accounted for each contributes a counter effect which, as jet velocity is reduced toward the flight velocity, will, in combination, overcome this beneficial effect, with the fundamental effect that for any specific level of losses and flight velocity there will be an optimum jet velocity (and FPR).

This analysis is quite consistent with the characteristics of real hardware because power dissipation due to cycle loss phenomena does tend to be proportional to the flow rate (for example, a given pressure loss ratio will dissipate twice the power for twice the flow), and installation drags are closely related to the size of the propulsion system, in turn closely related to the free stream capture area.

It is thus clear why an engine flying at high flight speeds demands higher FPR values. At fixed loss levels a higher V_o demands an equivalent increase in V_{jet} and FPR; further, the loss levels associated with supersonic propulsion tend to be much higher than those encountered in propulsion systems designed for subsonic propulsion.

In looking at actual engine designs this phenomena is obviously a predominant factor in the overall design concepts. The high BPR, low FPR cycles in use in subsonic transport aircraft are characterized by a careful avoidance of losses: Well rounded lip inlets with almost no ram total pressure loss and complete recovery of spillage drag; clean internal ducting and nozzles designed for very low internal and drag losses.

For high-performance-supersonic aircraft, on the other hand, a complex sharp lipped inlet is required having lower subsonic ram total pressure recovery, large spillage drag potential, and substantial parasitic flows. Relatively large pressure drops are unavoidable with reheat augmentors and the nozzles tend to be complex with internal losses, leakage flows, and drag levels. The overall result is that these engines must operate at relatively high FPR level even in the subsonic flight regime to produce thrust efficiently.

These effects are illustrated on Figure 4. The subsonic transport design level of losses gives the best performance in a FPR range of about 1.5. A propulsion system typical of a supersonic strategic bomber will best utilize a FPR of about 2.5. For a typical mixed mission fighter, where some avoidable losses are accepted in the interest of keeping the propulsion system weight and complexity at a minimum, a FPR of about 3.5 is needed.

typical fighter (installed) can be in excess of 50% at a subsonic flight condition.

COMPARISON OF INSTALLED SPECIFIC FUEL CONSUMPTION VARIATION WITH FAN PRESSURE RATIO FOR VARIOUS TYPES OF INSTALLATIONS

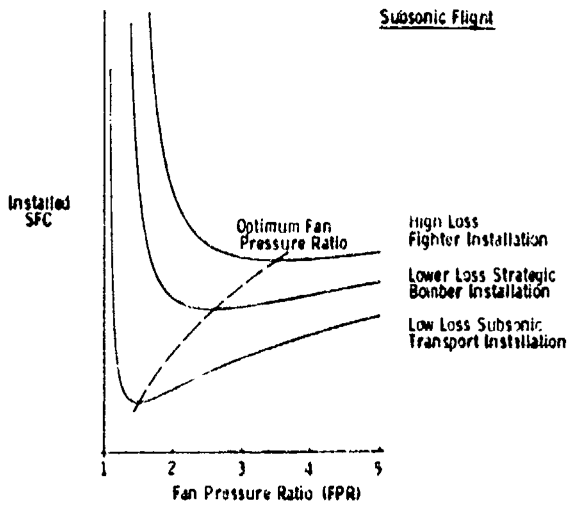


Figure 4

Low FPR designs can also present size problems for an augmented supersonic engine as shown on Figure 5.

AUGMENTOR SIZE VARIATIONS WITH FAN PRESSURE RATIO FOR ADVANCED TURBOFAN ENGINES

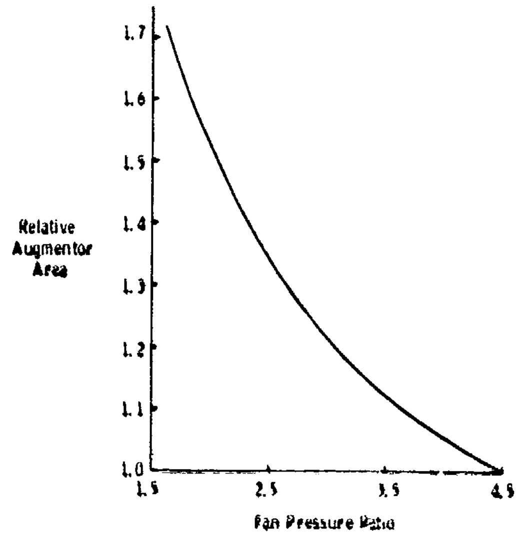


Figure 5

The VCE concepts currently being studied have a capability to accommodate large changes in effective flow size in going from the subsonic to supersonic flight conditions. This is accomplished, for example, by designing the fan for excess flow capability and allowing the excess flow to bypass the engine with an effective low FPR and relatively low losses at subsonic cruise conditions. In the supersonic flight conditions the fan speed is reduced, and variable geometry is used to low flow them, allowing the main engine to swallow most of the flow, with an effective high FPR-low BPR cycle.

One installation problem resulting from this capability is a subsonic flow demand in excess of that available from an inlet properly sized for the supersonic conditions, illustrated on Figure 6. As shown on Figures 7 and 8 the use of auxiliary inlets to accommodate this unusual flow demand will result in either high drag or low inlet ram total pressure ratio recovery, or both. These losses can completely overcome any cycle benefit from such cycle flexibility if it is pushed too far. Thus the fundamental conservation laws when properly applied tend to limit the range over which FPR and BPR can be effectively varied.

INLET AIRFLOW VARIATION WITH FLIGHT MACH NUMBER

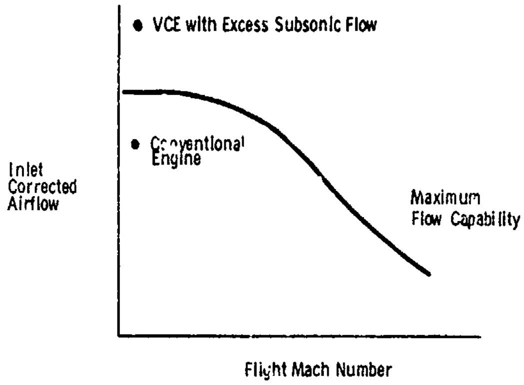


Figure 6

DRAG COEFFICIENT VARIATION WITH FLIGHT MACH NUMBER FOR A SIMPLE SCOOP AUXILIARY INLET

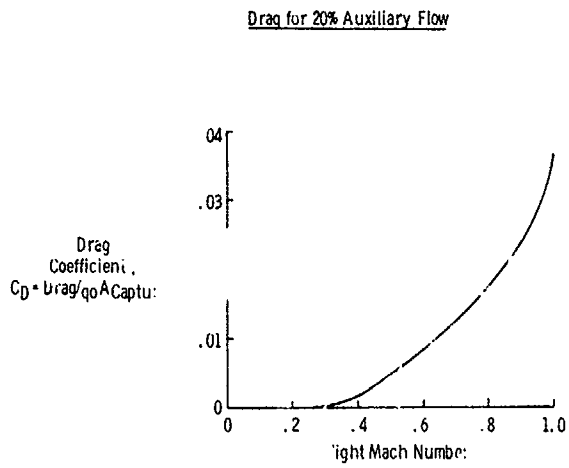


Figure 7

AUXILIARY FLUSH INLET RAM RECOVERY VARIATION WITH FLIGHT MACH NUMBER

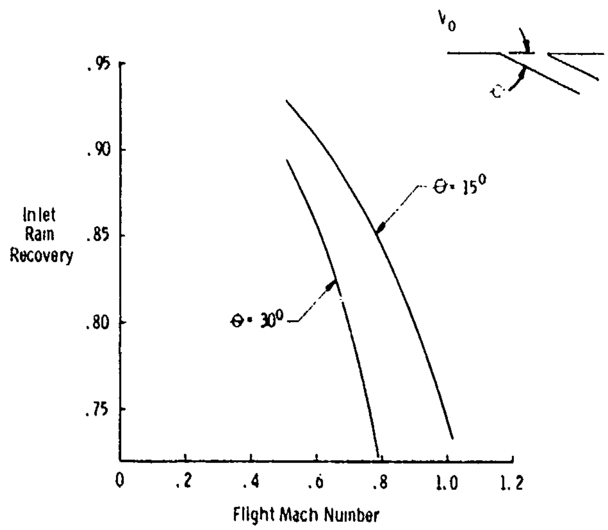


Figure 8

It can be concluded that studies of VCE concepts should concentrate on using this flexibility over a limited range of effective FPR values working toward good engine matching to the supersonic inlet characteristics while minimizing subsonic conditions installation losses.

SIZING EFFECTS

Another very important consideration in the variable cycle selection is the effect of the mission sizing requirements on the engine type. For the supersonic transport mission, the ratio of fuel weight to aircraft gross weight is very high and therefore any improvement to supersonic fuel consumption is extremely important. In this case, the engine type would favor the higher fan pressure ratios for good supersonic performance. However, the SST has another requirement which is acceptable noise levels at takeoff and approach. During this operation the most desired cycle is one that has low specific thrust and high airflow. At this flight condition from 0 to .3 Mach number, auxiliary inlets have good ram recovery so therefore higher bypass ratios which require high airflows at this condition pay off and the engine would revert to its normal supersonic cruise cycle. Similarly for VTOL applications, maximum thrust is required at take off. Like the SST this high flow is only required at conditions where auxiliary inlets are good. Therefore, high flowing or high flow conditions at takeoff would be beneficial for VTOL since in this case auxiliary inlets are possible.

Now for the normal type of combined supersonic/subsonic military mission requirements, the engine sizing can take on different aspects. The maneuverability, acceleration thrust, and combat ability all will influence the cycle selection and tend to drive it away from its cruise only optimum. Missions which have very little or virtually no maneuver requirements will tend toward the lower fan pressure ratios and higher bypass ratio when a significant amount of the mission is subsonic. This is shown in Figure 9. As the maneuver requirements increase in both subsonic and supersonic arenas, the engine cycles will optimize to higher fan pressure ratios and lower bypass ratios to give good combat ability. For the normal type mission, however, the major portions of it are subsonic where the higher bypass ratio is desirable. So in this case the variable cycle could have a significant payoff in allowing the high fan pressure ratio cycle with good combat ability to also have good subsonic performance.

COMPARISON OF AIRCRAFT GROSS WEIGHT VARIATION WITH FAN PRESSURE RATIO FOR DIFFERENT MANEUVER REQUIREMENTS

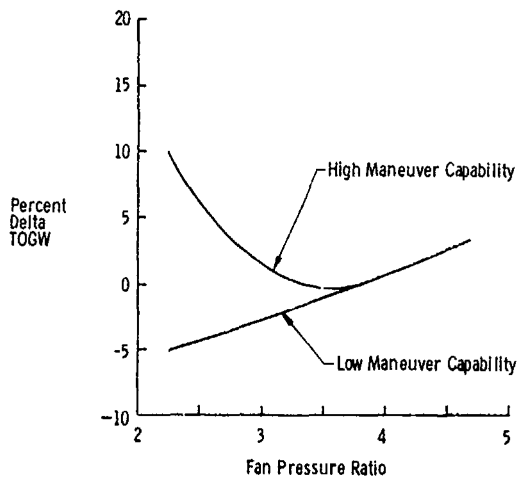


Figure 9

The final selection, of course, will depend upon the mission requirements. However, it is obvious that a possible attribute of the variable cycle engine would be one which could swing the bypass ratio to give good maneuverability during combat and improved subsonic fuel consumptions during other conditions.

COMPLEXITY AND WEIGHT

Many variable cycle engines have been studied and even tested in the past 10-15 years at the General Electric Company.

Let us now examine some of the early VCE types that were basically attempts at combining the desirable features of the turbojet and turbofan cycles in one package, but at that time were not completely satisfactory because of unacceptable weight and complexity or excessive losses due to unusual engine component requirements. Some of the early concepts and all of the ones evolved during the last few years work on the potentially large installation losses.

EARLY VCE CONCEPTS

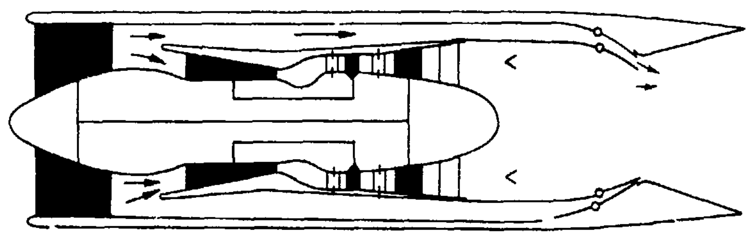
VARIABLE PUMPING COMPRESSOR (VAPCOM)

This concept was conceived by several engineers at the Aero Propulsion Lab at Wright Field around 1960. It was the first approach at trying to combine the best features of the turbojet and turbofan into one system. The concept, shown schematically in Figure 10 works in the following manner. During maximum power and supersonic cruise operation, the engine is run as a dual rotor turbojet engine with almost all of the front compressor flow being passed through the core compressor. The outer bypass duct is closed with only a small leakage flow allowed to pass through the duct and outer nozzle. For subsonic flight the outer duct and nozzle are opened and the core compressor is low flowed by closing all the compressor stators. This effectively increases the bypass ratio from 0 to 1.0 and converts the cycle from a turbojet to a separated flow turbofan. Variable turbine geometry is required on both the high and low pressure turbines.

While in theory the engine approaches an optimum mixed mission propulsion system, in practice the losses associated with the core flow modulation negated much of the higher operating bypass benefits. Also weight and complexity factors detracted from the overall value of the concept. Table 1 summarizes the operating principles and problems associated with the concept.

VARIABLE PUMPING COMPRESSOR (VAPCOM)
ENGINE SCHEMATIC

TOP VIEW - TURBOFAN MODE (LOW FLOWED CORE , MAX DUCT FLOW)



BOTTOM VIEW - TURBOJET MODE (HIGH FLOWED CORE , MINIMUM DUCT FLOW

Figure 10

TABLE 1

VAPCOM

VARIABLE PUMPING COMPRESSOR

- Vary Bypass from 0 - 1 by Compressor Stator Closure
  - Variable Turbines Required
- Variable Stator Fan Also Used for Bypass Control and Flow-Speed Optimization
- Problems
  - Core Compressor Performance with Wide Flow Swings
  - Overall P/P Drop with Bypass Increase Minimized Subsonic Improvement
  - Relatively Complex, Heavy (Core Sized for Full Fan Flow)

This engine concept was developed in the 1960 time period by the General Electric Company. It too was conceived as a system that combined the turbojet-turbofan features in one package but in an entirely different manner than the VAPCOM. Figure 11 illustrates the concentric turbojet principal of the Flex Cycle engine. The basic flowpath illustrated shows a front fan followed by a conventional core engine. A second burner is located in the bypass duct. The unusual aspect of the engine lies in the turbines that power the front fan. As can be seen, two separate turbine systems are located on the fan shaft. The first fan turbine located behind the core turbine supplies part of the energy needed to drive the fan. The large aft turbine - the critical component of the engine - supplies the remaining fan energy. The energy split is a function of the cycle parameters and operating flight condition.

The engine is run in the turbojet mode with the outer duct burner turned on. For low flight Mach numbers both the outer and inner burners are on for maximum performance. As flight Mach number increases the core engine can be effectively slowed down to minimize compressor discharge temperature, while the front fan is run at full speed to maximize supersonic flow. The duct burner and aft turbine supply most of the energy for supersonic flight. The ability to keep the fan at full speed while the core is slowed down permits the cycle to have a very high overall pressure ratio for subsonic flight while not incurring high compressor discharge temperatures during supersonic flight.

The turbofan mode of the engine results when the duct burner is shut off and the main burner alone produces energy for the entire compression system. A static pressure balance occurs in front of the aft turbine and basically mixed flow exhaust properties result.

Even with the duct burner off, the basic cycle operation requires the aft turbine to still supply a relatively high percentage of the horsepower needed to drive the fan. It is this dry mode of operation where radical swings in turbine energy, corrected speed and turbine flow function occur. In past studies, the large changes in turbine energy and their accompanying exit swirl swings and losses have proven to overcome all the apparent subsonic advantages of the system resulting in no net system payoff. Also, the engine is relatively low in supersonic specific thrust unless configured with an afterburner. The two burner configuration shown is already complex and heavy and the addition of another burner system would only add more complexity. Table 2 summarizes the operating characteristics and problems associated with this concept. This type of concept would be applicable to higher cruise Mach numbers.

FLEX CYCLE SCHEMATIC

. OUTER BURNER ON -CONCENTRIC TJ MODE-MAX POWER

. OUTER BURNER OFF- MIXED FLOW TF MODE

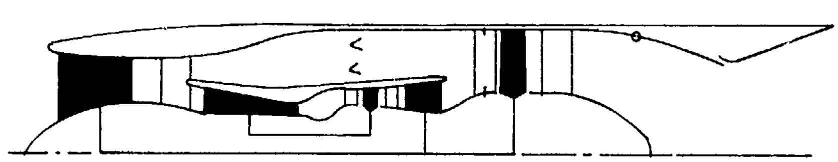


Figure 11

TABLE 2

FLEX CYCLE

- o Originally Studied in Early M3.0 SST Work (Called Composite Cycle)
- o Two Modes of Operation
  - Concentric Turbojets with Outer Burner - On
  - Mixed Flow Turbofan with Outer Burner - Off
- o This VCE Overall PR Trends in Right Direction
  - High for Subsonic
  - Reduced for Supersonic (Core Slows Down)
- o <u>Problems</u>
  - Aft Turbine Aerodynamics
  - Low Specific Thrust Unless A/B is Added
  - Complex, Heavy (Weight Increase Offset Cycle Improvements)

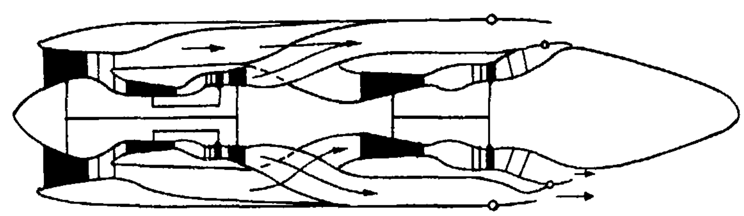
TURBO AUGMENTED CYCLE ENGINE (TACE)

A true marriage of the turbofan and turbojet occurs with the Turbo Augmented Cycle Engine. As shown in Figure 12 this engine is composed of two engines in series, a turbofan and a turbojet. A complete turbofan engine is available for use during subsonic flight. In this mode the bypass and core streams of the turbofan mix together and exhaust through a common nozzle. The aft turbojet is not in operation for normal subsonic cruise and the pressure loss in the fan stream should be lower than with augmented flameholders. For maximum power and supersonic flight the turbofan bypass flow is diverted into a duct that feeds the aft turbojet engine. This supercharged mode of operation results in the turbojet becoming a very efficient augmentor for the fan duct flow.

While the system is derived from two well proven engine concepts it does have several drawbacks. One is the overall weight of the total propulsion system. Another is the lack of specific thrust potential for non-augmented versions of the engine. Addition of an afterburner to the turbojet and possibly a duct burner to the mixed exhaust of the front turbofan would make the system competitive on a thrust basis but would magnify the overall maintenance and servicing problems. Table 3 summarizes the characteristics of the TACE concept.

TURBO AUGMENTED CYCLE ENGINE (TACE) SCHEMATIC

TOP VIEW - TURBOFAN ONLY MODE (AFT TJ OFF) - SUBSONIC



BOTTOM VIEW - COMBINED MODE (AFT TJ ON) - MAX POWER/SUPERSONIC

Figure 12

TABLE 3

TURBO AUGMENTED CYCLE ENGINE (TACE)

- Basically 2 Engines
  - Turbofan for Subsonic - TJ Shut Off
  - Turbofan Used as an Augmentor for Bypass Flow
    - Supercharged by Fan
- Problems
  - Primarily Weight and Configuration
  - Low Specific Thrust at High T2 Levels Unless A/B is Added
    - Core Flow from TF Might Need an Augmentor Also

The preceding three concepts were all attempts at trying to combine the turbojet and turbofan in one acceptable package. Either some component proved to have extreme aerodynamic or mechanical problems or the weight and complexity of the system overcame whatever cycle advantages was initially present. While a true turbojet/turbofan mode change is probably not achievable in a viable system, present VCE studies are indicating that advancements can be made over todays existing engines.

MODULATING BYPASS VCE CONCEPT

The Modulating Bypass (MOBY) cycle was developed in the 1973 time period in response to the U. S. Air Force's request for engine concepts that addressed the problems of throttle dependent installation losses. Figure 13 illustrates the initial cycle concept which is fundamentally a separated flow duct burning turbofan cycle configured as a three spool engine. The unique aspect of the configuration is found in the fan system. As can be seen, the fan is divided into two sections, each being on a separate shaft with a bypass duct located between them. Downstream of the second fan section is a conventional core engine and second bypass duct. A duct burner is located in this duct to provide thrust for acceleration and supersonic flight.

For maximum power and supersonic flight the engine runs as a standard duct burning turbofan. Almost all the front fan section flow is passed through the second fan section and then divided between the second bypass duct and core. A small leakage flow goes into the outer bypass duct and out the outer nozzle.

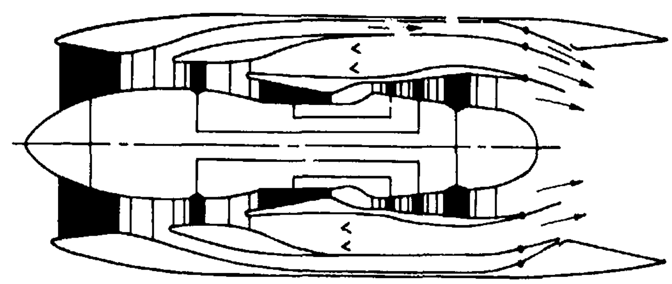
For part power subsonic conditions the front fan operating mode is set to perfectly match the inlet requirement. This airflow match is held while thrust is varied by reducing the second and third spool speeds. Turbine temperature is reduced at an optimum rate by utilizing the variable turbine systems. With this mode of operation overall bypass ratio increases markedly since flow rejected by the second fan section now goes into the bypass duct located between the fan sections. Also, overall fan pressure ratio reduces to more optimum levels for subsonic flight. Matched inlet flow can be maintained down to 50% dry thrust without any sacrifice in uninstalled SFC. Additional installed advantages occur since constant airflow thrust modulation results in more open nozzle settings that reduce aft end closure drag.

While this concept did effectively work on installation losses its overall complexity (three rotors, three nozzles, three variable turbines and two bypass ducts) was too great for serious consideration as a practical propulsion system. Also, since the basic concept was that of a separated flow turbofan, the resultant specific thrust potential was not as high as competitive mixed flow fan systems. Table 4 summarizes the MOBY concept.

MODULATING BYPASS (MOBY)
ENGINE SCHEMATIC

3 Spool Double Bypass

TOP VIEW–SUBSONIC–PART POWER (MAX OUTER DUCT FLOW)



BOTTOM VIEW–MAX POW MODE (MIN OUTER DUCT FLOW)

Figure 13

TABLE 4

MODULATING BYPASS VCE CONCEPT

- o Conceived as Inlet Matching Concept to Minimize Installation Losses
- o Concept Operation

Max Power Mode	- Runs as a Standard Duct Burning Turbofan
Part Power Subsonic	- 2nd Spool & Core RPM's Reduced, T4 Drops
	- Overall Bypass Increases, Fan PR & Overall PR Drop
	- Basically Thrust is Varied with P&T, Not W_A
	- Inlet Stays Matched, Afterbody Losses Reduced

- o Problems - Weight & Complexity (3 Nozzles, 3 Spools)
  - Relatively Low Specific Thrust Levels

SYSTEM PAYOFF FOR VCE'S

It is possible that the improved performance over the complete flight map may be more than offset by the additional weight and cost of the system. Figure 14 shows where a system payoff can be obtained by an improvement in subsonic fuel consumption versus an increase in engine weight. The fighter bomber with a fuel usage assumption of 50/50 subsonic supersonic used mixed mission allows the largest engine weight increase of all types mission. It shows that past variable cycle concepts weight increases were more than offset by the fuel consumption improvements. We can conclude, therefore, that the variable cycle engines to pay off must be relatively light or simple. Such types of engines have been identified and are currently being studied.

BREAK EVEN RELATIONSHIPS BETWEEN ENGINE WEIGHT INCREASE AND FUEL WEIGHT DECREASE FOR VARIOUS AIRCRAFTS

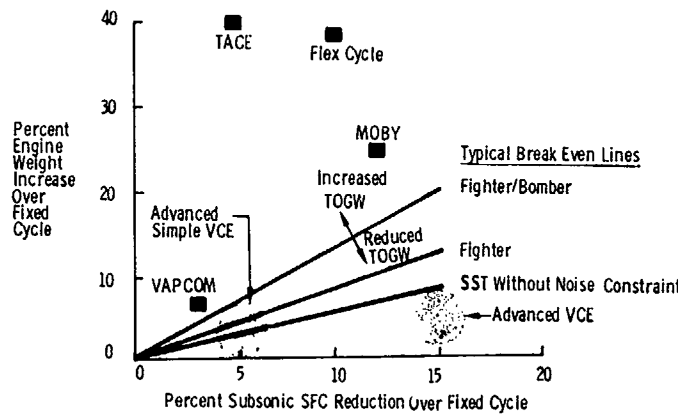


Figure 14

ADDITIONAL ADVANTAGES OF VARIABLE CYCLE ENGINES

For conventional turbofan engines compromises must be made in the selection of engine control areas (low pressure turbine, bypass duct areas, etc.) to allow engine operation over a wide range of flight Mach numbers. Normally, these areas are set to give near maximum performance at an expected key flight condition with operation at other flight points being somewhat poorer than a re-optimized engine could produce. These performance losses become more pronounced as the engine flight envelope is extended to higher Mach, higher altitude conditions.

Variable cycle engines that can adjust the engine control areas to suit each flight condition can show performance gains over conventional engines during off design conditions. Better inlet airflow matching at higher flight Mach numbers is one result of this. Also, the effect of engine bleed and horsepower extraction can be minimized by rescheduling duct and turbine areas. This becomes especially important at high altitude conditions. The effect of off-standard day operation can be better accommodated by using these variable area devices to keep spool speeds and operating lines at more optimum settings. Of equal importance will be the ability to recoup some of the performance losses that result from component deterioration. Original spool speed matches can be maintained by rescheduling turbine and duct areas.

A unique feature of the VCE's currently under study is their ability to supply large amounts of pressurized air for special purpose such as quiet, short take-off and landing and lift enhancement. Also, the great degree of flow control offered by the variable geometry engine features can lead to more simplified inlet and exhaust systems.

It is quite probable therefore that once payoff has been established in the basic mission variable cycles can have even additional payoffs some of which are not now identified. Just as variable compressor stators were used to extend engine operation in rather specialized cases, such as hot gas re-ingestion during thrust reverser or gun gas ingestion during gun fire on fighters, the variable cycle features will extend the capabilities of the system.

CONCLUSIONS

In conclusion, we find that although it is theoretically desirable to have an engine system that can operate in substantially wide variations of cycle conditions, there are practical limitations to what extent this can be accomplished. We find that installation effects, mission sizing effects, component complexity and off design operation, as well as weight, all place limitations there-on to the degree of flexibility that can be achieved in an installed engine. We find that practical attractive variable cycle engines may now be possible and the secret to this practical variable cycle engine is simplicity. Simple variable cycle engines have been identified and are currently under study and development. The next generation of engines for multi-mission use could very well be variable cycle engines if their increased weight, complexity and cost are more than offset by reduced airplane size, cost, more flexible practical operation and greatly reduced fuel usage.

DISCUSSION

J.F.Chevalier

Dans vos trois moteurs, vous avez en supersonique le diamètre du moteur finé par le débit masse au décollage, donc un grande trainée. Avez-vous des projets où l'encombrement est réduit en supersonique?

Author's Reply

I'm not sure I'm going to answer your question directly. But I did not mention one of the advantages of the variable cycle engines that are under study now, that allows you to make maximum use of the components: in other words, with the fixed cycle engines we will compromise all of the areas to give us a rather average performance over the flight map. By putting in these variabilities we now can improve even the supersonic performance. In my paper I really concentrated on subsonic because we do compromise the high supersonic by matching our engine at a lower Mach number, but indeed there are pays-off in supersonic performance too.

You mentioned that we do penalise the engine, I believe, because it may have a larger diameter. Yes, we try to do a very honest evaluation. We do not want to change technologies when going from a conventional engine to a variable cycle engine. And I have to admit that in our studies by everything we have learned on variable cycle engines, we found we have also been able to improve conventional engines. So they are catching up.

J.F.Chevalier

Je suis tout-à-fait d'accord avec votre dernière phrase; il faut chaque fois que l'on a fait un progrès dans les moteurs à cycle variable, introduire ce progrès technologique dans un moteur classique pour voir si l'on 'aurait pas le même résultat.

Author's Reply

However, I would like to say, just in terms of amusement, I find now that my cycle designers tell me that it is more difficult for them to design a fixed cycle engine because they now have to figure out where to fix it. It is much easier with a variable cycle engine.

HIGH EFFICIENCY ENGINE CYCLES FOR AIR TRANSPORT FUEL ECONOMY

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SUMMARY

Economic and environmental incentives imposed by future fuel conservation needs have created a resurgence of interest in the study of very high efficiency aircraft engine cycles. The effects of projected gas turbine technological progress, unconventional thermodynamic processes and advanced propulsion devices are assessed in terms of their fuel savings potential. The operating characteristics of two selected propulsion systems – an advanced turbofan and a turboprop engine – are projected and compared at points along a transport flight profile. The applicability of engine component geometry variation to improve propulsion system efficiency is also reviewed.

1.0 INTRODUCTION

The aircraft industry has always strived to provide the airlines with high efficiency, economical transport aircraft. Following the introduction of the first turbojet engines into commercial service, fuel consumption has been reduced by 30 percent as the result of the normal evolution of technology. However, with the advent of the energy crisis earlier in this decade, industry emphasis to further reduce fuel consumption and lower operating costs was increased.

Under NASA sponsorship, studies have been conducted to find ways of reducing fuel consumption in both current and future aircraft engines. This paper summarizes a propulsion feasibility study aimed at evaluating fuel conservative concepts that might be applicable to future commercial air transport engines.

As a result of these studies, it is apparent that substantial reductions in gas turbine fuel consumption can be realized through the development of turbofan technology. The technology necessary to achieve these reductions was identified under NASA Contract NAS3-19132. This technology was then used as the baseline for the study of unconventional propulsion systems to achieve further fuel savings. This study, in turn, resulted in the identification of a turboprop utilizing an advanced, highly efficient propeller as a possible future contender to the conventional turbofan.

In the following sections of this paper the turbofan engine and the unconventional engine possibilities are reviewed. The advanced turbofan and turboprop cycles are compared with the current turbofan in both domestic and international flight service to assess the fuel conservation possibilities with the two systems.

Because of the continuing and wide industry and Government interest in this subject, much of the material contained in this paper has been disseminated to the technical community via several technical papers and formal reports (1-5)*.

2.0 ADVANCED FUEL CONSERVATIVE CONVENTIONAL TURBOFAN

The major factor that influences the fuel savings potential of aircraft gas turbine engines is installed cruise TSFC. A reduction in TSFC produces a direct reduction in aircraft fuel load for a given flight time as well as an indirect reduction realized by the lower engine thrust required to propel the lighter airplane. From the time the first Pratt & Whitney Aircraft JT3 turbojet powered Boeing 707 was introduced, engine cruise TSFC has been reduced by over 30 percent to present day JT9D turbofan levels. This reduction has come about mainly by cycle improvements permitted by advances in component aerodynamics, materials, cooling, and structure-mechanics technology. These improvements have given the engine designer significantly greater freedom in selecting the best engine cycle based on TSFC considerations.

TSFC can be reduced by as much as 7-10% in the future by utilizing increased core engine cycle capability which is projected for the 1990's. Core cycle pressure ratios 50% higher than present and 111-220°C (200-400°F) higher maximum cycle temperature levels may be possible before material, gas path sealing, or hot parts cooling bleed air limits are reached. A further means of reducing future engine TSFC lies in increasing the efficiency of converting the core engine power output to useful thrust by utilizing the bypass fan principle. The turbofans of today represent the practical application of the principle. Rather than converting the available energy directly to thrust in an exhaust nozzle as in a turbojet, the bypass fan is used to accelerate an increased volume of air to a slower exhaust speed. In combination with the core engine advancements, the improved propulsion oriented components could result in turbofan engine cruise TSFC 10 to 15 percent lower than current turbofans by the 1990's (See Figure 1).

Slight additional cruise TSFC reductions are theoretically possible by altering turbine and primary exhaust nozzle flow capacities to increase engine efficiency. During typical cruise (75 percent of climb thrust rating) and in the absence of any penalties associated with the variable geometry, it is estimated that a theoretical reduction of slightly over 1 percent in TSFC can be obtained by reducing the flow capacity of the high pressure turbine and increasing the flow capacity of the low pressure turbine. These theoretical improvements are insufficient to justify the added complexity of the variable geometry features in the engine. For example, with variable stagger

*Numbers in parentheses designate References at end of paper.

turbine vanes, leakage is inevitable around the movable vanes. If even a one percentage point efficiency penalty is assumed in each turbine, all of the theoretical TSFC benefit is lost. Based on testing to date, this efficiency penalty is believed to be extremely optimistic. Therefore, the addition of variable turbines to improve the cruise performance of future subsonic transport turbofans would appear to be unwarranted.

3.0 ALTERNATE FUEL CONSERVATIVE PROPULSION SYSTEM POSSIBILITIES

Numerous variants of the conventional turbofan cycle have been studied in the past in an effort to maximize efficiency within the state-of-the-art. As part of the present studies, the technology of these variants was projected to the 1990's, consistent with the baseline turbofan, to examine the possibility of exploiting their energy-saving features. For the purposes of discussion and evaluation, the propulsion systems are separated into two subsystems, the primary cycle and the propulsor. To provide a common basis for understanding, Figure 2 shows schematically the components which are included in each subsystem.

3.1 Advanced Alternate Primary Cycles

Numerous alternatives to the Brayton cycle have been studied in the past in an attempt to improve thermal efficiency. Principal concepts include use of internal cycle heat exchange, non-steady flow primary combustion, and cycle compounding. Engines have been build and successfully run utilizing these concepts. The possibility of exploiting unconventional cycles to conserve energy in the future was reviewed as part of this evaluation. Results are summarized in the following paragraphs.

Internal Cycle Heat Exchange – Thermal efficiency of the primary cycle can be increased by raising the average temperature of external heat addition or by reducing the average temperature of heat rejection. A practical method of internal cycle heat exchange which accomplishes both of these is regeneration. Waste heat is extracted from the turbine exhaust gases and transferred to air entering the combustor which reduces the amount of fuel needed to achieve a given combustor exit temperature level. This concept is shown schematically together with a temperature – entropy Brayton cycle diagram in Figure 3. Heat extracted from the turbine discharge (Station 4) reduces the exhaust temperature to the level of Station 2, if complete heat transfer is accomplished, and preheats the combustor air to the Station 4 level. Fuel is then burned to further increase the temperature from the station 2a level rather than the customary station 2 value to the desired Station 3 level. Regeneration is useful up to the threshold cycle pressure ratio corresponding to equal compressor discharge and turbine discharge temperatures where the thermal efficiency equals that of the simple Brayton cycle.

Ideal regenerative gas turbine cycle performance was estimated and is compared with advanced technology simple Brayton cycles in Figure 4. As cycle pressure ratio is reduced below the threshold, at the example cruise condition, regenerative cycle thermal efficiency improves steadily until the cycle pressure ratio approaches a value of 5 or 10 where the thermal efficiency peaks out and then rapidly drops.

Regeneration is more effective in improving thermal efficiency at high combustor exit temperature levels and correspondingly higher turbine discharge gas temperatures. The average temperature of external heat addition increases as the threshold pressure ratio increases. Therefore, the regenerative cycle can be expected to require a higher combustor exit temperature than the corresponding simple Brayton cycle to maximize thermal efficiency.

The discussion so far has been based on ideal reversible heat exchange processes. Irreversibilities affect both the cycle choice and the benefit of regeneration. If less than complete heat transfer is achieved, for example, thermal efficiency trends change as shown on Figure 5. For the 1760°C (3200°F) temperature level example, a regenerator effectiveness of roughly 70 percent would appear necessary in a 1990's gas turbine, with a cycle pressure ratio of 20:1, to match the performance of the simple Brayton cycle with a 45:1 pressure ratio.

In arriving at a definition of a technologically possible future regenerator, several regenerative techniques were explored. In each case, an air-to-gas heat exchanger located behind the turbine last stage was selected as the most practical arrangement. The location of the regenerator was based on ducting requirements. A simpler system resulted by ducting the higher density compressor air rearward to the heat exchanger and forward to the burner, than by ducting the hot, lower density turbine gases in a reverse fashion. Other intermediary working fluids – such as liquid metal – to transfer heat from the turbine to the combustion region were considered and discarded because of large weight penalties.

Several air-to-gas heat exchangers were analyzed leading to the selection of a stationary, plate-fin counterflow matrix arrangement. Rotary regenerators, with wire screens or ceramic matrices, were also considered together with 2 pass, cross-counterflow tubular stationary recuperators. Although the recuperators were heavier than the rotory devices, they were more compact, presented fewer leakage paths, required no carryover of trapped gas flow between the two streams, and were simpler in construction. Of the two recuperators evaluated, the plate-fin solution was selected because of a significantly lower matrix and total recuperator weight requirement based on equal overall performance.

Evaluation of the plate-fin recuperator in context with nacelle geometry envelope constraints resulted in the modular concept with eight identical heat exchangers as shown in Figure 6. A hot gas approach Mach number of 0.2 provided a heat exchanger frontal area which required a heat exchanger height less than that of the rear turbine flange diameter and the width exceeded the turbine rear flange diameter by 55.9 cm (22 inches). The additional nacelle interior volume was used to advantage in providing regions for ducting the compressor air to and from the recuperator. A frictional pressure loss of 14.6 percent was estimated in turning the hot-gases into the heat exchanger core, proceeding through the core, and turning axially out the exhaust nozzle. A total pressure loss of 4.9 percent was estimated for the cold air in flowing through the headers and core. These high pressure losses were conterbalanced by a high effectiveness level of 0.90. Lower effectiveness level designs were evaluated which, in spite of lower pressure losses, provided equal thermal efficiency when considering the combined effects of pressure losses and regenerator effectiveness. With these designs, the desired engine nacelle allowances were greatly exceeded. Thus, only the 0.90 effectiveness regenerator was retained for further evaluation.

Theoretical evaluations, such as in reference 6, have shown a thermal efficiency advantage by combining other heat exchange processes with regeneration. Compressor intercooling, i.e., extracting heat from mid-compressor air, can theoretically supplement regeneration to provide up to an additional 3 percentage points in thermal efficiency in the absence of additional frictional losses or

... of this evaluation, these losses were considered and the alternate concept was evaluated. Parasitic pressure losses in the intercooler ducting and heat exchanger negated theoretical thermodynamic gains. Therefore, the simpler system of internal heat exchange, regeneration, was found to have the greater potential for conserving fuel in aircraft engines.

Non-Steady Flow Primary Combustion – In addition to heat exchange within the gas turbine, direct substitution of alternate processes has been considered in the past. Of particular note is the substitution of an intermittent-flow, constant volume process combustion in place of the standard steady-flow, nearly constant pressure process. Temperature – entropy diagrams are compared for the two cycles on Figure 7. The thermal efficiency advantage of constant volume combustion lies in the smaller entropy change during combustion (heat addition) and heat rejection which results in a lower average temperature of external heat rejection. A comprehensive study of the constant volume combustion process (7) analytically examined several combustion processes as related to feasible concepts involving constant cross-sectional area combustion tubes with valves at each end. As airflow proceeds through the tube, the outlet valve closes first, creating a hammer shock which propogates upstream to the tube inlet. The inlet valve is timed to close as the shock reaches its location reflecting the shock wave and causing a substantial increase in pressure level. Based on this evaluation, a 2 to 4 percent improvement in engine performance may be possible relative to a constant pressure combustor assuming a 20 percent pressure increase during combustion. In several independent test attempts to demonstrate constant volume combustion, the net pressure rise was limited to a few percent by valving dynamics. Mechanically, the non-steady combustor would substantially increase the engine complexity. Special ignition and fuel injection systems would be required together with the possible addition of extra bearings and a gear drive system. The application of constant volume combustion to future powerplants awaits an inventive concept which can demonstrate the performance potential with reliability and economy in the restrictive influence of the surrounding turbomachinery.

Cycle Compounding – The examination of the substitution of individual new processes within the general Brayton cycle concept leads naturally to the possibility of combining the Brayton cycle with other non-steady cyclic engines (8). For example, the use of an intermittent combustion cycle in place of the gas turbine burner can provide the efficiency benefits of the very high pressure and temperature levels achievable in this cycle with the large power output capability of the gas turbine. A notable example of this cycle is the Napier Nomad Compound Diesel Engine. To date, this engine, shown in Figure 8, has demonstrated a higher level of thermal efficiency than any other aircraft powerplant. This is remarkable considering that this engine was flight tested in 1954. On an equal output power basis, the Napier Nomad weighed approximately 2 to 3 times more than smaller first generation turboshaft engines while providing one third higher thermal efficiency. Recent studies indicate that this weight difference could not be significantly narrowed by proper selection of the pressure ratio of an advanced technology compressor and a modern rotary engine (9). Improvements in thermal efficiency could also be obtained by utilizing advanced technology turbomachinery and by insulating the rotary engine to limit the heat loss to 10 or 15 percent of the external heat input. By the 1990's, it is possible that the rotary engine, compressor, and turbine technologies will advance to the point where a compound cycle thermal efficiency 10 or 15 percent greater than the Napier Nomad level would be possible.

A composite thermal efficiency comparison of the Brayton, regenerative, and compound cycles is presented on Figure 9. Regenerative gas turbines and advanced compound gas turbines with projected 1985 technology levels can provide thermal efficiency levels similar to the simple gas turbine cycle. The regenerative gas turbine, efficient at low cycle pressure ratios, requires up to a 222°C (400°F) higher combustor exit temperature capability along with the added major engine element of a heat exchanger to equal the performance of a higher pressure ratio simple gas turbine. The very high pressure ratio compound cycle requires the complexity of the non-steady flow engine cycle. Of the three concepts, the advanced simple gas turbine cycle has the greatest potential for increasing thermal efficiency with low weight, low cost, and high reliability needed for air transport applications.

3.2 Advanced Fuel Conservative Propulsor Possibilities

Efficient conversion of available power into useful thrust shares equal importance with efficient power generation in conserving fuel. Thrust output theoretically can be increased by minimizing the air velocity increase induced by a propulsor disk for a given amount of available power. Since thrust output is proportional to the product of the air mass flow rate and the velocity increase through the propulsor, propulsor diameter must be increased proportionally with the square root of the mass flow rate to produce the higher thrust.

Trends of propulsor diameter and ideal thrust capability, shown as propulsive efficiency, were derived for a series of possible propulsors as shown in Figure 10. The calculations assume isentropic thrust conversion processes and no viscous losses. Turbofan diameters include an allowance for the fan cowl. Propeller diameters are based on information contained in references 10 and 11. A large gap exists between the two conventional propulsor diameters in which a major propulsive efficiency improvement potential exists. Two possible unconventional propulsors in this region were considered – a shrouded fan and an advanced propeller.

The selection of these concepts for evaluation was based on examination of the effects of irreversibilities on the performance potential of two conventional propulsors at the Mach 0.8 flight speed. Internal engine and external loss effects are illustrated in Figure 11. The propulsive efficiency of the advanced turbofan is 20 to 30 percentage points below the ideal over a fan pressure ratio range of 1.7 to 1.3. An additional 2 percentage point reduction could be expected with 1975 technology.

Rapidly increasing installation losses with fan pressure ratio reduction offset the ideal propulsive efficiency increase for the conventional turbofans. Turboprop propulsive efficiency with 1950's technology propellers is lower than the turbofan levels principally because of low propeller efficiency associated with blading compressibility losses. In an attempt to utilize a larger percentage of the ideal propulsion efficiency in the small diameter range, the benefits of reducing low pressure ratio fan installation losses, and improving propeller technology relative to the present operating levels were addressed.

Shrouded Fan Concept – The three major factors affecting low pressure ratio fan propulsive efficiency potential are internal ducting pressure losses, nozzle losses, and fan cowl drag. Sensitivity of propulsive efficiency to the three factors was calculated and is shown in Figure 12. The propulsive efficiency sensitivity to ducting and nozzle losses increases by a factor of five or greater between the selected pressure ratios of 1.7 and 1.1. Fan cowl drag could easily increase by a factor of 4.0 between these fan pressure ratios and for conventional nacelle geometry because of the progressively larger wetted area associated with the larger fan diameter. Therefore, a short thin nacelle, or shroud, would appear advantageous in utilizing the theoretical fuel savings potential of low pressure ratio fans. A 1.1 pressure ratio fan was selected to investigate the fuel savings potential relative to a 1.7 pressure ratio conventional fan. Adiabatic stage efficiency of the 1.1 pressure ratio, 12 bladed fan was estimated to be within 0.5 percent of the 1.7 pressure ratio, 42 bladed fan. A cowl, or static shroud, with a length-to-maximum diameter ratio of 0.5, less than one-third the ratio of the conventional fans, was needed with the 1.1 pressure ratio fan with minimum desirable inlet length, blade-to-stator acoustic spacing, and nozzle ducting allowances. The resulting fan configuration, shown in Figure 13, includes variable pitch fan blades which are needed for thrust reverse during landing and which can improve the fan stability margin during take-off operation where the sharp lip inlet imposes a highly distorted fan face flow field. Relative to the 1.7 pressure ratio fan installation, the shrouded fan is estimated to have an 80 percent lower internal parasitic pressure loss in the combined inlet and duct, a 0.2 percent higher nozzle velocity coefficient, and equal external fan cowl drag. With these differences, the 1.1 pressure ratio fan is estimated to offer up to a 9 percentage point higher propulsive efficiency than the 1.7 pressure ratio, conventional fan installation.

Advanced Propeller Concept – The primary indicator of the usefulness of an unshrouded propeller is propeller efficiency which directly relates the thrust output to the shaft power input. The achievement of high efficiency at contemporary flight speeds and altitudes is the critical factor in assessing the value of this propulsor.

Current operational propellers exhibit very high efficiency levels up to a flight cruise speed of Mach 0.65. At higher flight speeds, blading compressibility losses increase sharply to cause the performance downturn indicated on Figure 14. Thin cross-section, lightly cambered two bladed research propellers, tested in 1950, demonstrated high efficiency at the higher flight speeds (10). The thinnest model had an 80 percent measured efficiency at Mach 0.8. However, these models were structurally inadequate. Their demonstrated efficiency combined with composite structural technology form the basis for advanced propeller characteristic projections taken from reference 10.

The two blade efficiency data were converted to an eight blade configuration by established techniques which halved propeller diameter. Cruise efficiency was estimated to be 73 percent for the smaller diameter eight bladed propeller using ideal efficiency trends with diameter and blade number. The application of supercritical blading sections, a contoured nacelle shape, and blade tip sweep were each calculated to improve efficiency by over two percentage points. Supercritical airfoil improvements were extracted from limited published data of Whitcomb airfoils. Wind tunnel tests of several propeller-nacelle model configurations have shown substantial reductions in blading section compressibility losses through blade root flow retardation by increasing nacelle diameter. The high solidity root sections of an eight blade design could be expected to be especially sensitive to this effect. In accomplishing the improvement, care would have to be taken to avoid high nacelle drags. Limited aerodynamic theory indicates that sweep should be useful at the blade tips where the relative velocity is slightly supersonic. If all the indicated aerodynamic advances were achieved at the Mach 0.8 cruise speed, the projected eight bladed propeller efficiency would be 80 percent. The achievement of this propeller efficiency translates into a cruise propulsive efficiency level 14 percentage points higher than a 1.7 pressure ratio conventional fan taking into account the propulsor efficiencies, internal ducting losses, and fan cowl drag.

In reference 11, modern spar and shell blade construction consisting of a flattened metal tube spar, a composite airfoil shell and a titanium leading edge sheath for foreign object damage and erosion protection, is projected to meet the aerodynamic requirements with structural adequacy. Uncontained blade fragmentation, of obvious concern to the operator, is not considered to be a major problem, according to the manufacturer, based on the fifty million hours of turboprop operation without a single in-flight blade fragment separation.

An advanced propeller was selected for consideration with major characteristics as shown in Figure 15. The combination of multi-blading and small diameter should ease the mount stiffness and nacelle length requirements and provide more freedom in engine positioning within fuselage and ground to blade tip clearance constraints.

3.3 Installed Advanced Propulsion Systems

Installation studies were carried out for the turbofan, shrouded fan, and turboprop propulsion systems to determine the relative installed cruise propulsive efficiency. Advanced gas generators with high cycle pressure ratios and high maximum combustor exit temperature levels were selected to provide near maximum thermal efficiency.

The turbofan gas generator maximum combustor exit temperature was set 111°C (200°F) lower than the turboshaft engine cycle used to drive the advanced propeller or the shrouded fan. The lower temperature, 5 stage fan drive turbine provided sufficient work output at high efficiency to avoid the need for reduction gearing between the fan and turbine. The turboshaft engine cycle pressure ratio was reduced somewhat from the turbofan value to produce a comparable gas path height in the critical compressor exit region for equal cruise thrust capabilities. In this manner, comparable compressor and turbine blade tip clearances and losses could be assumed. The somewhat different turbofan core and turboshaft cycles provided nearly equal thermal efficiency levels as shown in Figure 16.

Offset compound idler gear drive systems were selected for the shrouded fan and turboprop systems with respective gear ratios of 8.5:1 and 7.8:1. A gear system efficiency of 99 percent was assumed in the evaluation for both cases.

Installation arrangements were worked out for the three systems as shown in Figure 17. The turbofan was assumed to be a conventional under-the-wing, pylon mounted arrangement. The fan cowl length is 1.55 times the maximum cowl diameter to provide adequate length for efficient inlet diffusion and sufficient acoustically treated wall area to attenuate fan source noise to FAR Part 36 minus 10 EPNdB levels at take-off, sideline, and approach conditions. Reverse thrust is provided by a cascade thrust reverser in the fan stream only. The shrouded fan and turboprop systems were treated as under-the-wing gas generator arrangements which are gloved to the wing. Complete justification for this selection would require further study. However, it appears that this arrangement can provide adequate propulsor tip to ground clearance, drag and internal inlet pressure recovery comparable to over-the-wing installations, and readily accessible engine modules. Thrust reverse is accomplished by altering the pitch of the two propulsors, blading ball-race retention and hydro-mechanical (mechanical) pitch change system have been assumed for this purpose. Integrated propulsion and gas generator controls represent an ideal application of advanced digital electronics which can provide multi-variable control within a small package.

The axial distance between the wing quarter chord and propeller planes was set at a value of approximately 1.0 to minimize the tendency for nacelle whirl flutter and vibration transmission to the cabin. Thin inlets were placed well out in diameter from the blade roots in the turboprop to benefit from the pressure rise through that section of the propeller. A total pressure recovery to the gas generator face of 1.0 is possible by carefully contouring the spinner and inlet for minimum loss. The maximum nacelle diameter on the turboprop was set equal to 35 percent of the propeller diameter to provide sufficient back-pressure and avoid blade root choking. While an attempt has been made to account for all of these phenomena, it is recognized that much additional analyses and testing is required to weigh the many factors involved in selecting a final installation arrangement.

Installation losses were estimated for each of the systems in order to compare installed propulsive efficiency potential (Figure 18). Blading efficiency levels, internal ducting losses, and fan cowl drags were taken from evaluations described earlier in this paper. Gear system losses, propulsor-drive turbine losses and the remainder of the installation drag were calculated for the selected installations. The large gains calculated for the advanced propeller, although preliminary, indicate that installation loss effects are small in relation to the improved propulsor performance (Figure 18). The shrouded fan is shown to be a third less effective in increasing propulsive efficiency than is the advanced propeller.

The advanced propeller shows adequate promise to suggest further detailed analytical and empirical evaluation. The shrouded fan is viewed as a possible contingency propulsor pending the outcome of advanced propeller testing planned in 1976. Although the shrouded fan has the potential for relatively large fuel savings relative to the conventional turbofan, the potential is highly sensitive to system parasitic losses associated with ducting the air through the propulsor. Major aerodynamic and structural questions related to the large diameter shroud also remain unanswered at this time. The remainder of this paper is therefore devoted to evaluation of the turboprop and conventional turbofan systems.

4.0 POTENTIAL ENERGY SAVINGS AND ECONOMIC BENEFITS

Advanced study aircraft were taken from reference 4 to determine the fuel savings and direct operating cost reductions possible with advanced turbofans and turboprops. Domestic and intercontinental 200 passenger transport aircraft, with respective design ranges of 5,556 km (3000 n.mi) and 10,186 km (5500 n.mi), were evaluated. Aircraft characteristics in both cases include high aspect ratio wings, supercritical aerodynamics, and advanced lightweight composite structural technology. The turbofan powered domestic transport was studied as a three engine configuration; the domestic turboprop and both intercontinental aircraft were studied with four engines. The trijet consisted of two under-the-wing pylon mounted engines and a tail mounted engine. The four engine aircraft consisted of wing mounted arrangements with nacelle placements based on engine-to-engine interference and cabin comfort considerations.

Inboard turboprops were placed to provide 0.8 of the propeller diameter clearance between the fuselage and the blade tips. In reference 10, this placement was indicated to provide cabin noise levels comparable to turbofan aircraft by adding fuselage wall treatment equal to 0.25 percent of the aircraft gross weight. A blade tip clearance between inboard and outboard propeller of 0.33 propeller diameter was assumed. The aircraft wing weight, tail size and weight, and landing gear weight calculations were all based on these engine placement criteria.

Propulsion system weight calculations were also carried out for the selected engine installations resulting in the comparison shown in Figure 19. The weights were taken from several sources. Gas generator weights are based on detailed analysis of the two cycles. Propeller and gearbox weights were taken from reference 12. Turbofan nacelle and pylon weights were based on unpublished correlations and assume the use of composites. Turboprop nacelle weight estimates were taken from correlations of current systems; results were found to agree well with several other independent studies.

With the definition of primary cycle thermal efficiency, installed propulsive efficiency, and the weight of the engine installation and aircraft, it was possible to calculate fuel requirements. Fuel consumption was calculated for the domestic and intercontinental aircraft on respective average flight stage lengths of 1296 km (700 n.mi.) and 3,704 km (2000 n.mi) and compared with present modern turbofan technology to assess the long term fuel savings potential by conducting mission profile analyses of "rubber" aircraft. The results of this assessment, Figure 20, indicate improvements ranging from approximately 10 to 35 percent to be possible with advanced propulsion systems. The cruise thrust specific fuel consumption (TSFC) improvement which is inversely proportional to the product of thermal and propulsive efficiencies, is the predominant factor in the resultant fuel savings with weight savings accounting for less than 1 percentage point of the benefit. The turboprop is more effective in saving fuel at shorter flight lengths where a large portion of fuel is consumed during climb. The inherently greater take-off to cruise thrust capability of the low pressure ratio propulsor provides both fast and efficient climb operation to significantly increase the fuel savings differential over that indicated during cruise.

The higher turboprop take-off thrust potential can also reduce noise level perceived at the measuring point. The turboprop aircraft were calculated to be approximately 183m (600 feet) higher than the turbofan aircraft at the 5.63 km (3.5 mile) measurement point. The propeller noise levels were estimated to be at a FAR Part 36 minus 12 EPNdB level with projected acoustic technology amounting to a 5 EPNdB noise reduction relative to present technology. The achievement of comparable levels in gas generator low frequency noise and 1.7 pressure ratio fan noise would require approximately a 4 EPNdB reduction below current levels. These reductions, which are possible in the future, can also provide approach and side-line condition noise levels below FAR-10 EPNdB. With advanced engine acoustics, therefore, the turbofan and the turboprop systems could both achieve a noise level 10 EPNdB below current FAR regulations.

In order to compare the economics of the two propulsion systems, acquisition and maintenance costs were estimated using both published and unpublished data. Similar costs were obtained for the turboprop system (gas generator, gearbox, propeller, pitch control, and engine control) and the turbofan system (gas generator, fan, and engine control). Nacelle and turbofan reverser costs were added in separately to arrive at total installed costs. The wider band of maintenance costs for the turboprop account for the added uncertainties in the free turbine, gear set, and propeller. Aircraft direct operating costs were calculated based on this information and the fuel costs shown in Figure 21 for comparison with present turbofan technology. The advanced propulsion systems are likely to be more costly to acquire and maintain. Higher maintenance costs, in particular, can erode the direct operating cost benefits with lower fuel consumption. The trends will depend on the relative cost escalations of fuel, labor, and materials. Nevertheless, in either future turbofan or turboprop systems, maintenance costs must receive attention equal to fuel savings potential to assure the economic incentives necessary for airlines acceptability.

5.0 CONCLUDING REMARKS

- Projected advances in gas turbine component technology can produce a 10 percent or greater energy savings potential relative to present turbofan technology without resorting to unconventional components such as heat exchangers or intermittent combustion processes. The benefits of these technological advancements apply equally to turbofan or turboprop systems.

- The advanced turboprop system presently shows the greatest potential for fuel conservation. This potential is tied to the capabilities of an advanced propeller at contemporary flight speeds. Aerodynamic, acoustic, and structural verification is critical to the further pursuance of the advanced propeller system.

- Numerous assumptions were made in the turboprop system integration evaluation which require additional substantiation. Propulsion system integration studies by airplane manufacturers are recommended together with a propeller technology development program to establish a firm technical base on which to assess the concept further.

- Future transport aircraft engine fuel savings can produce operating economic benefits with careful design control over engine maintenance requirements.

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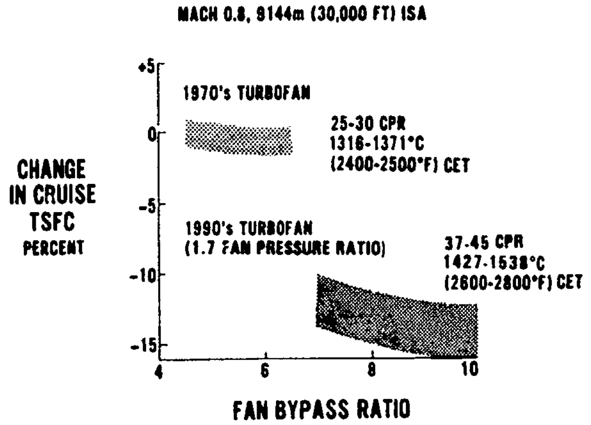
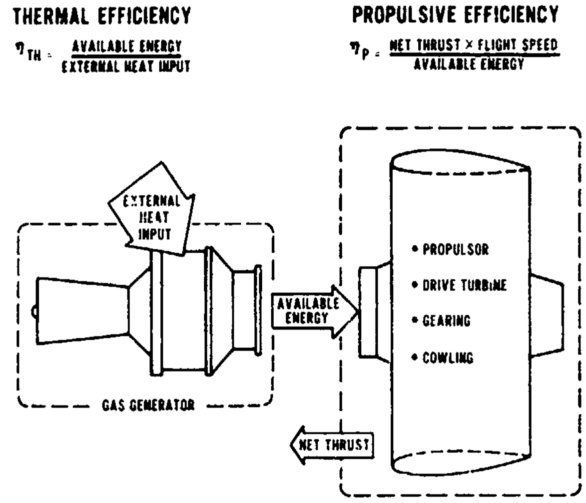


Figure 1 *Future conventional fuel conservative turbofan cruise TSFC potential.*



$\eta_{TH} = \frac{\text{AVAILABLE ENERGY}}{\text{EXTERNAL HEAT INPUT}}$ $\eta_P = \frac{\text{NET THRUST} \times \text{FLIGHT SPEED}}{\text{AVAILABLE ENERGY}}$

Figure 2 *Powerplant efficiency definitions.*

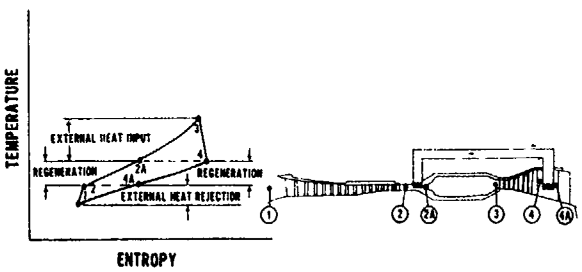


Figure 3 *Gas turbine regenerator concept*

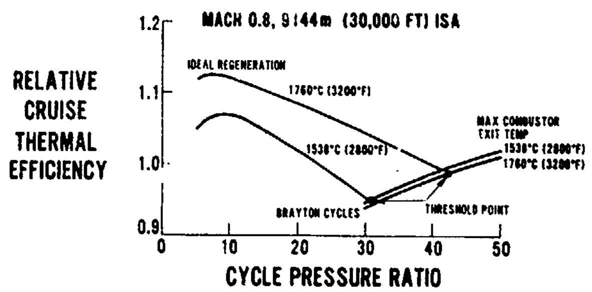


Figure 4 *Ideal regenerative Brayton cycle thermal efficiency*

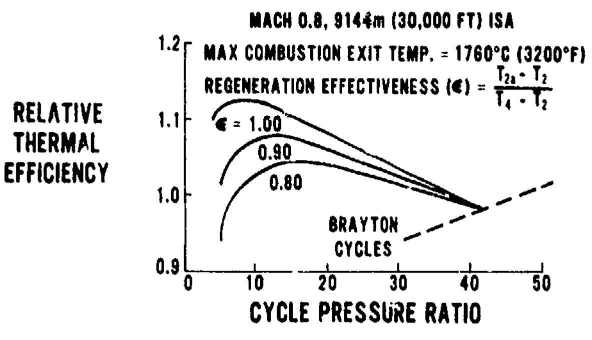


Figure 5 *Thermal efficiency trends with regeneration effectiveness*

• 90% EFFECTIVENESS
• 19.5% TOTAL PRESSURE LOSS

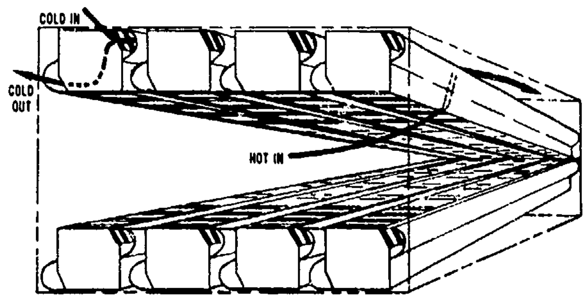


Figure 6 *Counter-flow plate-fin recuperator concept.*

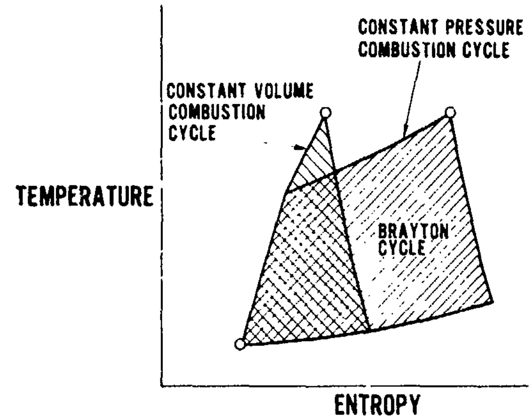


Figure 7 *Comparison of constant pressure and constant volume combustion processes*

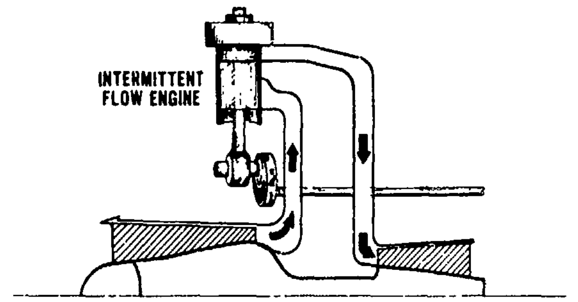


Figure 8 *Napier Nomad compound diesel engine*

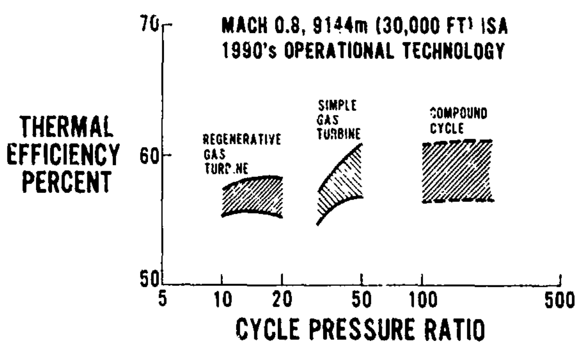


Figure 9 *1990's operation technology primary cycle comparison*

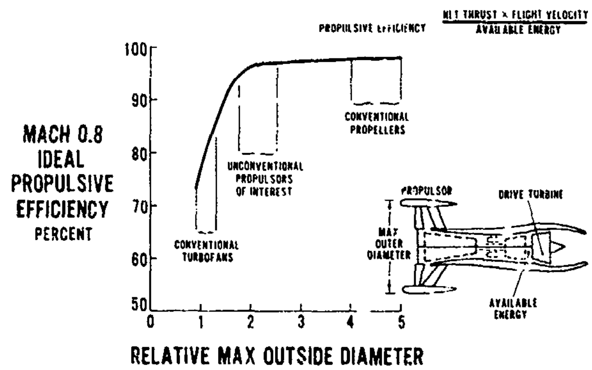


Figure 10 *Ideal propulsive efficiency trends with diameter for constant thrust*

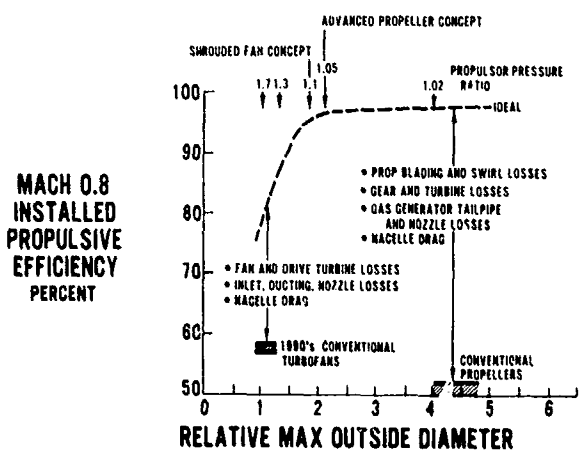


Figure 11 *Irreversibilities and propulsive efficiency trends*

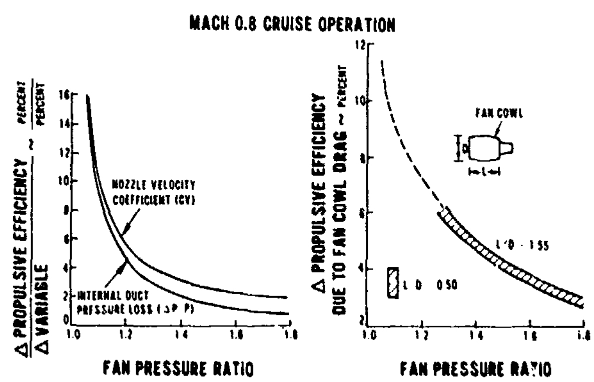


Figure 12 *Sensitivity of propulsive efficiency to selected irreversibilities*

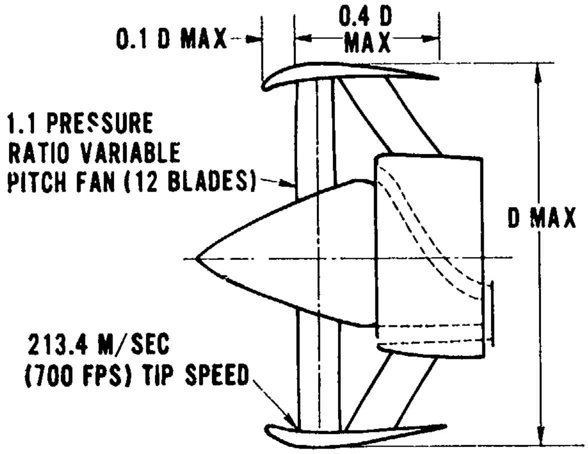


Figure 13 Shrouded variable pitch fan propulsion concept

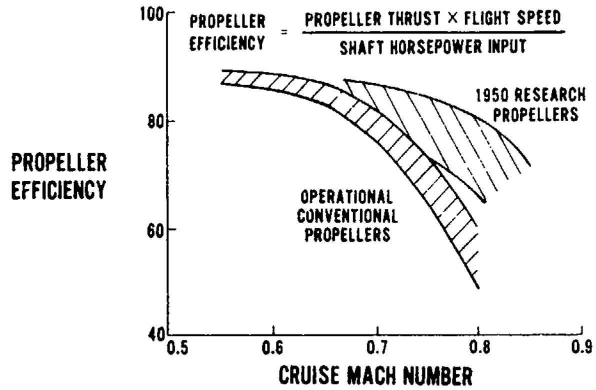


Figure 14 Propeller efficiency trends with flight speed

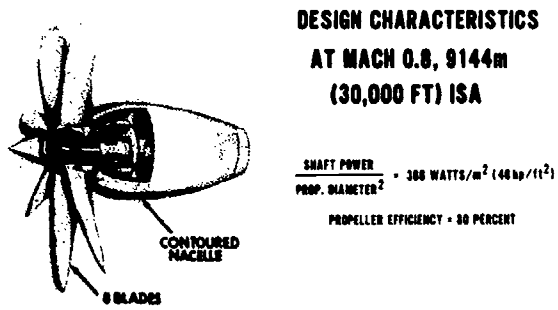


Figure 15 Prop-fan propulsion concept

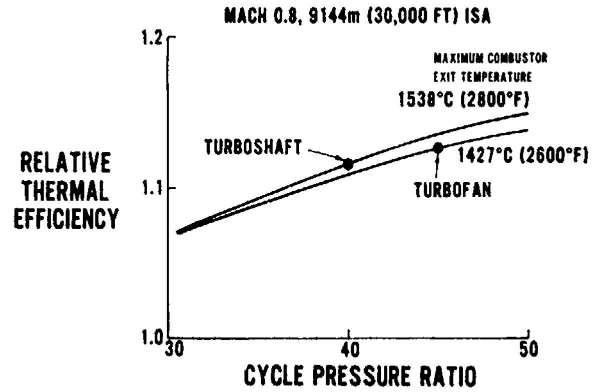


Figure 16 Thermal efficiency comparison of advanced technology turboshaft and turbofan engines

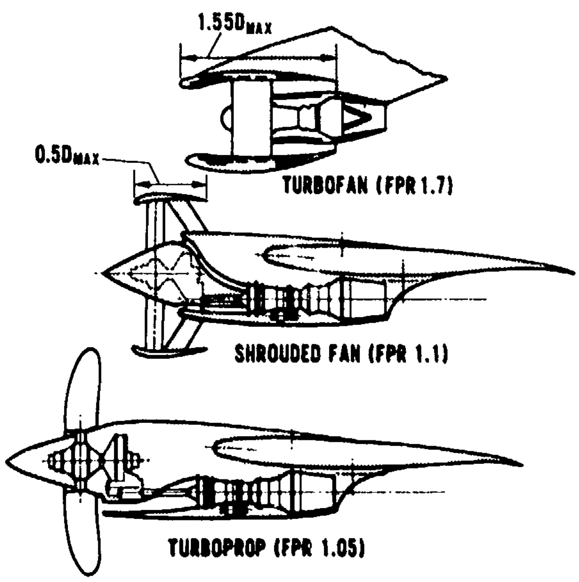


Figure 17 Turbofan, shrouded fan, and turboprop installations

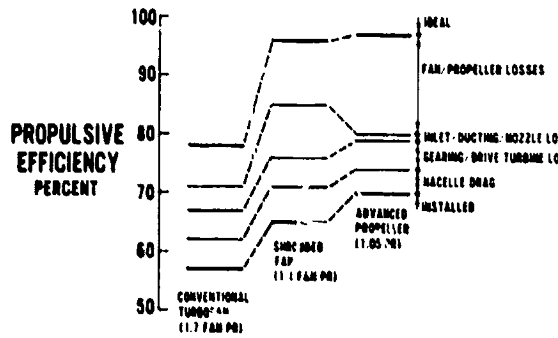


Figure 18 Installed propulsive efficiency at Mach 0.8 and 9144m (30,000 ft) ISA

	TURBOFAN	TURBOPROP
CYCLE PRESS RATIO	45:1	40:1
MAX TEMP	1427°C (2600°F)	1538°C (2800°F)
PROPULSOR PRESS RATIO	1.7	1.05

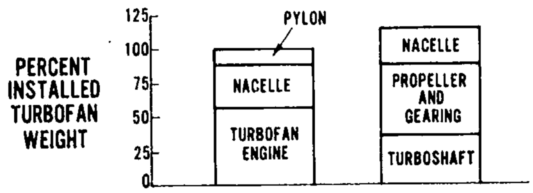
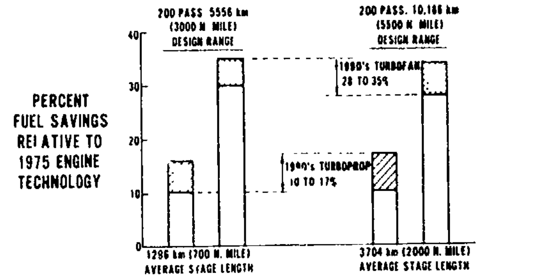
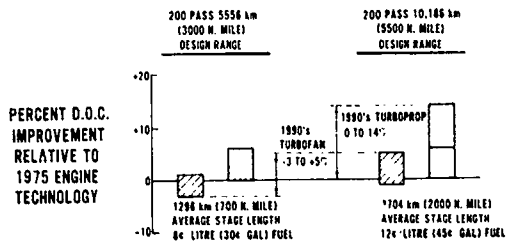


Figure 19 *Advanced technology turbofan and turboprop weight comparison based on equal cruise thrust capability at Mach 0.8 and 9144 m (30,000 ft) ISA*



CHANGE RELATIVE TO 1975 TURBOFAN	1990's TURBOFAN	1990's TURBOPROP
MACH 0.8 CRUISE THERMAL EFFICIENCY ~ PERCENTAGE PTS	+4 TO +6	+4 TO +6
MACH 0.8 CRUISE PROPULSIVE EFFICIENCY ~ PERCENTAGE PTS	+2 TO +3	+15 TO +16
INSTALLATION WEIGHT ~ %	-15	-4

Figure 20 *Projected fuel savings with future propulsion systems*



CHANGE RELATIVE TO 1975 TURBOFAN	1990's TURBOFAN	1990's TURBOPROP
ENGINE PRICE PER ENGINE ~ 1000 DOLLARS	+100 TO +150	+100 TO +150
ENGINE MAINTENANCE PER ENGINE ~ DOLLARS PER FLIGHT HOUR	+15 TO +25	+15 TO +34

Figure 21 *Direct operating cost with future energy efficient propulsion systems*

DISCUSSION

D.R.Higton

In our studies of the advanced propeller at NGTE we have had difficulty in estimating the impact on wing aerodynamics of placing such a propeller ahead of a modern wing section. Would you please comment on how significant you would expect the interaction to be?

Author's Reply

Yes, I have a few comments. We have had the same difficulty, frankly, but we have asked various aerodynamicists about their thoughts in terms of swirl effects on super critical airfoils. And we got some very interesting responses, both for and against. There is actually one school of thought that believes swirl effects do not need to be a problem but could induct positive interference. There is another school of thought which says that the propeller slipstream is going to be a tremendous problem and will possibly negate a good portion of the potential turboprop cycle benefits. In the evaluation reported in my paper, the increased dynamic pressure in the propeller slipstream was taken into account in bookkeeping the drag. No attempt was made to account of the effects of slipstream swirl on drag.

J.P.Vleghert

Have you investigated noise production of the turboprop?

Author's Reply

There is some work going on looking at both near field noise and also far field noise. What is your concern?

J.P.Vleghert

The question is concerned with far field noise.

Author's Reply

We have made noise estimates both for the advanced turbofan and the advanced turboprop. Presently, we are working towards a FAR-36 minus 10 EPN dB level as a future noise goal. Based on initial study results we are predicting essentially equivalent noise output for the turbofan and turboprop systems. In order to achieve the FAR-36 minus 10 EPN dB level, advancements in acoustics amounting to 4 or 5 dB noise reduction is required in either propulsion system.

J.F.Chevalier

Est-ce que cette différence n'a pas favorisé le turboprop et est-ce qu'elle n'aurait pas favorisé également le fan caréné?

Author's Reply

Yes, that's right. The combustor exit temperature of the turboprop cycle was set higher than that of the turbofan based on mechanical considerations. A speed reduction gear is required between the low speed propeller and high speed drive turbine on the turboprop. In the case of the turbofan, the need for a reduction gear is marginal. Therefore, we elected to reduce the turbofan cycle combustor exit temperature by 200°F relative to the turboprop to eliminate the need for the complex gear with its unknown reliability and maintainability. This resulted in an estimated 1 or 2 percent SFC penalty with the more conservative thermodynamic cycle, however.

J.F.Chevalier

Mais il est probable que vous aviez également mis dans la comparaison le "shrouded fan" à un grand taux de dilution, il aurait été meilleur que le turbofan, sans doute.

Author's Reply

Yes, that's correct.

MULTI-MISSION USES FOR PROP-FAN PROPULSION

A. H. JACKSON JR. AND B. S. GATZEN
HAMILTON STANDARD, DIVISION OF UNITED TECHNOLOGIES CORPORATION

SUMMARY

The severe impact of escalating fuel prices on aircraft operating costs has accelerated serious study of advanced propulsion systems with potential for major reductions in fuel consumption. The potential of the advanced turboprop has emerged as particularly attractive. Using recent technical advancements in aerodynamics and structures and configured to very high power loadings, advanced turboprops or Prop-Fans should achieve marked fuel savings when compared to the turbofan, even at the high subsonic cruise speeds that have become traditional in recent years. The Prop-Fan will display considerable operational versatility over a wide spectrum of potential subsonic missions, both military and commercial. Its ability to maintain excellent performance levels with large variations in aircraft speeds and altitudes results from the high rotor efficiencies gained with the use of variable fan blade angle. The use of Prop-Fan powered aircraft with their large fuel savings also should be accompanied by favorable environmental characteristics, comfort levels, and high levels of reliability and safety.

INTRODUCTION

Few recent events have attracted such worldwide attention as the 1973-1974 Arab oil embargo. Among its effects were the sudden realization of the coming permanent scarcity of petroleum resources and the inexorable escalation of oil prices that this shortage portends. Of the major energy users most susceptible to the effects of oil shortages, aviation stands out as particularly vulnerable, since nowhere on the horizon does there appear the prospect of alternative energy sources that have a reasonable potential for future economic viability. To maintain its prosperity, therefore, this industry must focus attention of both its private and public sectors on finding step improvements in fuel efficiency over that typical of today's turbojet/turbofan powered aircraft.

Responding to requests from the U.S. Congress in early 1975, the National Aeronautics & Space Administration (NASA) greatly broadened its studies to assess the potential impact of a number of advanced technologies on the fuel consumption of a future generation of commercial passenger transports. It concluded that six areas display considerable promise for significant improvements and should receive Government support for both research and follow-on advanced development, the latter provided their early promise continues to be sustained. Among the six, advanced turboprop propulsion displays perhaps the greatest potential for major fuel savings.[1]

The general adoption of the turbojet/turbofan equipped aircraft by the world's major airlines in the late 1950's and early 1960's was made at a considerable sacrifice in fuel productivity (fuel used per seat mile). The simultaneous realization of higher flight speeds up to 0.8 + Mach number (MN), however, and the introduction of aircraft larger than was then thought commercially practical with turboprop designs, proved so productive in terms of specific operating costs that an explosive growth in the aircraft industry resulted. Fuel costs fifteen years ago were only one-third of what they are today and since they were a much lower percentage of the overall costs of airline operation, did little to inhibit this revolutionary development in air transportation.

The recent reawakened interest in the turboprop has called forth the vision of the lower cruise speeds which characterized past designs and has stimulated concern that the process by which all parties benefitted during the 1960's could be reversed. It has been speculated that a decrease in flight speed would adversely affect passenger load factors, particularly for those operators using slower turboprops in competition with turbofan equipment. The validity of this concern has been sustained by the recent NASA-sponsored United Airlines passenger survey, which concludes that lower speeds would not be tolerated by the general passenger without an associated fare benefit.[2] Additionally, the modern air traffic control system is largely based on the operating speeds of today's airline fleet. Introduction of a mixed speed fleet including significant numbers of relatively low speed aircraft would likely result in added operating costs of a magnitude felt by some to be prohibitively high. This concern is frequently expressed by potential military as well as airline users of turboprop equipment, although, as far as can be determined, the subject has not been studied in great detail.

Based on such considerations, it has been concluded generally that even with significant future escalation in fuel costs, new passenger transport type aircraft designed to operate at cruise speeds significantly less than 0.8 MN probably would suffer adversely from higher direct and/or indirect operating costs. This assessment applies to both military and civil aircraft, except for those that have dedicated missions at significantly lower cruise speeds and/or operate outside the heavily travelled air traffic controlled airways. <u>Maintaining 0.8 or near 0.8 MN cruise speed is now conceded to be essential for any future advanced turboprop powered equipment which is designed to have a significant impact on our air transportation system.</u> To realize such performance, however, propulsion system technology advancement will be required as, with the possible exception of certain Russian designs, even the most advanced of past propeller configurations have not provided the levels of installed propulsive efficiencies necessary for these high cruise speeds. If the envisioned technology development is achieved, it is possible that advanced turboprop propulsion will be adopted for a wide range of subsonic aircraft missions.

ESTABLISHING THE PROP-FAN CONCEPT

To meet the demands of 0.8 MN cruise speeds, an advanced turboprop propulsion concept called the Prop-Fan has been established. While the aerodynamic details which support the projected Prop-Fan performance have been described in some depth in Reference 3, it would be useful to trace the steps by which the Prop-Fan configuration has evolved.

It is first important to note that the thrust specific fuel consumption (TSFC) of the free-air turboprop propulsion system can be compared directly to that of the ducted, high bypass turbofan engine producing equal cruise thrust by ratioing the energy being introduced at the power turbine of each. This energy can be considered all shaft power, assuming a negligible impact of different energy splits between the turbine and nozzle. The shaft horsepower (SHP) is related to installed propulsive efficiency (η_I) for either powerplant by the relationship SHP = TV_0/η_I. η_I is equal to the uninstalled propulsive efficiency (η_{NET}) minus the installation losses due to encasing the engine and mounting it to the aircraft. η_{NET} is equal to the ideal momentum efficiency (η_m) minus the uninstalled losses which include such items as blade tip losses and airfoil drag, unrecovered swirl energy, internal duct drag, etc. Finally, the ideal momentum efficiency accounts for the losses resulting from mixing the air exiting from the rotor with the free stream air.

To sum up, for either the Prop-Fan or high bypass turbofan, the TSFC $\sim \eta_I = \eta_{NET}$ - installed losses = η_m - installed and uninstalled losses. Since it is generally agreed that the installed losses for either the high bypass ratio turbofan or the Prop-Fan probably can be controlled to rough equivalency, an accurate view of the relative fuel consumption potential can be garnered from examination of the uninstalled propulsive efficiencies, η_{NET}, which in the case of the turboprop is sometimes labelled propeller efficiency, η_p.

Figure 1 summarizes the effects of advances in propeller aerodynamic technology which evolved in the post-war period. Almost without exception propellers designed for reciprocating engine powered aircraft in this era suffered from severe losses in propeller efficiency at flight speeds much over 0.5 MN. This was largely due to poor performance at high subsonic rotor tip speeds of airfoil sections highly cambered for peak takeoff thrust and thickened substantially to contend with the very high vibratory loadings originating within the engine and transmitted to the propeller through the prop shaft. As air transportation expanded and runways were lengthened, the need for highly cambered airfoils was sharply reduced. Also, and perhaps more importantly, the adoption of the turbine engine completely eliminated the engine induced vibratory loads, making it possible with the introduction of the Lockheed Electra to reduce typical section thickness ratios at the blade tip from 6% or 7% down to 2-1/2%, with comparable thickness reductions along the entire blade span.

The design of the Lockheed Electra/P3 propeller, which was subsequently to be produced in considerable volume and is still in production today, was laid down in 1955. At that time, wind tunnel research was under way on even more advanced blade designs incorporating still thinner, more lightly cambered airfoils which avoided severe efficiency degradation to beyond 0.8 MN. The last of these tests was completed in 1958, at a time when interest in turboprop equipment was waning while industry chose the path of jet propulsion for most of its subsequent needs. The proven potential for a major improvement in turboprop propulsion efficiency went largely unrecognized.

With the stimulation of the NASA interest during the studies it sponsored in 1975, the old high speed propeller wind tunnel data was disinterred and re-examined. This review revealed uninstalled propulsive efficiencies to be quite attractive compared to that characterizing current turbofan equipment. It was suspected, however, that full-scale versions of the test models, designed for conventional power loadings, could have a number of structural and installation problems in the areas of weights, clearances, vibrations, etc. To overcome these, a serious attempt to reduce the size and mass of the rotor appeared necessary. Studies followed examining the feasibility of increasing propeller pressure ratio by reducing diameter, but only to that point (55 to 60% of turboprop diameter) where serious penalties in ideal efficiencies (η_m) would be avoided (see Figure 2) and the large performance margin over the turbofan, even after absorption of installed and uninstalled losses, still would be maintained.[4] The resultant design was characterized by much higher power loadings (37-1/2 SHP/D^2 vs 12 SHP/D^2 at cruise speed conditions); higher fan rotor solidities (1600 TAF vs 600 TAF); 8 blades vs 4 blades; thin, very low cambered airfoils along the blade span duplicating the high speed model blades tested in the 1950's; and substantially reduced rotor and gearbox weights.

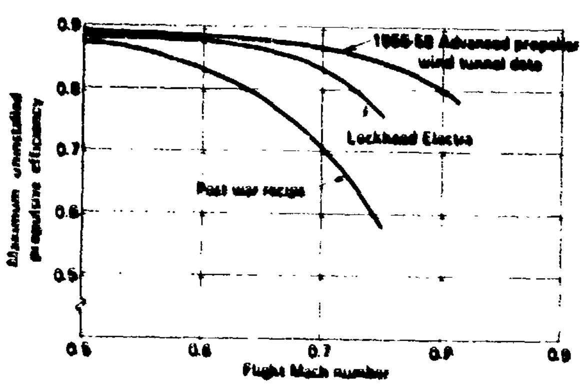


Figure 1. Effect of Flight Mach Number on Maximum Propeller Efficiency

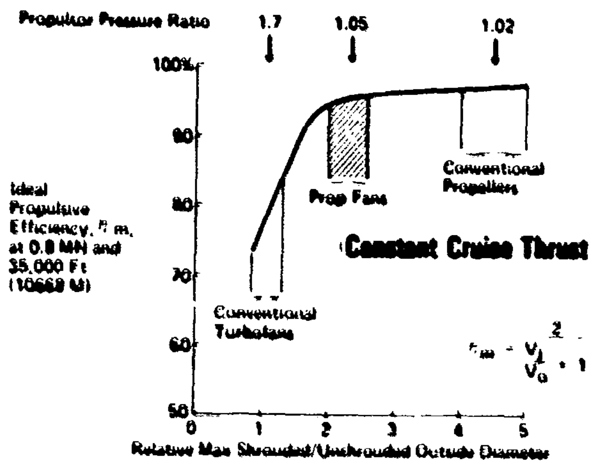


Figure 2. Ideal Propulsive Efficiency Trends

Extrapolated performance from the baseline high speed wind tunnel data suggested that, largely because of increased rotor slipstream swirl losses, the propeller net efficiency (η_{NET}) of a highly loaded Prop-Fan design would fall substantially below the 80% peak efficiency obtained with the earlier conventionally loaded designs. The 73% calculated net efficiency (see Figure 3) at the desired point design power loading would erode approximately 50% of the propulsive efficiency advantage forecast for the advanced turboprop over the turbofan, and was judged unsatisfactory. To counterbalance the increased swirl losses without resorting to the increased complexity of counterrotating rotors required an attack on compressibility losses, the other major area of efficiency degradation. A number of configuration changes were postulated to address this problem and an analytical/experimental program was initiated to provide the means of incorporating these modifications into hardware design. Major variables to be investigated included blade sweep, advanced airfoils, and various centerbody shapes. It was projected that as a result of the series of experimental wind tunnel tests planned by NASA, it should be possible to restore the performance to 80% efficiency, while maintaining the other advantages of the desired cruise power loading. Using the projected 80% Prop-Fan efficiency as the base point, efficiency characteristics across the operating envelopes of typical 0.8 MN transports were derived by Hamilton Standard and furnished to the engine and airframe manufacturers for their studies comparing aircraft powered by equivalent technology level turbofan and Prop-Fan propulsion. Additionally, Prop-Fan weights, noise characteristics, and economic data were made available for study purposes.

An ultimate comparison of fuel consumption for Prop-Fan vs turbofan powered aircraft is dependent upon the final installed efficiencies of the two propulsion systems which include the losses not encompassed in the isolated, uninstalled propulsive efficiency characterizations. The prime airframe manufacturers possess the best capability for determining the effect of these losses, which include not only the scrubbing drag on the nacelle and interference drag with the wing, but in the case of the Prop-Fan, the added wing drag resulting from the rotor wash and the offsetting, possibly beneficial swirl energy recovery realized by the wing's inherent flow straightening capability. It is likely, too, that to obtain optimum results, special design techniques to integrate the Prop-Fan with the wing will prove attractive. Studies made to date suggest that the propulsion system installation performance losses ultimately should be no greater than those assessed against the high bypass turbofan. On this basis, Figure 4 is considered a fair representation of the comparative installed efficiencies of the turboprop, Prop-Fan, and the conventional high bypass turbofan.[5] For any given Mach number, the differences in installed efficiencies represent the approximate differences in fuel consumption of comparably sized powerplants.

FUEL EFFICIENCY STUDIES

The baseline Prop-Fan and its associated characteristics including performance, acoustic signature, etc., were employed first in the NASA-Lewis sponsored unconventional engine studies as reported in References 4 and 6, and subsequently in the NASA-Ames sponsored RECAT aircraft studies which were reported in References 7 and 8. More recently Boeing has been conducting a two-engine commercial medium haul aircraft study but results were not officially available at the time this paper was being prepared.

A photograph of the Lockheed Prop-Fan powered RECAT aircraft model is shown in Figure 5. This aircraft was designed to carry 200 passengers, employed a design range of 1500 nautical miles (nm) and displayed fuel savings greater than 20% at block distances of approximately 600 nm (typical of 727 domestic U.S. operation today). Figure 6 shows a comparison of projected fuel savings for this aircraft alongside fuel savings projected by Pratt & Whitney (PWA) and General Electric (GE) for comparable payload aircraft operating over similar stage lengths. These savings are referenced to baseline, comparable technology turbofan powered aircraft. Savings in direct operating costs shown

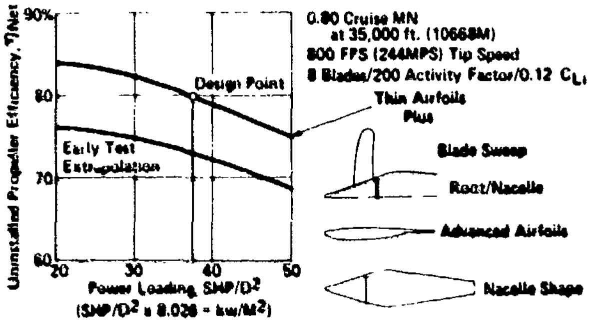


Figure 3. Prop-Fan Uninstalled Efficiency Variation With Power Loading

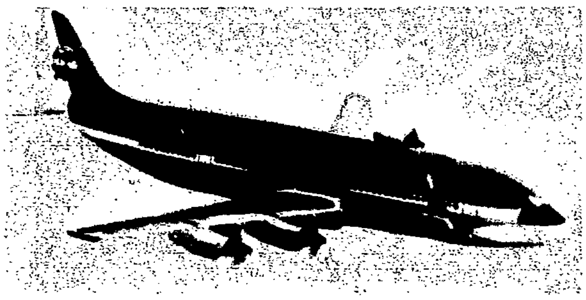


Figure 5. Lockheed Medium Haul Aircraft Concept

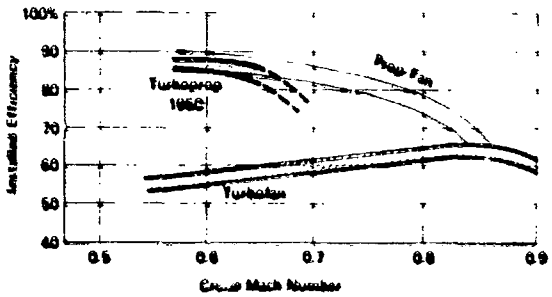


Figure 4. Variation of Installed Cruise Efficiency With Cruise Mach Number

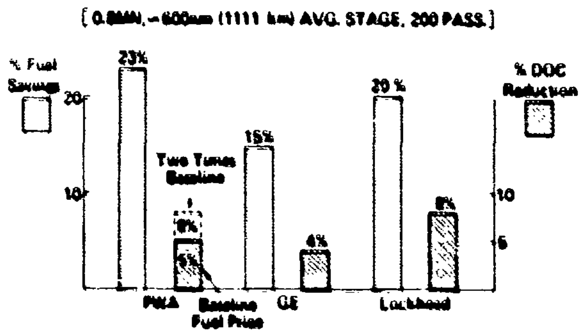


Figure 6. Medium Range Aircraft Studies Prop-Fan vs Turbofan

are for a fuel price of 30¢ per gallon in the case of the PWA and GE studies and 60¢ per gallon in the Lockheed study. Sensitivity of DOC to fuel price is indicated for the PWA results. Also, the design range of the PWA and GE study aircraft was 3,000 nm versus Lockheed's 1500 nm. The results of these studies are remarkable for the magnitude and consistency of the benefits shown. To date, studies of other projected ad ancements in the propulsion area, including the very high bypass, short ducted type turbofan suggested by Denning, Miller & Wright, have failed to show anything approaching these payoffs. [9]

MODEL TESTS

To establish the ultimate performance potential of the Prop-Fan, a series of model tests are planned by NASA in a program that is to extend over several years. Testing of the first model, shown in Figure 7 installed on the tunnel test rig, was completed in late May. This model was designed to attack the compressibility losses in two areas. Aerodynamic sweep up to 30° at the blade tip was incorporated in the design and the nacelle centerbody was somewhat expanded aft of the rotor to provide a retardation of airflow through the inboard cascade (fan) portion of the rotor. Measured performance was better than anticipated for this first model, with 77% propeller (net) efficiency being obtained at the design point, and approximately 80% peak efficiency at a somewhat lower power loading. Diagnostic evaluation of the results indicated several areas where performance of this particular model can be improved, and a retest with the blade spanwise loading reoptimized is planned for this fall. The test results along with more detail on the design of the model are reported in Reference 5.

Based on the remarkably good results from the first model, it is anticipated that the 80% design point efficiency goal will be reached much more quickly than indicated in earlier program plans. The model program is proceeding, however, as originally conceived. A second model is currently undergoing test evaluation and results should be available by early fall. Design of a third model will start in October and plans are well advanced for a fourth model which is to include the first evaluation of thin, supercritical airfoils.

THE ACOUSTIC QUESTION

Since the Prop-Fan will operate with conventional low propeller like tip speeds, it is anticipated that aircraft so powered will have no difficulty meeting existing FAR Part 36 far-field noise constraints, or for that matter, reductions in noise regulations as might result from the implementation of NPRM 75-37. However, there is some concern over potential difficulties in realizing a quiet passenger cabin, one that meets the acoustic criteria generally adopted by the industry for new aircraft designs. These acoustic criteria mandate significantly less cabin noise than was typical of past propeller aircraft. The substance of this concern is illustrated by Figure 8. The path to achieving acceptable interior noise levels is composed of two legs, one of increasing noise level and one of decreasing noise level. Using the noise levels of typical propeller-driven aircraft with their subsonic blade tip speeds and relatively lightly loaded rotors as a basis for comparison, an increase may be anticipated at 0.8 MN due to higher Prop-Fan source noise. The helical blade tip Mach numbers of the Prop-Fan will be in the transonic speed range, how far into it being dependent on the rotational speed selected for cruise at 0.8 MN, 35,000 feet altitude. The higher power loading employed by the Prop-Fan will also increase the source noise level. Several circumstances and design improvements, however, are acting to counteract these effects. The Prop-Fan aircraft will cruise at higher altitudes than the old propeller-driven equipment and greater noise attenuation will be realized. Also, the greater number of blades will raise tone frequency beneficially, and the smaller diameter will allow greater noise reduction due to increased rotor tip to fuselage clearance. Increasing the clearance historically has proven a powerful tool for reducing cabin noise, as shown in Figure 9 which plots measured cabin noise data for a number of aircraft vs their propeller tip clearances.

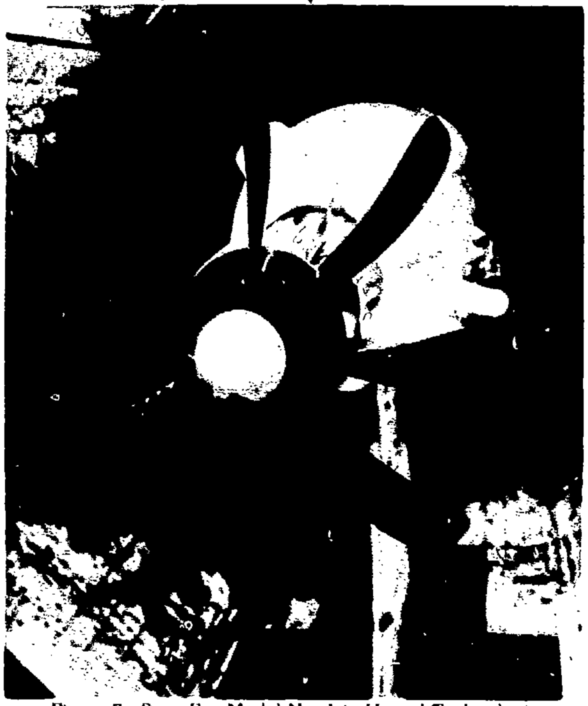


Figure 7. Prop-Fan Model No. 1 in United Technologies Research Center Wind Tunnel

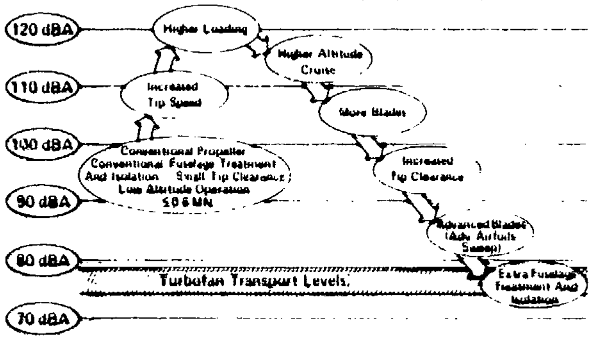


Figure 8. Steps to a Quiet Cabin at 0.8 MN Cruise

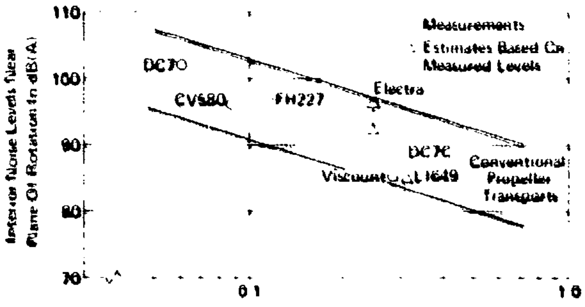


Figure 9. Peak Cruise Noise Levels at Fuselage Centerline

Further noise reductions to obtain the levels of today's jet equipment will have to be accomplished through a combination of quieter rotor design and/or additional cabin wall treatment. The weight penalties to the aircraft and the associated penalties to fuel consumption can range from negligible to significant, dependent to a large degree on the success obtained in realizing such a quiet design. Limited full-scale propeller acoustic tests performed in the 1950's have suggested that delaying the onset of section critical Mach number can have a very beneficial effect on propeller noise generation. As shown in Figure 10, the incorporation of advanced thin airfoils along with blade sweep should result in a marked noise benefit, and if predictions prove correct, little penalty in the way of added cabin acoustic treatment may be necessary.

Adequate test data to analyze thoroughly the cruise near-field noise character of the Prop-Fan is simply not available. The NASA program therefore envisions the conduct of a series of acoustic model tests to provide substantiation and enhancement of currently available data, and hence permit the development of improved analytical models for the study of noise reduction. Acoustic testing of the swept blade wind tunnel model is expected to be conducted late this year and will be followed soon thereafter by tests of other configurations, including models specifically tailored for acoustic purposes.

STRUCTURAL CONSIDERATIONS

The requirements foreseen for the design of an 0.8 MN Prop-Fan will demand little in areas of state-of-the-art structural advancement. The sciences of flow field analysis and rotor vibratory behavior have been substantially refined since the days of reciprocating engine propellers and even the early turboprops. The propeller safety record of the whole turboprop family results in part from some of the early advancements made in these analytical techniques, and represents a dramatic improvement over that compiled on the post-war family of reciprocating engine transports. In over fifty million flight hours of turboprop operation, Hamilton Standard propellers, most of them solid aluminum bladed, have experienced no in-flight structural failures, nor fractures of any type during the critical takeoff mode.[10] This is a remarkable achievement, reflecting both the fact that with the introduction of turbine power, engine excited propeller blade stresses, which were associated with the lion's share of reciprocating engine propeller failures, completely disappeared; and the additional fact that the susceptibility of the propeller blade to substantial strength degradation from operational damage, which accounted for the remaining problems, was eliminated.

Effective means for isolating propeller induced shaft vibration from transmission by the wing structure to the cabin while simultaneously maintaining adequate propulsion system mounting stiffness for purposes of nacelle/wing stability, have been evolved since the Lockheed Electra was designed. These techniques, currently being demonstrated on the deHavilland DHC-7 aircraft, promise low levels of cabin vibration previously unobtainable with propellers. Considerable engineering effort was directed during the design of the DHC-7 to optimize the nacelle mount system to minimize structure born vibrations. Success can be measured by the almost total lack of propeller induced vibrations detected in the cabin during the flight test program at Downsview, Ontario. Even more favorable vibration levels have been calculated for a large Prop-Fan powered aircraft as shown on Figure 11.

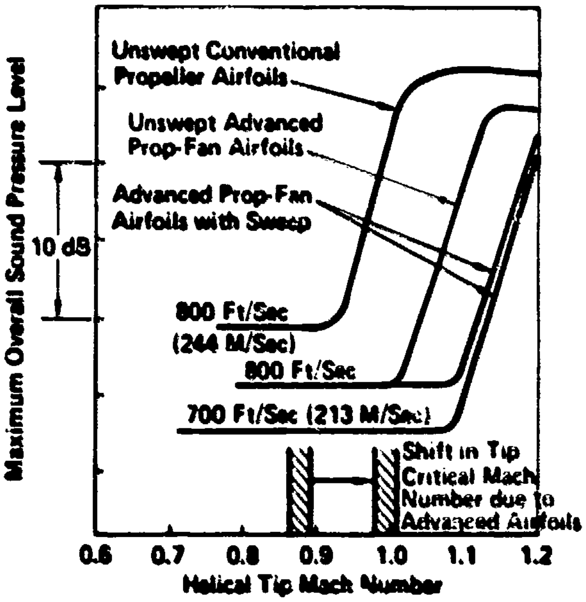
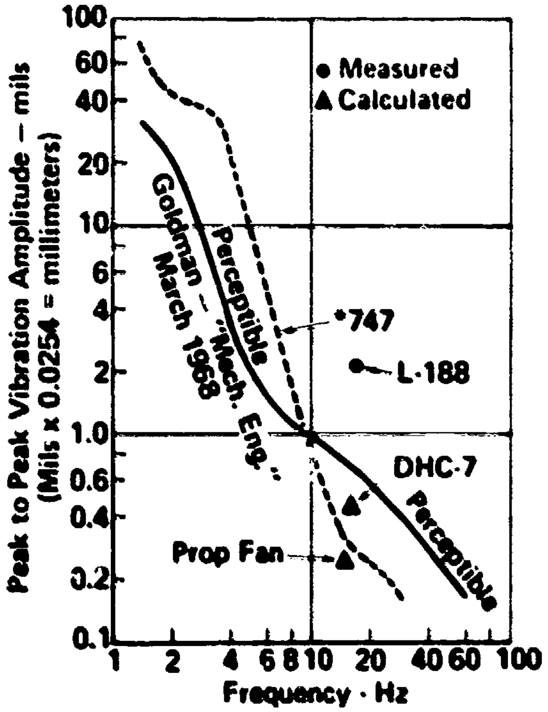


Figure 10. Comparison of Propeller and Prop-Fan Near Field Noise



*Reference NASA TMX-72008, Sept. 1974

Figure 11. Aircraft Ride Comfort (Vertical)

It is likely that a modest advancement in blade structural design will be required, although the technology step involved is less ambitious than if the relevant design improvements made over the last fifteen years were yet to be realized. In order to minimize propeller efficiency losses due to compressibility, the Prop-Fan will require use of thin airfoils. The comparison of blade spanwise thickness ratio distribution of the Prop-Fan versus the earlier Electra propeller is shown in Figure 12. From this it is readily apparent that Prop-Fan blade thickness ratios will most likely run between 65% and 80% of the earlier turboprop blade with the greatest reduction being in the mid-span region where the vibratory stresses are expected to peak. Although the individual Prop-Fan blades are anticipated to be wider than the turboprop's, it is expected that the section properties still will be somewhat less substantial and will require the use of a fundamentally stronger blade construction.

Starting in the early 1960's, Hamilton Standard, under Navy sponsorship, initiated the development of a high strength, spar/shell, composite blade. This blade in its Prop-Fan configuration is shown in Figure 13.[10] In its current production propeller configurations, it is composed of a high strength metal spar "backbone" surrounded by a non-structural fiberglass shell which fills the dual role of forming the blade airfoil and completely protecting and isolating the spar from environmental damage. The first version of this blade was developed for the U.S. Navy reciprocating engine powered P2V, was manufactured in substantial production quantities, and was operated for many years by the Navy very successfully throughout the world. More recently, following the use of other design versions on a variety of experimental propeller-driven aircraft, the fiberglass blade was introduced and is being used extensively by the Navy on its Grumman E-2 early warning aircraft and its C-2 COD transport. It is now in production for the commercial deHavilland DHC-7, and will be placed in production shortly for the Navy's latest, high powered version of the OV-10, the OV-10D. Several of the above listed propellers were designed to handle substantially higher vibratory loads than were typical of the earlier turboprops. The blades have performed flawlessly and have displayed and confirmed unsurpassed tolerance to extreme environmental damage.

From analysis of the predicted loadings anticipated for the 0.8 MN aircraft, it is projected that the thin Prop-Fan blade will not demand any higher component strengths than have already been proven and employed in the design of spar/shell bladed propellers that have experienced substantial successful service operation. Because of the wide, thin airfoil, however, use of high modulus composite materials replacing the fiberglass may be necessary to obtain satisfactory torsional stiffness. While it is certain this change will maintain the essential structural isolation of the spar, the blade's total structural behavior needs comprehensive substantiation which can be best accomplished by actual operational evaluation. A program to accomplish this is in the planning stage.

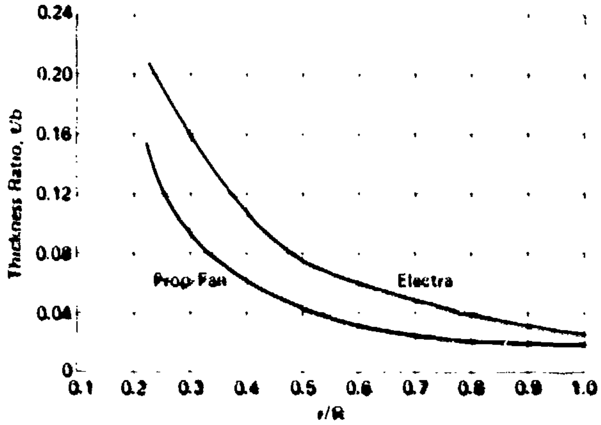


Figure 12. Thickness Ratio Comparison

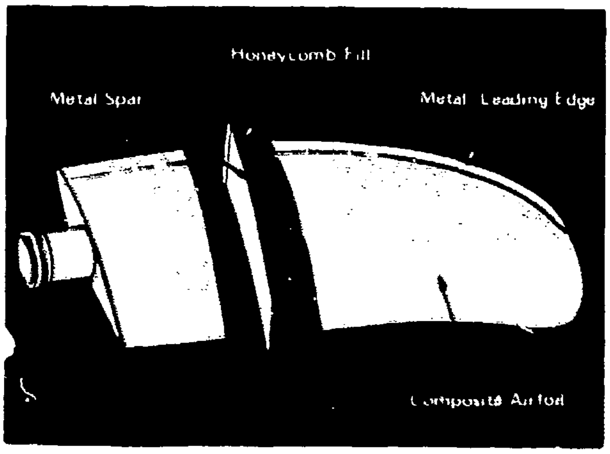


Figure 13. Typical Prop-Fan Blade

PROP-FAN VERSATILITY

The Prop-Fan's key attribute is propulsive efficiency, and hence, its ability to save fuel. As discussed earlier, this advantage is mainly attributable to its inherently high uninstalled efficiency based on very small momentum losses (typical of propellers) and the minimization of blade losses due to compressibility. Other advantages which are inherent in the Prop-Fan concept will now be explored.

Thus far concentration has been focused on a specific design operating point, that of 0.8 MN maximum cruise operation at altitude. During any specific cruise mission where MN is held constant, the thrust required changes over time as fuel is burned off and aircraft gross weight goes down. This results in engine power variations throughout the mission which are sometimes coupled with altitude variations with the intent to optimize TSFC. Today's typical operations at less than full payload and with restrictions in altitude variations force less than full power utilization.

The fuel savings associated with Prop-Fan operation is further enhanced compared to the turbofan as power level is reduced from full cruise rating. This improvement is noted in Figure 14 which shows the Prop-Fan and turbofan TSFC's as a function of power level.[4] The delta TSFC improves 13% as power is decreased from maximum cruise to 70%. The key to this advantage is the Prop-Fan's ability to optimize operative TSFC throughout a mission. Unlike a turbofan with its fixed fan blade angle, the Prop-Fan can either maintain 100% speed or reduce rpm with power variations. This is attributable to its variable pitch feature. Engine studies conducted by PWA show that maintaining 100% rpm with power reductions resulted in the largest fuel savings. This is achieved because both Prop-Fan and turbine stages are operating at high efficiency when the rpm is maintained at or near 100%.[4]

Similarly, the Prop-Fan's advantage over the turbofan improves as MN drops from design cruise speed. The Prop-Fan's uninstalled efficiency optimized for various MN is compared in Figure 15 to a Prop-Fan designed for 0.8 MN and operated at the lower speeds. It can be seen that the efficiency differences between an optimum and non-optimum Prop-Fan are small although there would be a comparable impact on fuel consumption. This is an important consideration because as fuel prices continue to become a larger percentage of operating costs, it may be desirable to effect a further modest reduction of aircraft cruise speeds. The increase in fuel prices in the 1974-75 time period resulted in commercial transports, with design cruise at 0.82 - 0.84 MN, lowering their cruise speed to 0.80 - 0.82 MN. Further speed reductions were not instituted because of an adverse impact on direct operating cost with the existing aircraft designs. A recent study conducted by Douglas Aircraft showed that as fuel prices approached \$0.45 per gallon, the optimum cruise speed for new turbofan aircraft designs would drop to 0.8 MN and lower for minimum DOC.[8] In the event of another oil shortage, fuel allocations, and/or drastic fuel price escalations, cruise speeds of a Prop-Fan powered air transport fleet could be reduced substantially with major fuel savings and without catastrophic economic results.

Studies conducted to date have indicated that the Prop-Fan, unlike the turbofan, is not sized by the takeoff thrust requirement for conventional transports. The thrust per horsepower developed by the Prop-Fan under low forward speed operation is greater than that of the high bypass turbofan engine. This low speed specific thrust characteristic results in a steeper thrust lapse rate (thrust vs velocity) and if both Prop-Fan and turbofan propulsion systems are sized for equal cruise thrust, the thrust generated by the Prop-Fan on takeoff will be greater than that of the turbofan. Figure 16 compares low speed thrust characteristics for the high bypass turbofan and the Prop-Fan (sized at 37.5 SHP/D^2 at 35,000 ft.). Below 0.05 MN the Prop-Fan's thrust is less than that of a turbofan but the thrust advantage above 0.05 MN still results in a significantly shorter field length (20-30%) for the Prop-Fan aircraft.[7] Figure 16 also shows the low speed thrust characteristic with varying design cruise SHP/D^2. Lower SHP/D^2 for constant design cruise thrust results in a larger low speed thrust, a lower slipstream velocity, and a larger slipstream area. The slipstream velocity in this low speed regime is about 200 ft/sec. for the conventional propeller (12 SHP/D^2) and 350 ft/sec. for the Prop-Fan (37.5 SHP/D^2). The turbofan velocity is above 1000 ft/sec., dropping to 600 ft/sec. for the

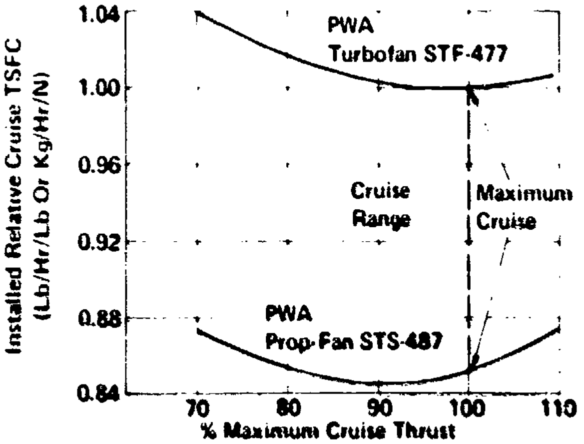
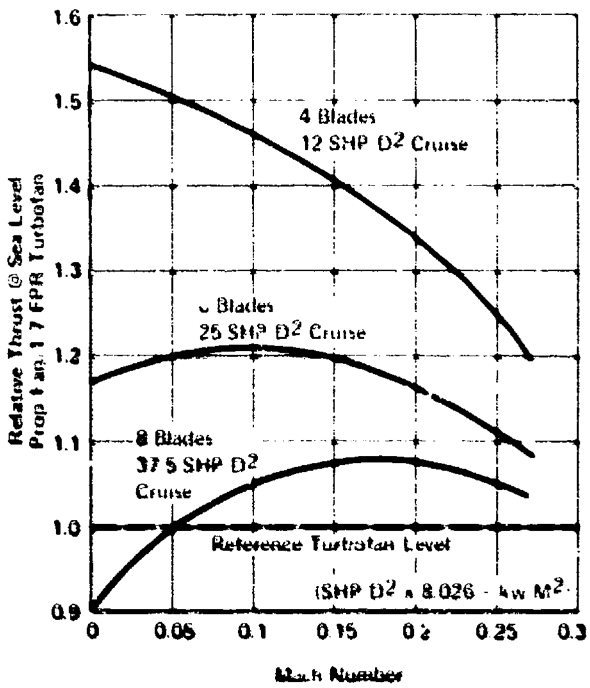


Figure 14. Cruise Thrust Specific Comparison Design VS Off-Design Power Level (0.8 MN @ 35,000 Ft. (10668M) Altitude



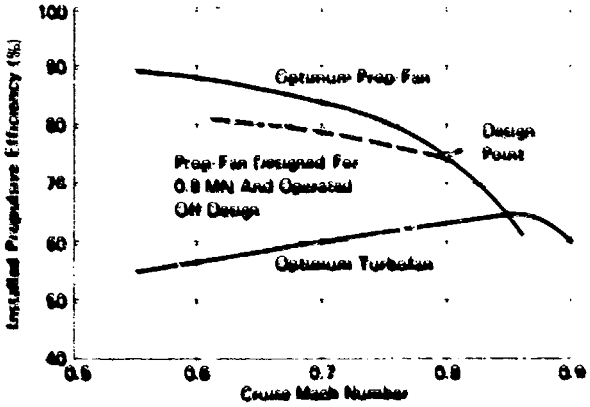


Figure 15. Cruise Efficiency Comparison Design VS Off-Design MN

Figure 16. Low Speed Thrust Lapse Characteristics (Sized for Constant Cruise Thrust @ 0.8 MN)

optimum externally blown flap STOL. The propeller's thrust and slipstream characteristics are desirable for slow STOL aircraft designs but conventional propellers have not proved particularly attractive for high speed STOL. Prop-Fan propulsion, however, should prove more suitable in a blown flap, high speed configuration. The wing and flap span length washed by the Prop-Fan slipstream is about 2 to 2 1/2 times that of the turbofan. As it should provide very effective wing/flap blowing, quite possibly superior to the very high bypass (QCSEE type) turbofan, the Prop-Fan should be studied in detail for high performance STOL applications.

The superior low speed performance of Prop-Fan propulsion can be used to benefit engine ratings. Lockheed capitalized on this advantage for RECAT by using less than maximum available low speed thrust and derating the Prop-Fan engines slightly. Even with the derating, the Prop-Fan aircraft had a 17% shorter takeoff field length and reached its cruise altitude in about 40% less climb distance than the turbofan. [7] It can be seen from Figure 16 that as SHP/D^2 decreases, further derating may be possible. If the Prop-Fan aircraft is designed for conventional takeoff and standard climb routines similar to that of today's turbofan aircraft, significant derating could be accomplished. Derating may allow reductions in either high pressure turbine rpm or maximum engine pressure or temperature levels during takeoff. Engine life may increase and maintenance costs may decrease insofar as they are affected by the takeoff rpm, pressure, and temperature parameters.

It is anticipated that the Prop-Fan's fuel savings over a comparable technology turbofan will vary as a function of range and cruise MN as shown in Figure 17. For short range (under 500 nm) the fuel savings is anticipated to be 25-30% since this mission is dominated by climb and descent operation where the Prop-Fan's TSFC advantage is higher than in cruise. It had always been anticipated that benefits for such missions would be substantial since little if any time is spent at maximum cruise speed. Because of a lack of sufficient advantage, most short range turbofan aircraft today cruise between 0.76 to 0.78 MN. The curves bottom out in the medium range category (1500-2500 nm) where the NASA funded studies so far have concentrated. As the range increases to the long range category (above 3,000 nm), the fuel savings with the Prop-Fan increases again due to the larger fuel fraction of this category aircraft. Where there is a significant amount of fuel used, improved fuel savings results in a lower gross weight aircraft and hence an additional fuel reduction associated with the lighter vehicle. This suggests a major Prop-Fan payoff for long range aircraft. Here commercial freighters and military logistic transports are attractive candidates as there is no need to deal with the passenger sensitivity to block time.

It is the Prop-Fan's variable pitch feature that results in three more identifiable attributes. First, it allows for immediate thrust response upon power application. This can be achieved by maintaining power turbine speed near or at 100% and letting the variable blade angle control thrust. This is especially beneficial on approach where thrust response and the accompanying immediate gain in wing lift could be, in an emergency, the crucial safety margin. Second, reversing is accomplished without any increase in mechanical complexity such as is required with turbofans. The available reverse thrust level during landing coupled with pitch control for taxiing significantly reduces the use of aircraft wheel brakes. Not only is there the elimination of the complicated thrust reversers, there is also a significant increase in brake life and decrease in brake maintenance cost. Third, the Prop-Fan can operate either rotating (windmilling) or not rotating (feather) during engine out. The object of engine out operation is drag minimization. As can be seen from Figure 18, the minimum drag is with the Prop-Fan in a controlled windmilling mode.

Windmilling was, of course, not desirable on the old reciprocating engines due to the high drags which resulted with non-feathering propellers of that era. But today, windmilling rotors are commonplace on turbofan engines where the fan blade position is fixed and there is no pitch control. Since the idea of engines rotating while shut down is now accepted, it seems reasonable that an advanced turboprop system should operate in a similar manner if it enhances engine-out operation. With the advent of significantly improved pitch control concepts, operation of the Prop-Fan in a controlled windmill condition is worthy of pursuit. The pitch control would lock the rotor blade angle in a favorable position such that the drag with the Prop-Fan would be lower than past transport propellers which were feathered.

The functional features of variable pitch discussed above are complemented by the relative mechanical and hydraulic simplicity of modern pitch change mechanisms. As discussed in reference 10, variable pitch mechanisms and controls for Prop-Fan will exhibit the technological advancements brought to the industry over the last twenty years. New variable pitch mechanisms will have about half the parts of those in existing production turboprops and the controls will be even more simplified through the use of digital electronics.

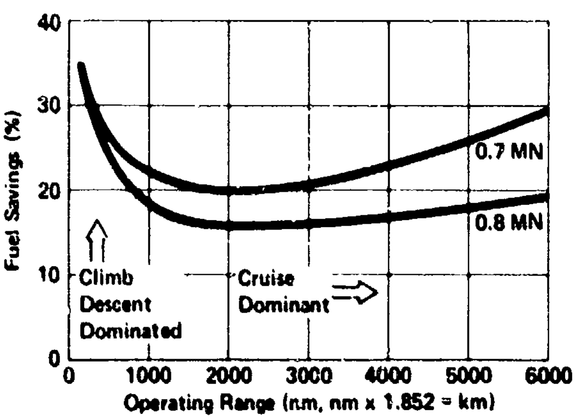


Figure 17. Fuel Savings with Prop-Fan over Comparable Technology Turbofan

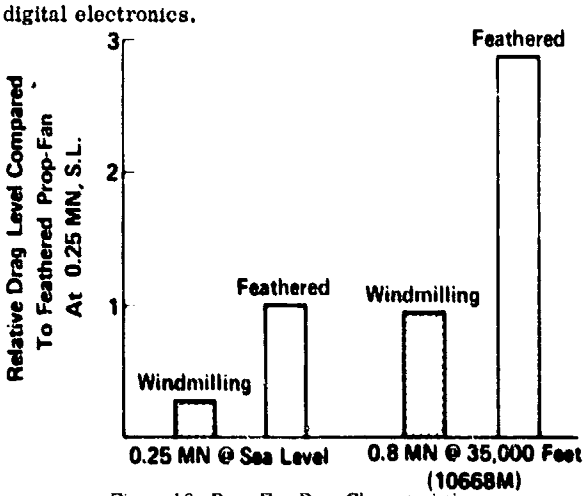


Figure 18. Prop-Fan Drag Characteristics Windmilling and Feathered

Since the Prop-Fan is a variable pitch, free-air device, it has certain advantages over fixed pitch, ducted devices in terms of growth (higher thrust) potential. Typically propellers have been able to provide more thrust, when the gas generator grows, without any configuration changes. This has been accomplished merely by adopting an available higher operating blade angle. Although higher power with the same blade is slightly less than optimum from an aerodynamic efficiency standpoint, this is usually a small price to pay for increased thrust capacity. Additionally, more thrust is achieved by the Prop-Fan for a fixed power change in the engine than with a turbofan. This is because of its higher propulsive efficiency and hence higher thrust per available horsepower. If either improved efficiency or greater structural capacity is desired, blade configuration changes can be accomplished without having to reconfigure a duct. Blade tailoring can be accomplished in a free-air rotor rather easily and probably without any major changes to either the disc or the pitch change module. Historically, this has been the case with propellers. Table 1 presents data on Hamilton Standard's production turboprops with regard to increases in thrust. In all four cases, significant thrust increases were achieved both without hardware changes and with minor hardware changes. Thrust increases over that of service introduction ranged from 30% to 60%. The ability to increase thrust on a high bypass turbofan is rather limited. Thrust increases without redesigning the entire fan stage are usually limited to minor increases in either the turbine temperature or compressor supercharge. Engine redesign and improvements, without a new fan, have traditionally resulted in about a 15% thrust increase over that of service introduction as evidenced by the PWA JT8D-1 to - 17 history where maximum static thrust increased from 14,000 to 16,000 pounds.

ENGINE TECHNOLOGY

In preparation for production starts, the technology level of candidate core engines must be upgraded. Shaft engine performance levels have lagged behind their turbofan cousins simply because of a lack of a need. Moreover, the major engine manufacturers have indicated that either existing or advanced turbofan core engine technology can be transferred to shaft engines and made fully effective in either existing engine derivatives or new designs. The importance of taking action in this regard is illustrated by Figure 19. Here the turbofan TSFC levels shown are based on published high bypass turbofan data. The widening band indicates the variations in technology level predicted for the future. The comparable technology Prop-Fan TSFC and BSFC lines were established using a 15% TSFC advantage for the Prop-Fan and an 80% net efficiency to calculate BSFC. The 15% TSFC advantage is based on the NASA funded studies and the 80% efficiency is that projected by Hamilton Standard for 0.8 MN, 35,000 foot altitude operation. The lower dashed line represents actual shaft engine BSFC's and these are projected back to Prop-Fan TSFC's, again assuming 80% efficiency. It can be seen that today's shaft engines, typically represented by the GE T64 and DDA 501, do not exhibit the core technology of today's turbofans. Comparable technology levels will be approached if the development of derivative shaft engines such as the DDA PD370-22 and the PWA STS 476 is pursued. The PD370-23 engine, initially developed for the Army's Heavy Lift Helicopter, exhibits an 18% improvement in BSFC over the 501-D22A and this increases to 27% with the PD370-22 or STS 476. The "New" engines are those which resulted from the PWA and GE engine studies mentioned earlier.

It is anticipated that with the renewed interest in the turboprop and the Navy's interest in a 100% V/STOL carrier based fleet, there will be more attention devoted to new shaft engines exhibiting the technology levels projected for turbofan engines. Table 2 is a listing of potential shaft engines using near-term advanced cores. This list is not all inclusive and it is expected that more new shaft engine configurations such as the PWA STS 487 will appear in the next several years.

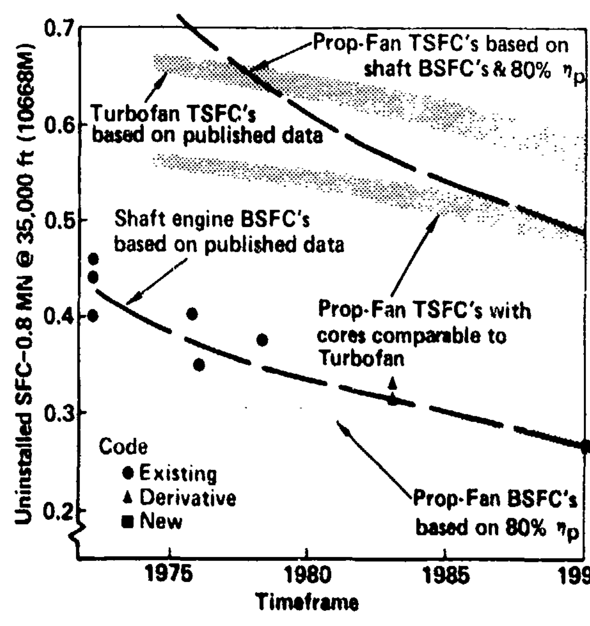
Table 1. Growth in Turboprop Thrusts

Propeller Model	Thrust* Level At Intro Into Service	Thrust* Levels Without Changes	Thrust* Levels With Minor Hardware Changes	% Change* Final/Initial
33LF	1395	1802	2230	60
53C51	2273	3181	3750	65
63E60	6339	7577	8510	34
54H60	6815	8234	8985	32

* Thrust levels are at 100 knots (185.2 km), sea level. % changes for cruise are comparable. Pounds thrust x 4.448 = Newtons

Table 2. Potential Prop-Fan Core Engines

Take-off Thrust Class (Pounds x 4.448 = Newtons)	Core Engine
7 - 12,000 pounds	AVCO ALF-502 GE TF-34 RR RB401
10 - 18,000	DDA (PD370)
20 - 30,000	PWA JT10D (STS 476) GE CFM56
30 - 40,000	PWA F100/GE F101
60 - 80,000	PWA JT9D/GE CF-6/RR RB211



TSFC = Lb/Hr/Lb x 0.1 = Kg/Hr/N
BSFC = Lb/Hr/HP x 0.608 = Kg/Hr/KW

Figure 19. Engine SFC Technology Comparison

MISSIONS

There are many potential missions which the Prop-Fan propulsion system would enhance. Table 3 is a listing of the potential new aircraft programs for which the Prop-Fan is well suited. These missions all involve 0.8 MN operation or less. Specifically, those which demand operation at or near 0.8 MN are all commercial requirements, i.e., short, medium, and long haul with the higher speed more critical with increasing range. Other potential missions are those which do not particularly benefit from speeds as high as 0.8 MN. These include military transports of all sizes and commercial freighters. Of particular interest are the first two listed, the relationship of which will now be discussed.

In the military long endurance mission, an aircraft cruises to a point of operation, stays "On Station" for as long as possible, and finally cruises back to base. The objective of this mission is to maximize time on station within reasonable limits of crew endurance. Transit time to station may become important with regard crew fatigue depending on the distance to station. The P3 aircraft is an example of a long endurance patrol aircraft. It cruises at speeds up to about 0.6 MN and while on station its airspeed is much less than its cruise speed. The Prop-Fan is ideally suited for long endurance patrol missions since a large percentage of time is spent at low flight speeds. The Navy is presently studying a long endurance/patrol aircraft. This new vehicle is likely to have a higher cruise speed than the P3, but its loiter speeds will still be lower than its cruise speed. It can be seen from Figure 20 that the Prop-Fan's efficiency improves for a loiter situation (on station) compared to a cruise to station condition for which its propulsion system is designed. This is opposite from the turbofan efficiency characteristic which deteriorates rather than improves at loiter speeds. For instance, a Prop-Fan designed at 36.4 SHP/D^2 for 35,000 foot, 0.75 MN cruise would have a net efficiency of about 85.5% when it is operated at the two off-design loiter conditions shown. The optimum tip speed is still 800 ft/sec. for 0.45 MN operation but at 0.25 MN, sea level, the optimum tip speed is less than 800 ft/sec. Unpublished Navy aircraft studies show significant benefits with the Prop-Fan over the turbofan and these have, in general, confirmed the potential indicated by a recent in-house Hamilton Standard study, which was based on the following ground rules:

- 4 hours time on station (TOS)
- 2000 nm radius of action (ROA)
- 0.7 MN cruise speed and 0.8 MN dash speed
- Comparable airframe aerodynamics and engine technology as turbofan powered aircraft
- 7000 foot maximum takeoff field length
- 180,000 pound airframe takeoff gross weight (TOGW)
- 40% useable fuel/TOGW ratio

For the in-house study, four-engine turbofan and Prop-Fan powered aircraft were configured to comply with the above ground rules. Takeoff field length sized the turbofan while the Prop-Fan was sized by dash speed. This resulted in a Prop-Fan engine core power level significantly less than the turbofan core size. The aircraft operational results are shown in Figure 21. The Prop-Fan aircraft at 180,000 pounds TOGW is more than adequate in meeting the TOS and ROA ground rules. Figure 21 also shows a reduced gross weight Prop-Fan aircraft which more than meets the ground rules. This final Prop-Fan aircraft has a 28% lower TOGW and 30% lower fuel usage for a 39% increase in range, compared to the turbofan powered aircraft. A further reduction in range would result in further gross weight reductions and fuel savings improvements, however, this patrol mission requires a specific payload and attendant fuselage size, indicating a floor on TOGW. The 130,000 pound Prop-Fan aircraft is about the same weight as the Lockheed P3. Compared to the P3, the Prop-Fan aircraft carries about 20% more payload, has a range more than double, uses less fuel, and has higher cruise and dash speed capability.

Table 3. Potential New Aircraft Programs

Aircraft	Production Start	User
Military long endurance/patrol	Mid 80's	U.S. Navy
Medium haul	Mid 80's	Commercial
Military airlift/tactical	Mid 80's	U.S. Air Force
Military airlift/strategic/tanker	Mid 80's	U.S. Air Force
Military carrier based patrol	Late 80's	U.S. Navy
Short haul	Late 80's	Commercial
Military airlift/large long range strategic/tanker	Late 80's - Early 90's	U.S. Air Force
Long haul	Early 90's	Commercial

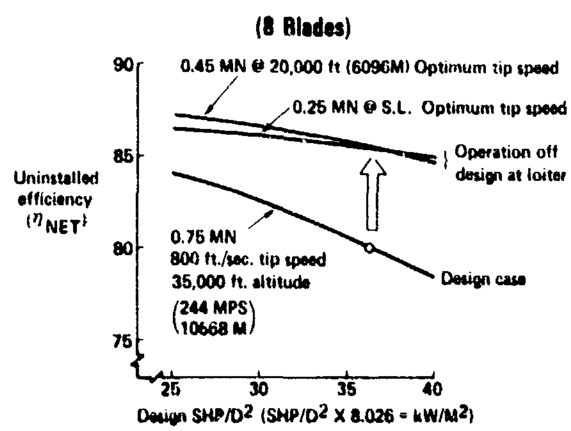


Figure 20. Prop-Fan Efficiency Characteristics for Long Endurance Missions

Development of a new Navy long endurance aircraft should produce a propulsion system compatible, with only minor modifications, to the needs of a new commercial medium haul transport in the mid-1980's. This would be a 1500-2000 nm design range, 0.8 MN aircraft slated as a possible 727 replacement. As mentioned earlier, Lockheed studied such a Prop-Fan powered aircraft under NASA contract and reported on its work in Reference 7. Table 4 presents data for the Lockheed aircraft and for a new patrol aircraft (P3 replacement) typical of those being studied by the Navy. It appears quite reasonable to project a course by which Prop-Fan propulsion designed and developed for Navy requirements could also be used with minimum changes for a new commercial medium haul aircraft. Based on the historic growth potential with propellers (Table 1), making the transition from military patrol to commercial medium haul requirements should be routine. The Navy aircraft would use a smaller diameter Prop-Fan whose blading might have aerodynamic characteristics which are slightly different from the 0.8 MN commercial requirements. The increase in thrust required for the commercial transport would probably require a blading change but it is likely that the disc, pitch change and control hardware would be only slightly affected. The growth in horsepower is well within the capacity of the DDA T701 family. The horsepower range between the PD370-23 and -22 is projected to be from about 8000 to 12000. An alternative, perhaps more attractive, path to the derivative medium haul commercial aircraft would call for the Navy to employ a propulsion system designed for 0.8 MN from the start. As indicated earlier, very little efficiency degradation over an optimum design would be encountered for the long endurance/patrol operation (see Figure 15). The potential of this approach for reducing the combined start-up costs for these two aircraft should be studied carefully.

It is anticipated that the Prop-Fan's versatility may permit use of the developed Navy long endurance aircraft propulsion system for a military tactical type transport and military carrier based patrol aircraft as well. Such general utility is typical of existing turboprop equipment where similar propulsion systems are used on the existing Lockheed P3 and C130 aircraft and Grumman E2 and C2 aircraft. The precedent, therefore, has been set for the successful adaption of a basic Prop-Fan propulsion system to these various missions. Such missions require good, low speed thrust and low speed aircraft control and have been effectively served by propeller type equipment for many years.

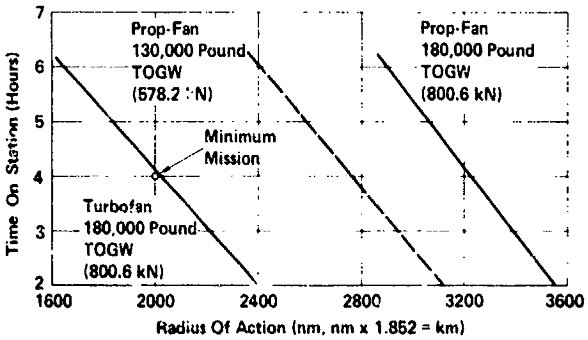


Figure 21. Long Endurance Mission Comparison

Table 4. Aircraft Requirements

	Military Long Endurance Patrol	Commercial Medium Haul
Mach number	0.75	0.8
Cruise altitude (ft.)	35,000 (10668 M)	35,000
Gross weight (lbs.)	175,000 (778.4 kN)	220,000 (978.6 kN)
No. of engines	4	4
Cruise thrust required per engine (lbs.)	2750 (12.23 kN)	3350 (14.9 kN)
Maximum horsepower per engine	9000 (6710.4 kW)	11000 (8201.6 kW)
Prop-fan diameter (ft.)	11.6 (3.54 M)	12.6 (3.84 M)
No. of blades	8	8

CONCLUSIONS

Recently completed aerodynamic model testing has confirmed initially expressed confidence in the projected high levels of Prop-Fan performance at 0.8 MN. There is little doubt that the required 80% net propulsive efficiency at the selected cruise design point will be achieved in the near future Based on such performance, NASA funded engine and airframe studies have confirmed very attractive fuel savings and operating cost reduction potential for Prop-Fan powered aircraft designed for 0.8 MN. The United Airline study indicated passenger acceptance of such aircraft provided these cruise speeds along with today's traditional passenger comfort levels are maintained.

The off-design capability of the Prop-Fan has been examined and found to be excellent from the standpoints of low speed performance and near 0.8 MN cruise SFC optimization. Fuel savings of the Prop-Fan, over the turbofan, should grow from 15-25% for 0.8 MN operation to 30% or more at less than 0.8 MN. Accordingly, the Prop-Fan's operational versatility is well suited for a wide range of potential subsonic missions, both military and commercial. In particular, initial studies show that an advanced Prop-Fan powered long endurance patrol aircraft would benefit greatly from this propulsion system. A derivative of the powerplant for such an aircraft should also prove ideal for an advanced commercial medium haul transport, and should find uses, perhaps in slightly modified form, in aircraft designs for other specialized missions.

To prepare for production development, Government support of technology research and advanced development is essential. The planned NASA program, if fully implemented, will accomplish much in this regard. Supplementary support from other Government agencies as well as private industry will also be necessary if the full potential of the Prop-Fan is to be realized.

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DISCUSSION

A.C.Willmer

Does the type of propeller described in this paper have any significant performance loss for take-off and low speed flight?

Author's Reply

I did not have time obviously to cover this particular area. We have given off a significant amount of the thrust augmentation which the conventional propeller provides at very low speed however the laps rate compared to the 1.5 - 1.6 pressure ratio turbofan is still favorable. So far providing the same cruise thrust, we are still providing 20% higher thrust at the lift-off and higher thrust throughout the climb regime. So this aircraft will be quite different from the conventional turbojet. Once it gets off the ground the climb rate of the turbojet will deteriorate very rapidly because of its laps rate.

I would also mention that at static conditions there will be a modest fall off of thrust because of a fairly significant spanwise stall. This washes out by about 20 – 30 knots. forward speed.

M.Dabbadie

Pouvez-vous nous dire quelle est la vitesse périphérique au point d'adaptation de votre prop-fan?

Author's Reply

The rotational tip speed is 800 ft/sec. The helicap tip speed is in the nature of 1.1 Mach.

R.M.Denning

I would like to ask Mr Jackson when he talks about a reference turbofan engine from which he gets his improvements in efficiency. Is he talking about the standard formula for todays turbofans which have a fan pressure ratio about 1.7? Because it seems to me that if we are looking for the estimate efficiency all the turbofan designers will say you have to go to a lower specific thrust. I am rather anticipating my paper but in all the design studies we have done there is another 15% of performance to come just by optimising the turbofan cycle, probably which is at 0.8 Mach just about to eat up the differences.

Author's Reply

I guess we would really have to get into detail. I am just dealing with the propulsion efficiency and of the unit obviously. What we have seen from the studies conducted in the States the potential for advancements is not anything like the numbers that you are stating. I guess I am really not in the position to comment on that very forceful. I can't take a very strong position on the potential for propulsive efficiency of this kind of device and where it can get to compared to fan engines of today.

VARIABLE GEOMETRY IN THE GAS TURBINE

- THE VARIABLE PITCH FAN ENGINE -

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SYNOPSIS

High Bypass Fan Engines offer advantages for wide speed-range subsonic aircraft and those requiring high relative take-off thrust and low noise. The addition of variable pitch adds a steep approach path control facility plus a reverse thrust capability and allows optimisation of performance, ratings and noise but imposes certain limitations on fan design. The results of work leading up to a demonstration of the M45SD-02 variable pitch engine are discussed.

1 INTRODUCTION

A variable cycle engine must have the ability to make significant changes to its thermodynamic cycle at a particular thrust and flight condition without incurring a performance and weight penalty which would cancel the value of variability. The geared Variable Pitch Fan can claim to belong to this class of engines as can the propeller turbine and the variable final nozzle Olympus engine on the Concorde.

A number of experimental variable pitch single-stage fan engines have now reached the test bench development stage and one, the Turbomeca 'Astafan' has been flying for some years in an experimental aircraft.

As with many innovations, the reasons behind the use of variable pitch are diverse and complex. In the various applications proposed there is not universal acceptance of the need for such a facility although many believe that the benefits would be of considerable value and a pattern of controversy, of some similarity to that of the late 1920s around the virtues of variable pitch propellers, is taking place. The technical issues are somewhat different. In the case of the propeller, the single very clear need was to have variability in order to be able to exploit the full power of the piston engine at all aircraft speeds. With a ducted fan this need, while desirable, is not so marked since the mis-match in fan operation that occurs across the aircraft speed range is small by comparison. This is because control of the air velocity at the fan face is largely dominated by the fan propelling nozzle and therefore does not experience much effect of the aircraft speed. There are however many additional advantages which collectively make the case for the VP fan engine.

A specific example of a variable pitch engine, the M45SD-02, built by Rolls-Royce and Dowty Rotol with the assistance of SNECMA and the financial support of the British Government, is examined and some of its characteristics described.

Other possible uses for the principle are discussed and in the final part of the paper consideration is given to the advantages and disadvantages stemming from the use of such engines in aircraft.

2 DESCRIPTION OF A TYPICAL VP FAN ENGINE

Designs so far have been confined to single-stage units driven through a reduction gear, Figure 1, and have followed generally the practice for propellers with respect to stressing and integrity. They have been based on the use of a mechanism within the hub which rotates the circular roots of the blades by either mechanical or hydraulic power. The fan aerodynamic loading and vibrational stress constraints on blade design rather than the accommodation of this mechanism and the blade root bearings, which have to restrain the centrifugal loads of blades, have dictated a high hub to blade tip diameter ratio - typically 0.5. Further the accommodation of the root bearings tends to limit the number of blades (Figure 2).

At its design blade angle, the aerodynamic or blade profile standard, performance and noise can be generally as for a conventional fixed pitch design though the following factors may result in departures:

a) The blending of the profiles into a circular root.

b) The mode of achieving reverse pitch if thrust reverse is a requirement. To achieve 'reverse pitch' from forward pitch via the fine condition requires a blade-chord-to-space ratio of less than unity in order that the blades will pass one another. This can produce a progressive modulation of thrust through into reverse pitch whereas the alternative method of moving to reverse through coarse pitch and stall is not continuous and may introduce handling problems.

c) Necessity for a cylindrical fan casing in order that the blades can turn and this results in a shallow hub angle tending to give high rotor loading.

d) Restriction on the number of blades imposed by the mechanical arrangements for accommodation of root bearings etc.

Reference 1 details these mechanical and aerodynamic points.

The operating characteristics that have been shown to be possible from estimation and demonstration are as follows:

a) Pressure ratios of up to about 1.45. The limit is imposed by tip speed (blade root bearing stressing) and blade chord in the case of the 'reverse through fine' mode.

b) Efficiencies of better than 85% depending on pressure ratio and blade design (Figure 3).

c) Reverse thrust levels of 35-50% depending on direction of blade rotation to achieve the 'reverse pitch' condition - the 'reverse through fine' results in an adverse camber of the blade with respect to the reverse air flow while the 'reverse through feather' has a favourable camber condition but in both cases the blade twist is in the wrong sense. Reverse thrust levels may also be affected by the design pressure ratio of the fan. This is because at high values the de-supercharging of the core of the engine resulting from the reverse of the fan blades may not leave enough shaft power available within the top temperature of the engine to drive the reversed and now relatively inefficient fan up to maximum rpm.

d) Response time from full coarse forward pitch to full reverse pitch of the order of one second.

While in the light of current views on applicability to various aircraft these characteristics appear to be well matched and in no way restricting it can be foreseen that, if required, designs having for example higher pressure ratio and a faster response could be possible.

3 ENGINE DESIGN AND VARIABLE PITCH

A number of different layouts of VP fan are possible.

The following examples can be listed:

A Single-shaft, eg Astafan.

B Twin-shaft with both the fan and an intermediate compressor driven by a power turbine.

C Twin-shafts with the fan alone driven by a power turbine.

D Blow fan or multistream engine specially designed for Augmentor Wing or similar aircraft.

These are illustrated in Figure 4.

Experimental versions of A, B and C are now in existence while D has been projected.

A <u>The Single-shaft Engine</u> - By attaching a variable pitch unit to the gas generator via a gearbox and linking the LP turbines to the same shaft it is possible to simplify the engine layout and control the gas generator speed by pitch angle up to a governor limit. Such an engine is only practicable because of variable pitch which makes starting possible by unloading the gas generator with the fan in fine pitch during the starting cycle.

This system has a progressive disadvantage as the gas generator pressure ratio is increased. The higher compression and expansion ratios of the system produce an increasing annulus area change across the engine which leads to the blade stress in the final stage of the turbine dictating the shaft speed for the rest of the engine. Figure 5 shows that this leads to a rapid increase in the number of compressor stages in the gas generator portion of the engine particularly when the pressure ratio rises above 20:1.

The LP turbine profile stress parameter chosen for these particular studies is typical of the limits set on today's current rules for large turbine designs. It might be argued that such stresses are by no means upper limits and with the necessary incentives could be increased. A 33% increase in stress would reduce the number of stages indicated by 25%.

Starting powers may become a problem with larger high pressure ratio engines.

For a small thrust size of engine such as the Astafan where a high pressure ratio is not justified this is probably the optimum layout.

B <u>The Twin-shaft with Linked Fan and Intermediate Compressor System</u>
For larger engines where higher pressure ratios are required to combine the highest thermal and propulsion efficiencies then Type B is probably the bes compromise. The centre portion of the engine which in A would otherwise run at unrealistic low stresses is now placed on a separate shaft and run at perhaps twice the speed of the LP/IP system.

By this means the fan is speed stabilised by this IP compressor system particularly in fine pitch and in turn the fan exerts a positive control over the compressor speed and hence the core flow. Much of the advantage of the single shaft concept is then retained. A consequence of this system creates a mismatch of the IP and HP compressor exit and entry flows - a situation which can be corrected by blow-off between the compressors in extreme situations.

C The Free-shaft Variable-pitch Fan - In this case the fan is completely independent of the gas generator, it being driven by a separate LP turbine system. Because the fan in fine pitch has little power absorption the steady state pitch setting has specific limits outside which it cannot go without danger of fan overspeed. Its operation becomes similar to that of a free-power-turbine turbo-prop. The scope for varying the cycle is very limited compared with A or B.

D The Blow Fan or Multi-stream Engine - This type of engine is a logical extension of layout B in that the incidental blow-off is built up into a permanent bleed capability which can be varied in size to suit an aircraft requirement for large amounts of bleed air at take-off and landing. In conventional flight the bleed must be efficiently exhausted through a propulsion nozzle.

The fraction of bleed thrust which is provided determines the bypass ratio as shown in Figure 6. When 80% of the thrust is in the bleed flow, the VP fan has ceased to have any relevance since the flow bypassing the IP compressor has fallen to zero, ie the fan is now the first stage of the IP compressor.

A good compromise layout which provides the right amount of thrust modulation by the VP unit would have 40% thrust in the VP fan bypass, 40% bleed thrust and 20% residual hot thrust.

4 DEMONSTRATOR ENGINES

Engines of three of the above types have been built and run to demonstrate the geared VP fan principle, viz the Astafan, the UK demonstrator, and Hamilton Standard units. The US QCSEE programme is also underway.

The UK demonstrator constructed by Rolls-Royce/Dowty Rotol is designated M45SD-02 and was created by conversion of the standard M45H-01 (currently in service on the VFW Fokker 614 aircraft) by replacing the direct drive fixed pitch fan with a geared VP fan of a lower pressure ratio and adding a 4th stage to the LP turbine. The layout is shown on Figure 7.

Apart from the fan, major differences from the M45H-01 are as follows:

1 Addition of 4th stage LP turbine with increased area exhaust unit.

2 Bleed duct and valve at IP compressor outlet to cope with changes in IP compressor running line caused by altering fan pitch and hence shaft power.

3 Swan neck duct between fan and IP compressor due to greater hub/tip ratio and flow of VP fan.

4 Epicyclic reduction gear of 2.38:1 ratio capable of transmitting 14000 shp. (Note: This gear ratio is much lower than for a conventional turbo-prop and is near the limit for epicyclic gearing. Hence it was necessary to prove the design.)

5 Translatable reverse-flow intake in fan outlet duct.

6 Noise absorption linings in fan intake cowl, inner and outer surfaces of both the fan outlet duct and the hot jet nozzle. (Note: A very large area of acoustic lining was employed to meet a severe STOL noise target.)

The cycle was based on a full power hot core velocity of 900 ft/sec aimed at achieving a peak noise level of 95 PNdB at 500 ft. The engine is currently on test with the following objectives:

(i) To demonstrate the practicability of:

a) An engine of low specific thrust and high efficiency giving a high take-off/cruise thrust ratio appropriate to the steep climb angles of STOL aircraft.

b) A low tip speed fan made possible by use of a light-weight epicyclic reduction gear.

c) An engine having the flexibility of control and rapid response characteristics achieved by the use of a variable pitch fan which are features necessary for the steep approach and landing.

d) An engine designed to have source noise not greater than 95 PNdB at 500 ft under forward speed conditions.

e) The ability to achieve reverse thrust levels similar to those of a conventional reverser by means of the variable pitch fan.

(ii) To explore and investigate the char[illegible]cteristics of:

a) The engine for the purpose of defining and exploiting the use of a control system to link the variable pitch, fuelling and spool matching facility.

b) The noise of a variable pitch fan.

c) The core engine exhaust noise. This will be aided by the ability to vary turbine loading by changing fan pitch setting.

5 CYCLE VARIABILITY

To be able to control the engine cycle to a significant extent with this type of engine it is necessary for the fan to be linked to the gas generator as in types A, B and D; otherwise the gas generator is a semi-independent unit which can only be mildly affected by the variation in fan supercharge brought about by VP (Type C).

The cycle variability arising from the use of types A and B VP engines is:

(i) Independent control of cycle pressure ratio and thrust for a significant part of the thrust range.

(ii) Adjustment of specific thrust by selection of an appropriate pitch.

(iii) Matching of fan and nozzle characteristics for efficiency and handling purposes.

(iv) Matching of gas generator flow and turbine temperature to maximise thrust at non-standard temperature conditions.

(v) Selection of fan operating condition for optimum total engine noise.

(vi) Selection of fan/IP and HP compressor speeds to provide optimum acceleration characteristics.

(vii) Reversing thrust with depressurised gas generator.

All these features stem from the fact that on these engines the fan pitch uniquely controls the gas generator flow while the fuel flow controls the HP spool speed.

The range of variability open is illustrated in Figure 8 which shows the comparative performance of the M45SD-02 model fan at both design blade angle varying speed and design speed varying blade angle.

A typical map of an engine operation is given on Figure 9 which shows the massive control of the thermodynamic cycle exercised by the fan pitch. This can be interpreted as allowing an independent selection of fan/IP speed at take-off at varying intake temperature while TET is maintained constant at a limit value leading to a result such as in Figure 10 (Reference 2).

Alternatively the VP facility can be used to select the optimum conjunction of fan pressure ratio, compressor and fan efficiencies at low thrust as shown in Figure 1[illegible] (Reference 2). This can lead to sfc changes of the order of 5% but which depe[illegible] particular combinations of component characteristics of the engine.

Acceleration from low thrust levels depends on the matching of the compress[illegible] tion relative to its surge line and the level of pressures in the core engine. The acceleration torques are a simple function of the inlet pressure to the spool to be accelerated. Hence if the fan/IP compressor cycle is maintained at high speed at low thrust, the HP compressor spool will accelerate far more rapidly than if the whole of the gas generator is rotating at low speed. In addition the engine has only to increase fan pitch to absorb the power in the LP/IP spool.

The single-shaft engine is the extreme case where the engine can rotate at constant speed and almost instantaneously vary thrust by changing fuel flow and blade angle.

On the Type B engine the result is almost as good as with Type A as shown in Figure 12. An acceleration time of 1 second dictated by the physical limit on fuel flow change has been calculated on a dynamic model compared with about 4 seconds for an equivalent fixed pitch engine limited by surge.

6 AIRCRAFT APPLICATION

For various reasons it can be seen that the VP fan can best exploit its advantages in engines of relatively low fan pressure ratio and specific thrust. A good compromise probably lies in the range 1.3 - 1.4 pressure ratio. To go too high invites excessive fan unit stress and to go too low, excessive powerplant diameter.

The potential applications for such a lower specific thrust engine are:

1 Short-haul regional airliners

- (fuel efficiency and correct thrust match).

2 Long-range anti-submarine patrol aircraft

- (good low speed sfc and ability to lift off high take-off weights).

3 Military tactical STOL transports

- (high take-off cruise thrust, good sfc, superior thrust control).

The results of an extensive parametric aircraft/engine study are shown on Figure 13 which indicates that a 4000 ft field twin-engined aircraft has its performance characteristic correctly matched with a 1.35 fan pressure ratio engine. A four-engined tactical transport design for STOL (2500 ft field) would be just matched at 1.32.

Many of the merits of fan/cycle variability in such aircraft are clearly evident but difficult to quantify. The tables in Figures 14, 15 and 16 summarise the results of various studies to evaluate qualities where possible.

One particular quality of note is the precise thrust control at near-zero or negative thrust without acceleration problems. This can be of use on approach to land to reduce flap settings and noise, on descent from altitude to avoid use of dive brakes and to maintain high rpm values in icing conditions. Accessory drives and cabin conditioning may also be made simpler.

7 PROGRESS WITH THE M45SD-02 DEMONSTRATOR

Completion of the build of the engine was achieved in early 1975 and it has since been undergoing tests as planned for the early stages of the programme for demonstration of the objectives referred to above. At this time of writing (June 1976) the testing has yielded the following:

a) Demonstration of easy and reliable starting.

b) Proof of the design concept of the reduction gear by running at maximum speed over the full range of power of the engine.(ie fan blades fully coarse pitch to fully fine pitch).

c) Achievement of maximum design static thrust of 10000 lb.

d) Assessment of performance over a wide range of speeds and fan blade angles.

e) A good indication that the full power noise target of 95 PNdB at 500 ft has been achieved.

In addition the variable pitch mechanism has been extensively tested at all speeds on an electric drive and has been shown to have all the intended characteristics of response etc.

This situation has given high confidence that all of the objectives of the programme will be achieved.

CONCLUDING REMARKS

The Variable Pitch Fan can claim to produce a variable cycle engine. The benefits of this variability, while not in any one case overwhelming, are numerous and in some cases difficult to quantify without designing a complete system and checking its operation.

There are certain specific applications where such an engine could show to advantage.

Demonstrator programmes are in hand to study the feasibility of the new features such as variable mechanisms, fan aerodynamics, gear boxes, noise and reverse thrust.

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ACKNOWLEDGMENTS

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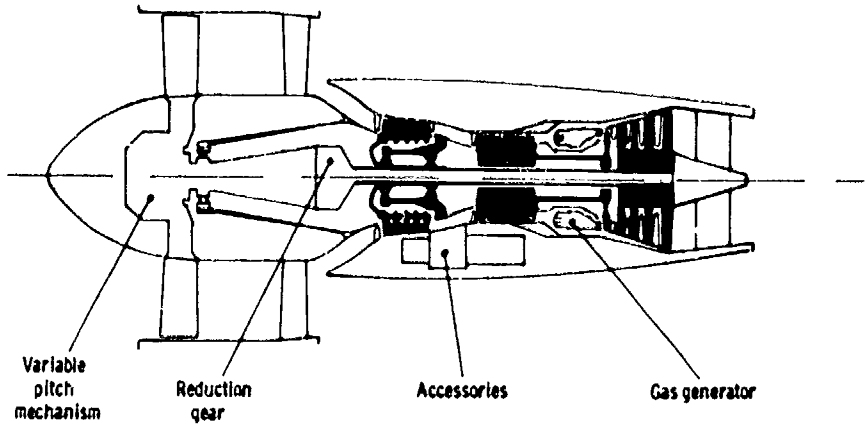
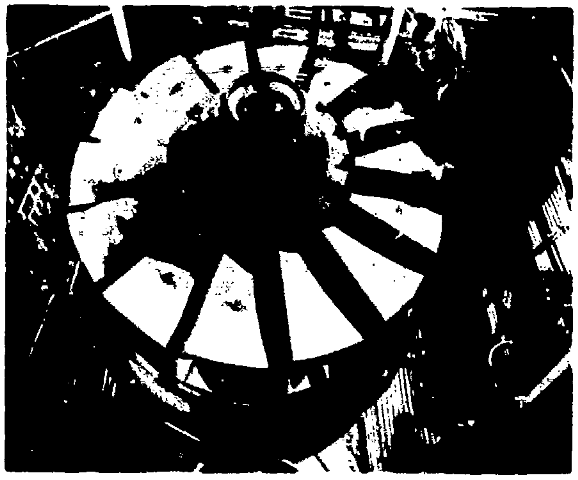
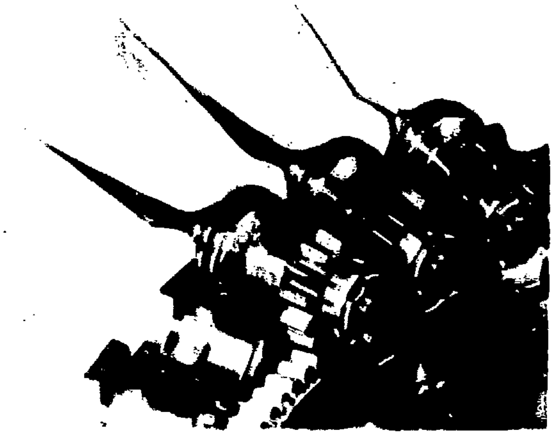


Figure 1 The M45S variable pitch geared fan engine



Assembly of variable pitch fan



Fan pitch change mechanism

Figure 2
M45 SD-02 demonstrator engine components

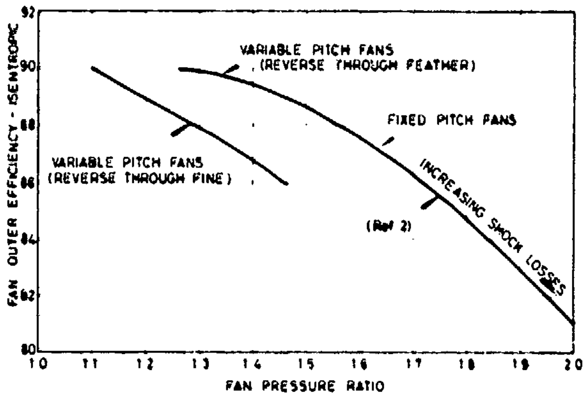


Figure 3 Predicted maximum single stage fan efficiency

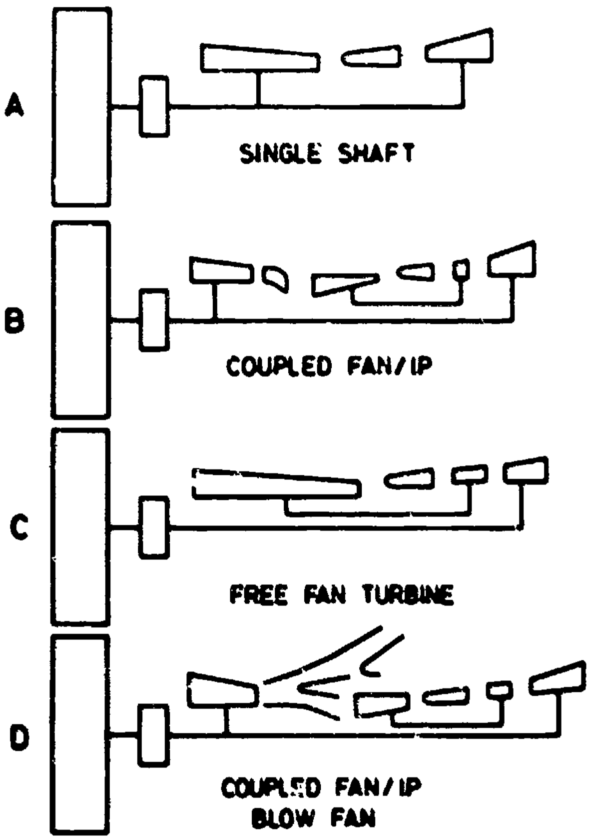


Figure 4
Alternative V.P. fan engine arrangements

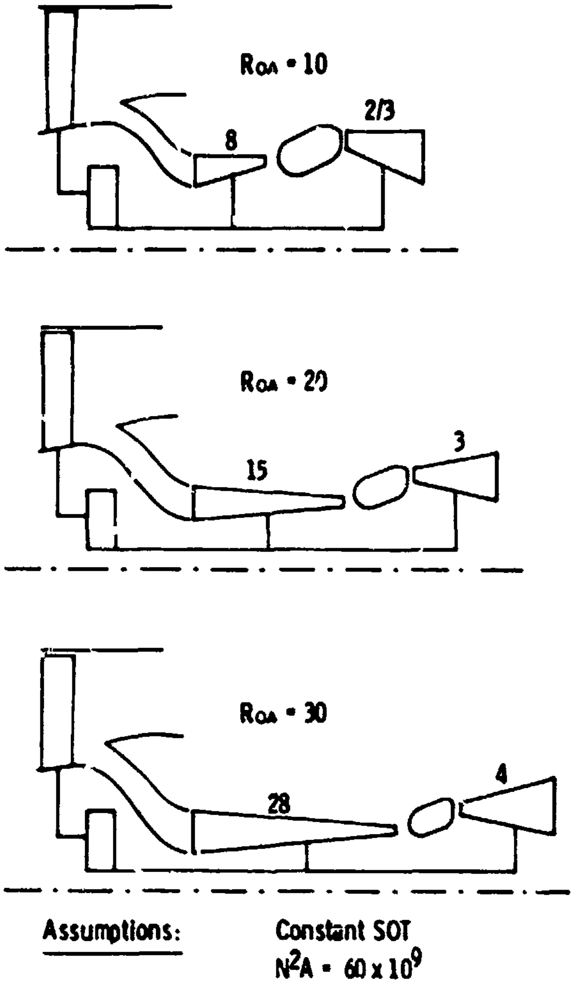


Figure 5 Single shaft engines - effect of pressure ratio variation

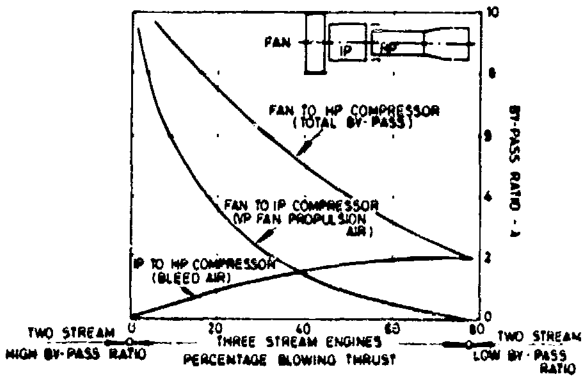


Figure 6
Blowing and propulsion engines.
The 'blowfan' typical bypass ratio variation

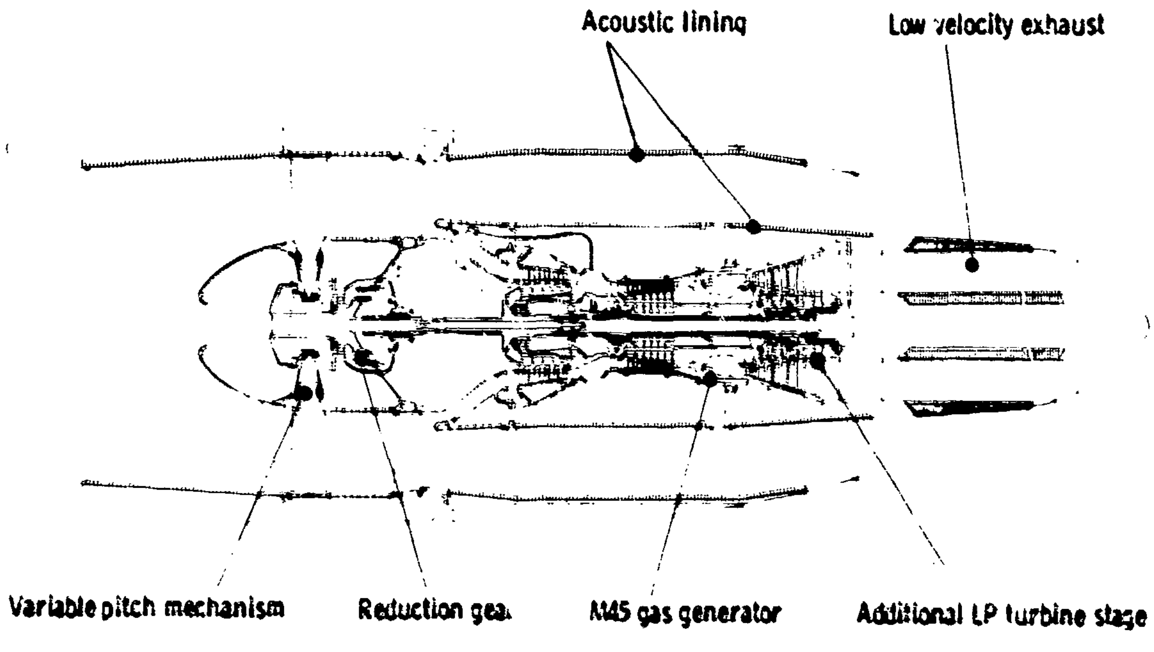
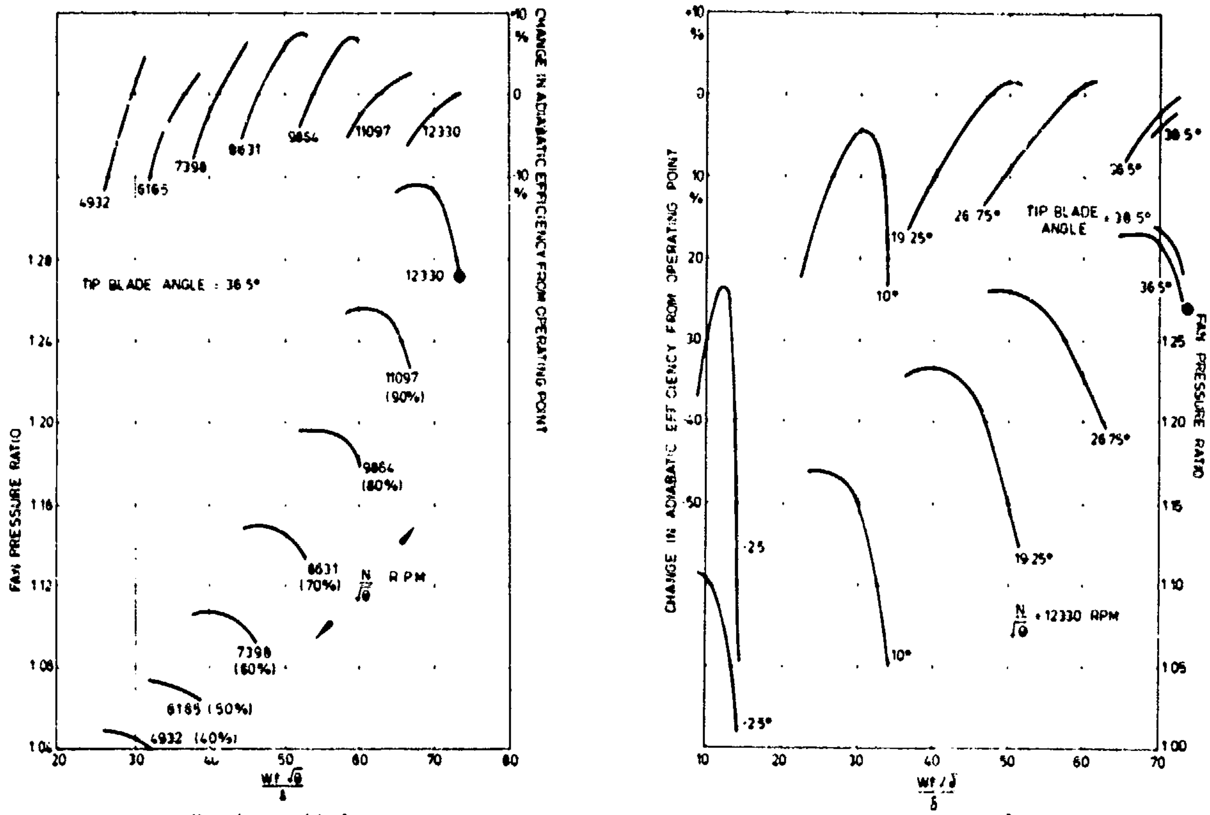


Figure 7

M45 SD-02 demonstrator engine arrangement



Design pitch   Design speed

Figure 8

M45 SD-02 model fan characteristics

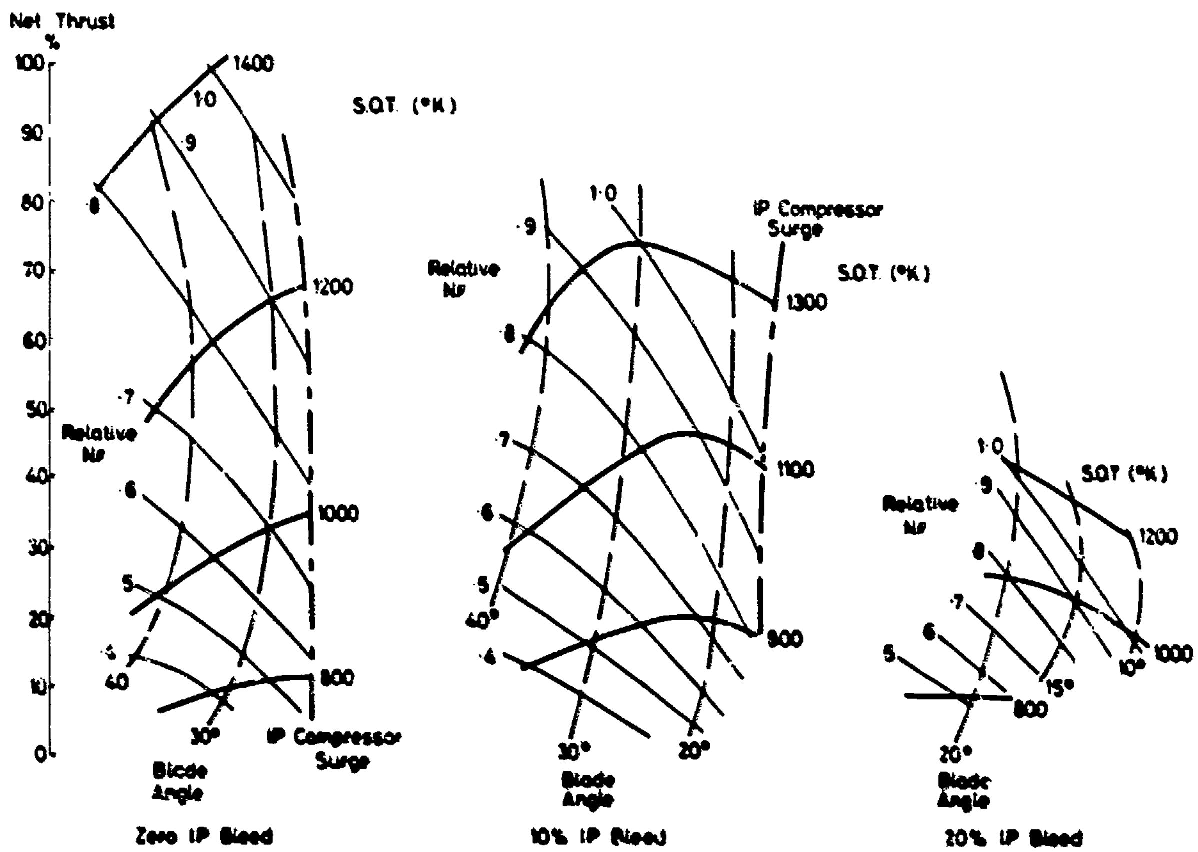


Figure 9

V.P. fan engine cycle variation characteristics

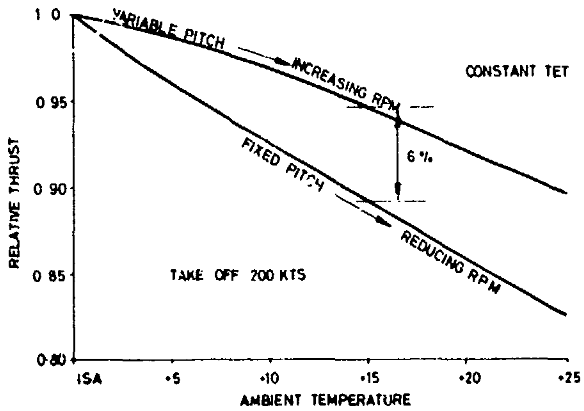


Figure 10
Comparison of variable pitch and fixed pitch fan engines

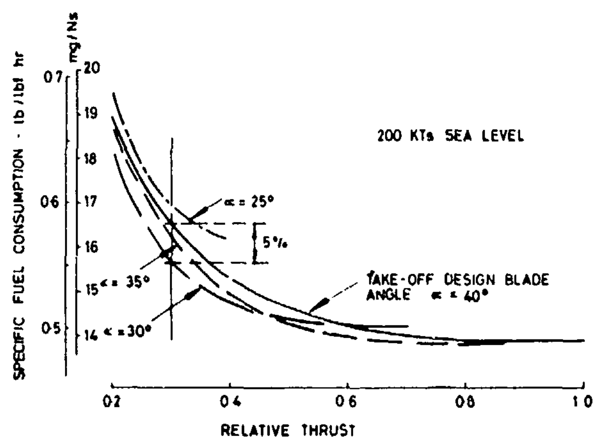


Figure 11
Variable pitch fan - optimum performance at low thrusts

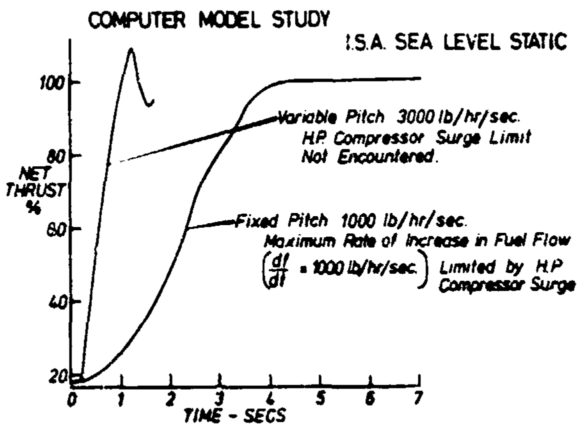


Figure 12
M45S variable pitch fan engine acceleration characteristics

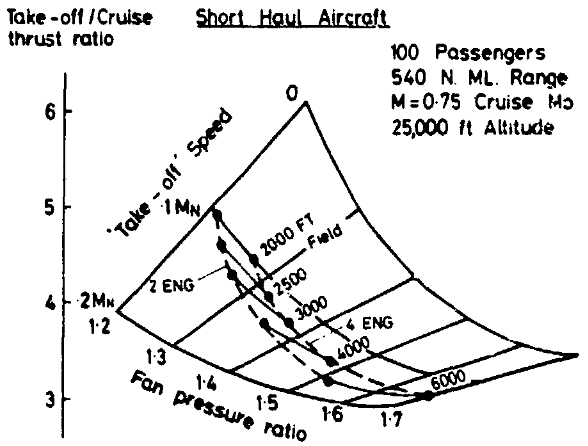


Figure 13
Engine cycle selection

FLIGHT PLAN PHASE	V.P. BETTER THAN F.P.	V.P. EQUAL WITH F.P.	V.P. WORSE THAN F.P.	AIRCRAFT INDICATIONS (SEE DETAILED REVIEW)
ENGINE STARTING*	✓			F.P. MAY REQUIRE MORE POWERFUL STARTER
GROUND MANOEUVRES	✓			V.P. PERMITS REVERSING WITHOUT GROUND EROSION
TAKE-OFF:				
ENGINE RUN UP TIME*	✓			1-2 SECS. FOR V.P. COMPARED WITH 5-10 SECS. FOR F.P.
THRUST LEVEL	✓			V.P. ENABLES OPTIMUM ENGINE MATCHING
NOISE		✓		-
ABORTED TAKE-OFF*		✓		F.P. REQUIRES HIGH RESPONSE REVERSER

* DENOTES SUBJECT OF PRACTICAL DEMONSTRATION (M45 SD-02)

Figure 14
Merit assessment of V.P. fan engined aircraft

FLIGHT PLAN PHASE	V.P. BETTER THAN F.P.	V.P. EQUAL WITH F.P.	V.P. WORSE THAN F.P.	AIRCRAFT INDICATIONS (SEE DETAILED REVIEW)
CLIMB: CUT BACK NOISE LEVEL	✓			V.P. PERMITS OPTIMUM PITCH/R.P.M. FOR MIN. NOISE
CRUISE: THRUST LEVEL *	✓			V.P. ENABLES OPTIMUM MATCHING TO BE SELECTED AT CRUISE CONDITIONS
DRAG		✓?	✓?	V.P. LARGER DIAMETER IF "REVERSE THROUGH FINE" USED
FAN EFFICIENCY		✓?	✓?	V.P. 1-2% WORSE IF "REVERSE THROUGH FINE" USED
ENGINE FAILURE	✓			V.P. CAN FEATHER TO REDUCE DRAG
DESCENT: THRUST LEVEL *		✓		F.P. CAN ACHIEVE EQUALITY BY SPOILING
CABIN CONDITIONING ◉	✓			F.P. PROVIDES COLDER BLEED AIR FROM I.P. AT HIGH R.P.M.
ICING	✓			V.P. SELF SHEDDING DUE TO HIGH FAN SPEED

* DENOTES SUBJECT OF PRACTICAL DEMONSTRATION (M45 SD-02)

◉ DENOTES MERIT ONLY WHEN IP COMPRESSOR & FAN ON SAME SHAFT

Figure 15
Merit assessment of V.P. fan engined aircraft

FLIGHT PLAN PHASE	V.P. BETTER THAN F.P.	V.P. EQUAL WITH F.P.	V.P. WORSE THAN F.P.	AIRCRAFT INDICATIONS (SEE DETAILED REVIEW)
FINAL APPROACH: THRUST LEVEL *	✓			F.P. CANNOT ACHIEVE LOW ENOUGH THRUST WITHOUT NOISY SPOILING
THRUST RESPONSE FOR CONTROL *	✓			F.P. REQUIRES HIGHER POWER SETTING & HIGH RESPONSE FAN DUCT SPOILER
THRUST RESTORATION (WAIVE OFF) *	✓			F.P. REQUIRES HIGH & VERY NOISY THRUST SPOILING FOR RAPID RESPONSE
NOISE *	✓			HIGH POWER & SPOILING MAKES F.P. NOISY
WING PERFORMANCE	✓			SPOILT THRUST COULD HAVE ADVERSE EFFECT ON WING FLOW FIELD.
LANDING: REVERSE THRUST *	✓			V.P. CAN REVERSE DOWN TO ZERO AIRCRAFT SPEED
TIME TO MAX. REVERSE *	✓			ONE SECOND FOR V.P.
REVERSE CANCELLATION	✓			NO INGESTION PROBLEMS WITH V.P. EVEN AT ZERO AIRCRAFT SPEED
NOISE *		✓		-

* DENOTES SUBJECT OF PRACTICAL DEMONSTRATION (M45 SD-02)

Figure 16
Merit assessment of V.P. fan engined aircraft

DISCUSSION

A.H.Jackson

Of course we are great proponents of the variable pitch fan ourselves and we are continuing to work on it very actively. We have had a great deal of difficulties over the years showing a performance advantage at 8/10 Mach number and we pushed this very very hard. We would really like to do this and the only way we have been able to make it show any improvement over the conventional turbofan and propulsive efficiency at all is to do exactly the same what you have been implying and what Mr Gray was covering this morning: a greatly shrunkened or shortened shroud down to length/diameter ratio probably under 1. Is that in the major gains you are talking about which are in the range of 10 to 15%? Is that all coming from that or are there other areas in as well?

Author's Reply

Well, a few years ago when we were involved in these STOL studies everybody was comparing engines with very long cowls which had a large amount of acoustic treatment and I think that the results of these studies showed that you very rapidly lost the thrust SFC advantage of this sort of engine at high Mach numbers, because of the installation penalties. The weights of those engines were certainly not very competitive. I think you can still come down to a quite low fan pressure ratio and maintain a reasonable thrust to weight ratio but the difference is that we have now discarded those objectives of ultra-low noise without discarding the idea of meeting new FAR 36 reduced require-ments. With minimum length fan cowls, our estimates indicate that the penalty for installing a high bypass engine is very flat. Therefore any gain that you get in the propulsion cycle or the thermodynamic cycle is a direct gain relative to the conventional lower bypass turbofan engine.

L'ASTAFAN

Double-flux à pas variable et vitesse constante

par

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présenté par

Henri Dabbadie

TURBOMECA presents here the historical background of the different variable geometry turbo-fan engines which have been experimented for the past twenty years and whose main characteristic is a constant speed operation. The present formula - which is the most sophisticated - is more specially dealt with in this document: it includes a FAN whose rotating blades have a variable pitch, thus permitting a correct adaptation to the different flight configurations.

Le premier "ASTAFAN" a tourné au banc d'essai à la veille du Salon de l'Aéronautique de 1969 et a volé pour la première fois à la veille du Salon 71.

Depuis ces deux dates, des centaines d'essais au sol et en vol ont été effectués et, aujourd'hui, cette expérience unique permet d'y voir plus clair dans le domaine de la propulsion économique entre mach 0,4 et 0,8.

Depuis une décennie, nous sommes habitués, sur les gros transports, à la formule Double Flux qui a apporté un gain substantiel en consommation, mais ce système de propulsion était réservé à l'aviation commerciale.

Pour l'aviation d'affaires, les exigences demandées au moteur étant différentes, il était nécessaire de rechercher un nouveau compromis coût-efficacité.

Si, dans le passé, les moteurs civils pouvaient être dérivés facilement des moteurs militaires, il n'en n'est plus de même actuellement du fait en particulier des contraintes nouvelles relatives à l'économie de carburant et à la pollution.

Le problème se complique encore pour les avions d'affaires légers qui demandent des solutions originales à leurs exigences particulières. On ne pouvait donc pas reprendre pour les moteurs de cette catégorie le même dessin désormais classique, en double flux, double corps à vitesse variable.

L' "ASTAFAN" est né pour cette raison.

C'est en fait une famille de moteurs dont les caractéristiques communes sont : vitesse de rotation constante et "FAN" à pas variable entraîné par l'intermédiaire d'un réducteur, car la géométrie variable est nécessaire non seulement pour le démarrage mais aussi pour l'adaptation optimale aux différentes caractéristiques de vol.

Il est bon de rappeler que les premiers essais au sol et en vol d'un moteur à géométrie variable ont été effectués à TURBOMECA au début des années 50 avec l'ASPIN monté sur le FOUGA GEMEAUX.

Dix ans plus tard, au début des années 60, l'Aubisque construit en série pour l'avion SAAB 105 était aussi un monocorps dont le "FAN" était équipé de volets d'entrée variables comme sur l'ASPIN.

Cette technique de la circulation variable reprenait les principes établis avant la guerre par Mr. Szydlowski pour les compresseurs de suralimentation des moteurs alternatifs.

Les progrès de l'aéro-dynamique des compresseurs transsoniques étudiés depuis les années 50, conjugués à ceux de la métallurgie, ont permis de franchir le pas décisif conduisant à la pale actuelle à pas variable.

Des essais systématiques tant au sol qu'en vol ainsi qu'au Centre Officiel Français (CEP), ont été conduits pour déterminer l'influence des nombreux paramètres influant sur le rendement et le bruit tels que :

- des rapports de moyeu compris entre 0,4 et 0,55,
- des vitesses périphériques comprises entre 250 et 375 m/seconde puisque, grâce à plusieurs rapports de réduction on a la possibilité de balayer un grand domaine en utilisant le même générateur de puissance,
- des profils différents pour les pales mobiles et le diffuseur,
- un nombre de pales variables pour ces mêmes éléments,
- les sections de sortie des flux chaud et froid.

Ainsi, à l'heure actuelle, plus de 300 configurations différentes ont été essayées.

L'utilisation d'un réducteur intermédiaire permet d'avoir à sa disposition un paramètre supplémentaire et d'autre part, pour les dilutions élevées envisagées dans cette taille de moteur, sa présence devient indispensable pour avoir l'adaptation voulue aux caractéristiques du décollage et du vol de croisière.

De plus, ce schéma permet la mise au point séparée des deux constituants principaux : le "FAN" et le générateur de puissance, le même générateur pouvant entraîner, nous venons de le voir, des "FAN" à différents diamètres et à des vitesses différentes, de manière à optimiser l'ensemble pour la mission demandée à l'avion.

A ce propos, il est peut-être nécessaire de revenir sur quelques notions simples lorsqu'on compare les "FAN" de différentes dilutions.

Le choix de la dilution n'est en fait que la conséquence du choix du rapport de pression du "FAN" qui lui, est déterminé par le compromis désiré entre les performances au décollage et en vol de croisière.

La Fig. 1 montre les variations de la poussée spécifique c'est-à-dire de la poussée du flux froid par KW fourni en fonction du rapport de pression du "FAN" pour différents mach de vol.

Dans la phase décollage, cette poussée spécifique diminue fortement lorsque le rapport de pression "FAN" augmente, mais, pour le vol de croisière, les courbes sont beaucoup plus plates et l'influence du rapport de pression est beaucoup moins sensible. Ainsi l'optimum théorique est de 1,35 pour mach 0,6 et 1,42 pour mach 0,8.

Mais on voit qu'on peut améliorer, si c'est nécessaire, de 10 % la poussée au décollage, en diminuant le rapport de pression en ne perdant que 1 % en vol de croisière.

On voit ici comment optimiser le rapport de pression du "FAN" en fonction de la mission, l'augmentation de la dilution entraînant un diamètre et un poids plus grands pour la partie "FAN". Cette dilution sera ensuite donnée par la puissance spécifique du générateur, comme le montre la Fig. 2.

En dehors de toute considération de démarrage, le calage variable des pales du "FAN" est une nécessité pour s'adapter aux lignes de fonctionnement correspondant au décollage et au vol, puisque le rapport de pression du "FAN" n'étant pas sonique, c'est le moyen d'avoir l'équilibre des puissances et des débits.

La Fig. 3 représente les variations de débit pour un "FAN" de rapport de pression de 1,5 nominal de dilution de 5 à 7 suivant la puissance spécifique du générateur.

En vol et à mach 0,7 par exemple, le débit réduit augmente de 10 % et le rapport de pression passe à 1,56, point se situant encore dans une zone de bon rendement. Une géométrie variable n'est donc pas nécessaire.

Par contre (Fig. 4), si le rapport de pression nominal choisi pour le "FAN" est plus faible, 1,3 par exemple, s'éloignant donc de la valeur sonique, le point de fonctionnement se situerait sans pales variables au point C, en vol de croisière.

Grâce au pas variable, le diagramme compresseur se déplace ainsi que les iso-rendements vers les grands débits et le nouveau point de fonctionnement pourra se situer en C' avec un bon rendement.

Pour des points de croisière à mach plus faible, le rapport de pression du "FAN" pourra être choisi inférieur à 1,3 pour avoir le rapport $\frac{\text{poussée décollage}}{\text{poussée croisière}}$ désiré.

Le "FAN", étant donc à pas variable, peut être entraîné par une turbine à un seul arbre, le démarrage s'effectuant facilement en position "petit pas".

Ce schéma, comparable au turbo-propulseur à un seul arbre, en possède toutes les caractéristiques de simplicité, de régulation et de facilité de pilotage.

La régulation de la turbine maintient la vitesse de rotation constante à la valeur choisie pour les conditions optimales de décollage, de montée ou de croisière.

La manette de puissance agit directement sur le pas du "FAN", les variations de poussée demandées pouvant s'effectuer ainsi en une fraction de seconde, puisque la vitesse de rotation est maintenue constante.

En configuration de descente, toujours à vitesse de rotation constante, on peut choisir un pas de "FAN" négatif donnant un freinage appréciable permettant d'augmenter l'angle de descente, sans nécessiter des aérofreins.

Pendant la descente, le conditionnement de la cabine est correctement assuré car le rapport de pression du compresseur, à vitesse constante, ne chute que faiblement, 20 % environ, entre le pas de croisière et le pas de descente du "FAN".

Ceci est un avantage important car avec les moteurs double corps on est obligé, en descente, de maintenir une vitesse de rotation élevée pour assurer une combustion correcte, le conditionnement et éventuellement le dégivrage, ce qui entraîne une poussée résiduelle importante nécessitant l'utilisation d'aérofreins puissants et une consommation inutile de carburant.

Avec les double corps à vitesse variable, on est conduit pour les mêmes raisons à un cycle thermodynamique à fort taux de compression pour que, en descente et à poussée résiduelle admissible, la pression de sortie compresseur soit suffisante. Ceci entraîne, pour une même puissance nominale, une augmentation du nombre d'étages du compresseur et donc des turbines, ce qui augmente la complexité de la turbine et son prix, mais aussi son poids spécifique, d'autant plus que la puissance spécifique par Kg d'air primaire est maximale pour un rapport de pression voisin de 12 à 14 pour les rendements et températures actuels.

Tenant compte de ces considérations, nous avons entrepris, depuis 1971, une expérimentation en vol sur un, puis récemment, un deuxième AERO-COMMANDER sur lesquels des "ASTAFAN" de différents types ont remplacé les turbo-propulseurs d'origine.

Les premiers "ASTAFAN" utilisaient comme générateur de puissance l'ASTAZOU 16 de 1000 ch. Actuellement, l'expérimentation se poursuit avec l'ASTAZOU 20 de 1400 ch, qui, en turbine de base, a une consommation spécifique de 210 g/ch.h (.47 Lb/HP/h) bien que le rapport de pression nominal ne soit que de 9,6.

Entraînant un "FAN" donnant 1,25 de pression, avec une dilution de 11, la poussée au point fixe est supérieure à 1000 Kg pour une consommation spécifique de 0,31.

Comparé à un double flux à géométrie fixe dont le "FAN" a un rapport de pression de 1,5/1,6 et une dilution de 3 de même compression totale et de même poussée en croisière (par exemple à 30 000 pieds et mach 0,65), on obtient 35 % de plus de poussée au point fixe, 25 % à mach 0,2, 10 à 15 % en montée pour un poids de 10 % supérieur.

La descente peut se faire avec une pente augmentée grâce à la possibilité de freinage, tout en diminuant la consommation.

Quant au niveau du bruit, les vitesses périphériques d'un "FAN" donnant 1,25 de pression sont voisines de 280 m/sec. (Fig. 5) contre 450 m/sec. pour 1,55, les conditions des normes acoustiques officielles sont plus facilement respectées.

Pour toutes ces raisons, cette formule de "FAN" à pas variable et vitesse constante nous paraît un bon compromis en ce qui concerne le bruit, le poids spécifique, la consommation et les performances relatives de décollage et de croisière. Le pilotage est facilité par la possibilité de varier rapidement le niveau de poussée, augmentant le facteur de sécurité particulièrement à l'atterrissage (Fig. 6).

Ces considérations sont la synthèse de 7 ans d'essais et de plus de 10.000 heures de mise au point et d'endurance dont 2.500 en vol.

Pour terminer, nous évoquerons une utilisation non encore expérimentée, mais qui pourrait être très importante pour les avions STOL utilisant l'hypersustentation par soufflage.

Le Double Flux monocorps à vitesse constante offre la possibilité de prélèvement d'air important au décollage mais aussi à l'atterrissage et à poussée réduite, car la vitesse de rotation étant maintenue constante, cet air est sensiblement au même niveau de pression dans les deux configurations et le débit de soufflage est donc également le même dans les deux cas, et pourrait, même, être supérieur à l'atterissage, puisque le niveau thermique du moteur est plus faible.

Ceci ne peut être obtenu que grâce à la possibilité de maintenir, à poussée réduite, la vitesse maximale de rotation du moteur.

C'est pour toutes ces raisons que Mr. Szydlowski est particulièrement attaché à cette formule.

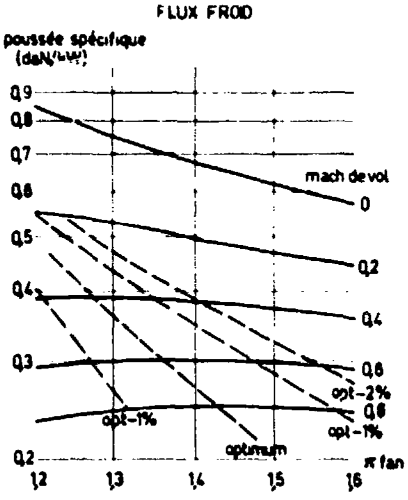


Figure 1

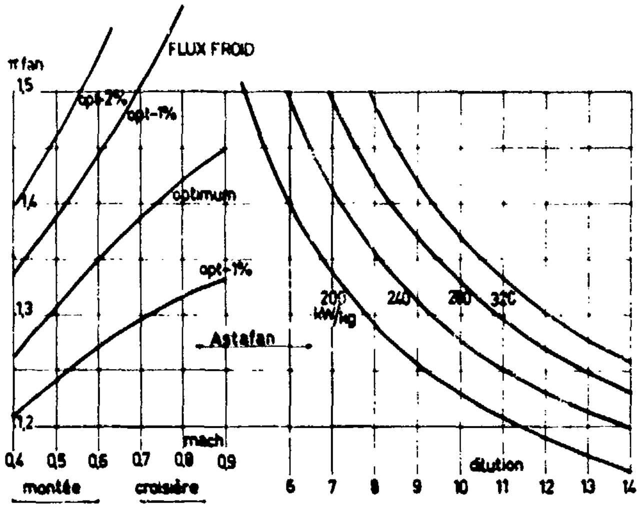


Figure 2

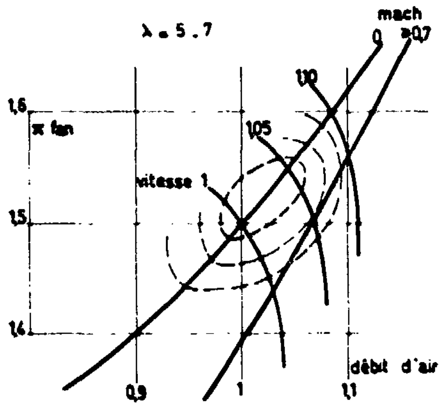


Figure 3

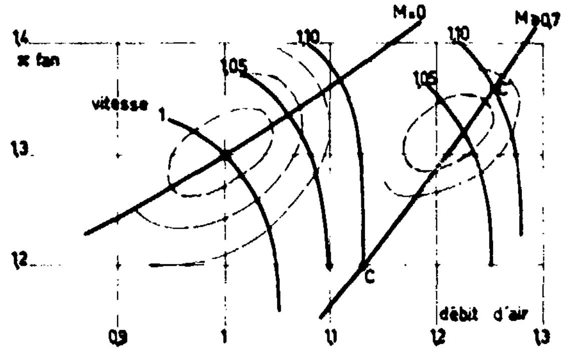


Figure 4

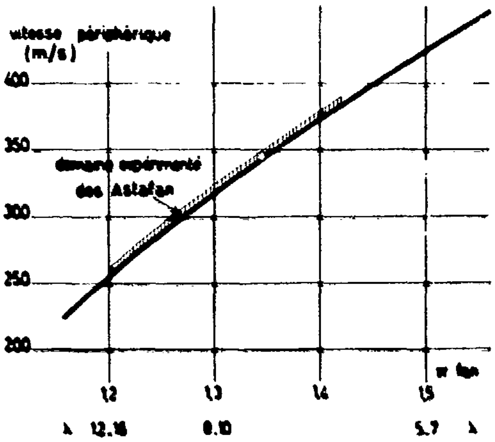


Figure 5

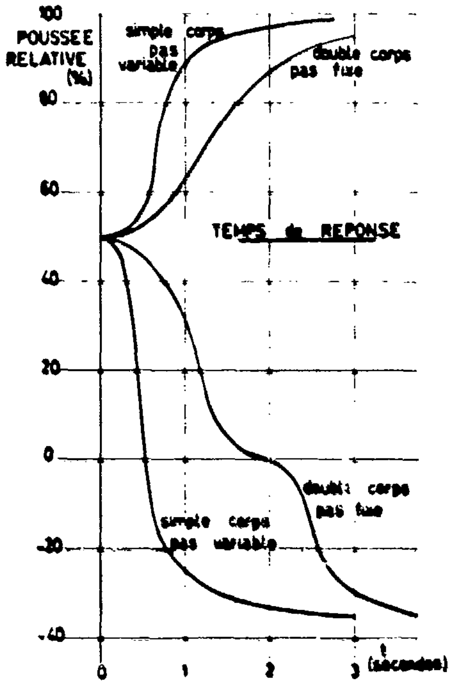
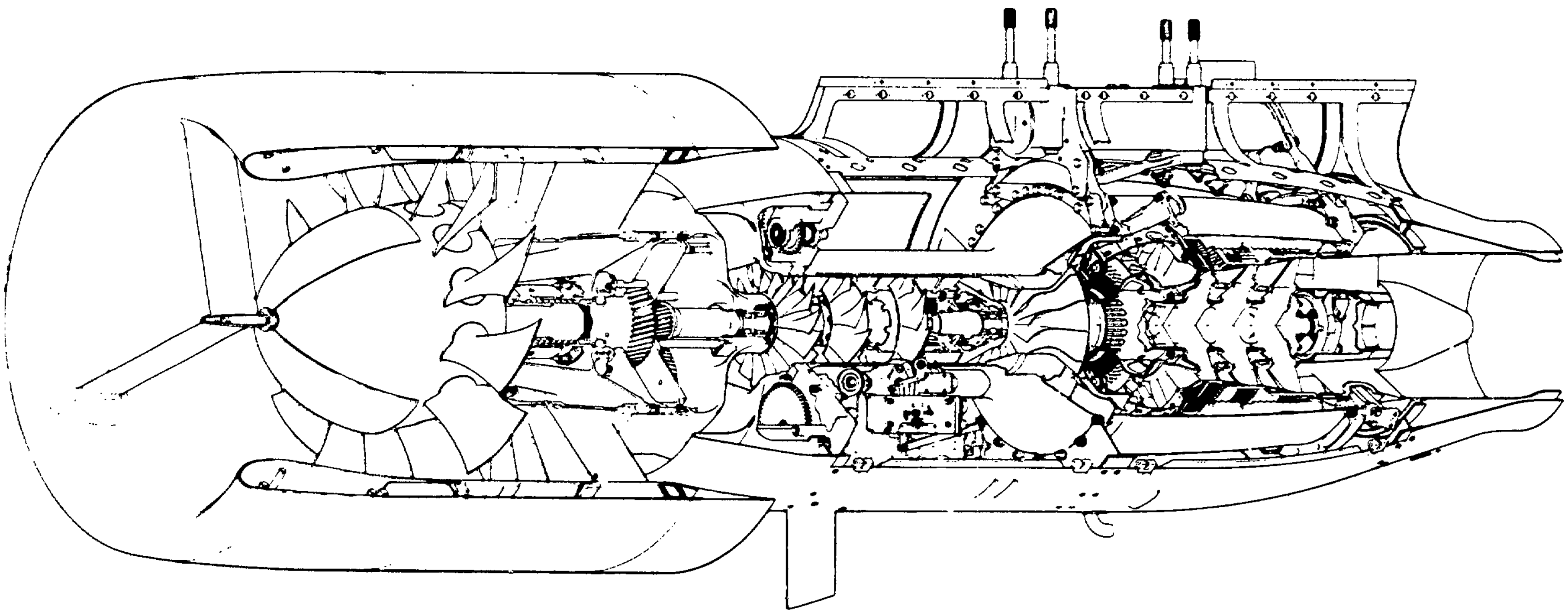


Figure 6



ASTAFAN IV : Performances au point fixe : Poussée : 1.020 kg

C_s : 0,31 kg/kg.h

LE PULSO-REACTEUR POUR REGIME SUBSONIQUE ELEVE AVEC PASSAGE AU FONCTIONNEMENT EN STATO-REACTEUR - UN MOTEUR DE FAIBLE PRIX DE REVIENT -

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RESUME

L'auteur expose brièvement les résultats d'études préliminaires menées sur un réacteur sans soupape, à combustion pulsée influencée par la fréquence de l'onde de choc frontale, avec passage au régime continu, dont la gamme de vitesses atteint Mach 0,9.

A divers stades des essais fondamentaux, on a étudié en détail les paramètres essentiels et déterminé leur influence sur les caractéristiques de poussée et la consommation de carburant. On a pris soin de conserver la simplicité de conception du moteur d'essai afin d'obtenir un système propulsif peu coûteux pour missiles guidés. Les recherches ont été menées sur des montages d'essais avec jet libre et vitesse réduite à l'arrière du moteur du fait de l'écoulement limité. Les essais ont démontré que les moteurs de ce type présentent des caractéristiques fonctionnelles stables.

Il ressort des résultats d'essais que l'on peut utiliser ce moteur dans deux versions: soit comme moteur consommable, soit comme moteur ré-utilisable de faible prix de revient. Au cours des essais fondamentaux, on a également réalisé le passage aux conditions du statoréacteur. En outre, l'une des versions de ce moteur peut être conçue comme un générateur de gaz intermittent pour turbines à gaz de petites dimensions dont les gaz d'échappement actionnent un rotor de turbine.

LISTE DE SYMBOLES

A	=	Section
F_{Brut} (kp)	=	Poussée brut
F_N (kp)	=	Poussée nette
$F_{Résist.int.}$	=	Résistance interne
M	=	Nombre de Mach d'écoulement
m/m	=	Rapport de masses correspondant aux débits
P (ata)	=	Pression
$P_{cab.}$ (atm rel.)	=	Pression d'injection de carburant
P/P	=	Rapport de pressions
SFC (kg/Kph)	=	Consommation spécifique de carburant
t (°C)	=	Température
T (°K)	=	Température
t (ms)	=	Temps
V (m/s)	=	Vitesse

1. HISTORIQUE

La première méthode pour la génération de gaz chauds moyennant une combustion pulsatoire fut documentée dans un brevet de Esnault-Pelterie, il y a environ 70 ans, et c'est en 1909 que Marconnet a développé le premier pulso-réacteur pour la génération d'une poussée directe. Plus tard, en 1930, SCHMIDT obtint un brevet concernant des "réacteurs à fonctionnement périodique" et ses études formaient la base du pulso-réacteur qui devait être réalisé plus tard sous le nom de "type As-014" de la maison ARGUS, à fonctionnement selon le principe de clapets (1/2). En France, la Société SNECMA commença le développement de pulso-réacteurs sans éléments mécaniques (clapets) en 1942, études qui furent intensifiées depuis 1947. Des réalisations éprouvées telles que les pulso-réacteurs 3340 Escopette, Ecrevisse et HS 1 dénotèrent de nouvelles méthodes de développement. En 1962, la nouvelle répandue, concernant des essais avec un pulso-réacteur dont l'entrée était d'une configuration spéciale, fut accueillie avec grand intérêt.

Grâce à cette nouvelle configuration aérodynamique, il était possible d'obtenir un fonctionnement des réacteurs avec des vitesses d'écoulement égales à M = 0.85, avec une poussée nette croissante. (3/4).

Dans les années de 1955 et 1956, des études à l'Instituto Aerotécnico de Córdoba en Argentine avaient pour but des séries de mesures sur les deux systèmes, c'est-à-dire des réacteurs à et sans clapets, sur des réacteurs d'essai, séries dont les résultats furent comparés les uns aux autres, pour autant que c'était possible, tout en gagnant de nouvelles expériences. Cependant, par manque d'une soufflerie, les recherches durent se limiter au point fixe (5) .

En Amérique du Nord et en Angleterre, des études technico-scientifiques ont été réalisées aussi dans ce domaine au cours des années passées.

2. OBJET DES RECHERCHES SUR DES PULSO-REACTEURS DANS LA RFA, À PARTIR DE 1945

Aprés des travaux préparatoires concernant des recherches sur des pulso-réacteurs, des essais furent entamés en 1970 dans la RFA, avec le but d'étudier des pulso-réacteurs sans clapets, études qui se basaient sur des brevets accordés en connexion avec des travaux réalisés et licenciés en Argentine.

Il s'agissait là d'un programme d'expérimentation qui devait être réalisé en collaboration avec la DFVLR à Braunschweig. L'object des recherches était intitulé "Recherches sur des composants d'un stato-réacteur à plusieurs phases de combustion". Il était clair que l'air d'écoulement dont on pouvait disposer auprès du Luftzentrale (Centrale d'Air) à Trauen devait servir exclusivement pour l'alimentation d'air du réacteur d'essai.
Pour la pleine alimentation du tuyau d'admission, cette quantité d'air était suffisante jusqu'à un nombre de Mach $M = 1,5$. Tout d'abord les écoulements d'air sur le réacteur et surtout sur la partie arrière de celui-ci ne furent pas pris en considération. Conformément à ces conditions, le banc d'essai était donc disposé de la sorte à offrir les conditions d'un banc pour réacteurs à traînée libre.

Le programme d'essai projeté devait démontrer les phénomènes gazodynamiques surgissants dans le tuyau pendant les essais du réacteur et en plus, il fallait traiter des problèmes de combustion tels que stabilisation de flamme et changement du fonctionnement intermittent en fonctionnement continu, pour surmonter de cette manière la limite de vitesse qui se trouvait, pour des modèles de série, aux environs d'un nombre de Mach $M = 0,5$ (2), tout en transposant cette limite au voisinage de la vitesse du son. A côté de ce programme, l'on menait dans la RFA des recherches simultanées sur un pulso-réacteur à composants mobiles, recherches qui devaient atteindre un but pareil. (6).

3. BUTS DU PROGRAMME D'EXPERIMENTATION

Afin de saisir le changement d'état thermodynamique ou bien de déterminer un état gazodynamique d'un écoulement constant, il suffit d'avoir une notion d'une part des grandeurs de la vitesse, de la pression statique, de la température statique respectivement de la pression totale, de la température totale et du nombre de Mach dans une section d'écoulement déterminée.
Pour des propulseurs à combustion pulsatoire, un autre paramètre est requis, c'est-à-dire un changement d'état gazodynamique à tout temps et partout. En plus, les sections de l'écoulement à l'intérieur du réacteur sont en partie très difficiles à déterminer en vue de l'existence simultanée d'écoulements partiels qui sont contraires l'un à l'autre. Le procédé de prise de la moyenne ou d'approximation ne peut être utilisé que sous certaines conditions pour déterminer ces écoulements instationnaires. Il n'est donc pas possible de réaliser un calcul exact de tous les processus qui se déroulent. Par des conditions d'une interaction rapide et successive des écoulements tant pour des tuyaux ouverts dans un sens et tout particulièrement pour ceux ouverts dans les deux sens (modèles sans clapets), les différents composants d'un réacteur comparés à un milieu à écoulement unilatéral présentent, en partie, un double mode de fonctionnement.
Ainsi, l'entrée est non seulement à concevoir pour une admission maxima d'air frais, mais aussi d'une manière convenable pour les gaz d'échappement sortant en amont. Le jet du carburant introduit à l'entrée doit avoir des qualites d'une diffusion favorable à l'égard de la formation momentanée du mélange tout en produisant un effet freinant, comme celui d'un clapet d'étranglement, pour des gaz d'échappement en amont. La partie arrière de la chambre de combustion présente, elle aussi, un fonctionnement multiple pour autant qu'elle est d'une certaine robustesse, ainsi que la partie d'échappement arrière, des caractéristiques qui sont dûes à l'influence de l'air et des gaz résiduels en aval. Dans ce cas là, selon le principe du contre-courant, de l'air et des gaz résiduels et périphériques peuvent pénétrer dans l'entrée déjà pendant la sortie des gaz chauds. De différentes méthodes d'approximation et des études concernant des projets de pulso-réacteurs furent réalisés pendant les années précédantes en se basant sur les caractéristiques et les performances de ces réacteurs, méthodes dans lesquelles les phénomènes d'écoulement et de combustion de même qu'une évaluation des autres possibilités de développement furent plutôt négligés. Cet état de choses était la raison proprement dite pourquoi la tâche citée en haut fut déroulée de la sorte que son objet principal consistait en un programme d'expérimentation.

La consommation spécifique de carburant favorable jusqu'alors, pendant le fonctionnement au point fixe des réacteurs sans clapets vis-à-vis de ceux avec clapets indiquait, pour une conception de missiles, une certaine supériorité des réacteurs cités en premier lieu, pourvu qu'il était possible d'obtenir des valeurs à peu près pareilles avec l'écoulement.

En même temps il fallait limiter la combustion à la chambre de combustion elle-même, tout en évitant un déplacement de la flamme dans le sens d'échappement. Ce phenomène s'était produit sur le réacteur AS-014, avec extinction consécutive du réacteur.

Les consommations spécifiques de carburant atteintes pendant le fonctionnement au point fixe étaient les suivantes pour des réacteurs sans clapets (4) (5):

SNECMA 3340 Escopette	b_o	= 1,80 Kg/Kph
SNECMA HS 1		= 1,45
Hiller Aircraft		= 1,0
IAME Escopeta		= 1,62

pour des réacteurs à clapets selon (2/5/7):

Argus As-014	b_o	= 3,0 Kg/Kph
Pulso-réacteur Arsênal		= 2,34
I A M E (F_o=240 kp)		= 2,5

Une comparaison de ces consommations spécifiques de carburant prouvait que les valeurs de consommation des réacteurs sans clapets se montaient jusqu'à la moitié environ de celles de réacteurs à clapets.

4. PROGRAMME D'ESSAIS REALISE AVEC DES VITESSES MOYENNES (RECHERCHES DE COMPOSANTS)

4.1 Fonctionnements-réacteurs à des vitesses d'écoulement à l'entrée correspondantes à M = 0.57

Les premières séries de mesures exécutées sur le terrain d'essai de la DFVLR à Trauen furent réalisés avec un réacteur d'essai dont la configuration était celle du I A M E Escopeta, à l'exception de quelques divergences insignifiantes (Planche de fig. 1). Pour les essais l'on pouvait s'appuyer sur des recherches partielles menés à Córdoba. Le developpement du système d'injection du carburant sous pression élevée comporta des résultats favorables.

En connexion avec la section type "Borda", le jet de combustible échappant qui se déplaçait à la façon d'un jet oscillant d'un cône creux, servait d'un vrai clapet d'étranglement vis-à-vis des gaz sortant vers l'amont. Par le positionnement de ce brûleur et grâce à une augmentation de la pression du carburant jusqu'à une valeur de 30 atm rel. pour des vitesses d'écoulement élevées, cet effet d'étranglement put être conservé, de manière que les séries de mesures pouvaient être exécutées jusqu'à une valeur de M = 0.57. La planche de figure 1 montre aussi les consommations spécifiques de carburant pendant la période en question en fonction de la pression d'injection du carburant et les poussées statiques ainsi que l'examen conduit moyennant une barre d'essai pendant le fonctionnement du réacteur au point fixe, afin d'obtenir des résultats concernant le déploiement du cône de carburant, la formation du mélange et la combustion de même que la récirculation, phénomènes visibles par les traces d'oxydation survenues après un fonctionnement du réacteur pendant une durée de 5 minutes.

4.2 Fonctionnements-réacteurs à des vitesses d'écoulement à l'entrée correspondantes à M = 0.7 et plus hautes

Il s'avérait utile de monter un brûleur supplémentaire et axialement orientable (voir schéma Planche de Fig. 2). En vue de faciliter les conditions d'essai, le tube d'admission télescopique ainsi que le deuxième brûleur furent fixés (voir schéma en dessous, Planche de fig. 2). Pendant d'autres essais d'écoulement l'on a pu atteindre M = 0.72. Après montage de stabilisateurs rigides de flamme dans la chambre de combustion, des essais-réacteurs furent réalisés avec des vitesses d'écoulement jusqu'à M = 1, essais au cours desquels la consommation de carburant augmenta sensiblement à cause du type d'accroche-flamme choisi (Planche de fig. 3, diagramme à gauche). Par contre, ce montage permit un perfectionnement de la chambre de combustion quant à une variation de son volume et de sa configuration (des configurations élancées ou convexes). La stabilisation des flammes désirée fut obtenue à tous les égards avec des vitesses d'écoulement jusqu'à M = 1, citées en haut.

4.3 Injection du carburant

Ces essais étaient précédés par une autre étude partielle, qui avait pour but le choix optimum des brûleurs à employer. Là il était possible de se servir des études de Kling (8) pour obtenir des éclaircissements sur l'angle de diffusion du carburant, la grandeur des gouttes, le diametre "Sauter", des diffuseurs en et sans giration ainsi que des diffuseurs radiaux. En partie, ces diffuseurs furent conçus et fabriqués pour les essais selon Planche de fig. 4.

4.4 Angle d'inclinaison à la partie AR de la chambre de combustion

Au cours de ces séries de mesures, des essais permirent de déterminer l'angle d'inclinaison (demi-angle de cône) sur la partie arrière de la chambre de combustion. Le diagramme sur le côté droit de la Planche de fig. 3 montre 6 configurations de tuyères, parmi lesquelles se trouve aussi celle du réacteur As-014 à clapets. Les résultats des essais démontrèrent que pour des pulso-réacteurs fonctionnant au voisinage de la vitesse du son l'angle d'inclinaison devait être au moins 20°.

4.5 Accroche-flamme, tuyau perforé

Bien que le réacteur pût être expérimenté à des vitesses d'écoulement très élevées et la combustion eût lieu dans la chambre de combustion proprement dite, ce qui était verifié par des mesures de température, les résultats n'étaient pas satisfaisants. Les consommations de carburant étaient trop élevées (au dessus de 5 Kg/Kph). L'on se décida donc à entamer une autre recherche partielle qui avait pour but de concevoir un accroche-flamme approprié pour une combustion pulsatoire. Des accroche-flammes du type "V" qu'ils sont employés pour des stato-réacteurs restaient inefficaces pour une combustion pulsatoire. La solution fut apparemment trouvée sous forme d'un accroche-flamme carêné qui est illustré dans la Planche de fig. 5.

Au cours des essais il s'avérait que le phénomène d'une couche limite d'une grande intensité de chaleur se manifesta entre le tuyau perforé et la paroi de la chambre de combustion, couche limite dont l'effet était favorable pour l'allumage et la combustion intermittents du mélange frais pendant toutes les phases de combustion d'un cycle. Les consommations spécifiques de carburant ainsi mesurées se trouvaient, à vitesse d'écoulement élevée, en dessous de b_o = 3 kg/kph (par rapport à la poussée brute, moins la résistance intérieure du réacteur). Ce type d'accroche-flamme était un élément stabilisateur de flamme aussi pour des fonctionnements de réacteur à des vitesses d'écoulement supersoniques à l'entrée.

4.6 Comportement de la recirculation

En connexion avec les recherches sur le type d'accroche-flamme une mesure d'écoulement d'air au débit constant fut fait en vue de déterminer le comportement du flux de récirculation en fonction de la grandeur de l'écoulement primaire.

L'intensité de la récirculation fut obtenue par un élargissement subite de la section de tuyau et par la dépression correspondante, ce qui ne survenait cependant qu'avec des vitesses de jet au dessus de 150 m/sec. (Planche de fig. 6)

Trois à cinq pour-cent des masses circulérent autonomement sous l'effet de la dépression comme il a été constaté par les rapports des masses relevées. En vue de l'allumage périodique dans les pulso-réacteurs, ce résultat était très intéressant. Le report de ces résultats sur combustion conduisit à une conception dans laquelle l'accroche-flamme était monté avec des espaces intermédiaires permanents dans la chambre de combustion entre la face avant et la partie arrière de celle-ci.

4.7 Effet de contraction à des vitesses élevées

En vue de déterminer le comportement du mélange lors de son admission dans la chambre de combustion pendant la phase de dépression d'un cycle en connexion avec l'élargissement subite de la section de tuyau l'on se servit de mesures de courant froid réalisées en Argentine qui avaient abouti à déterminer l'influence de la contraction au plan d'élargissement pendant des vitesses d'écoulement élevées et constantes dans le tuyau d'admission (5). Tandis que des études américaines (9) se bornaient à des vitesses inférieures jusqu'à 30 m/sec. environ, les séries de mesures précitées concernaient des vitesses qui surviennent pendant la phase d'admission de l'air frais sur des pulso-réacteurs sans clapets.

Le diagramme (Planche de fig. 7) montrait les distributions de pression au plan total d'élargissement pour des rapports de sections de 3,76 et 5,67 en fonction du rapport de la pression totale dans le tuyau d'admission (points de mesure jusqu'à 2,5); le montage d'essai put être retiré du schéma des points de mesure contenu dans cette note. La configuration de la tuyère (cône) avant l'élargissement évitait un refoulement de la dépression locale, (semblable à la configuration de la partie d'entrée arrière avec brûleur, sur le réacteur-essai). Le haut degré de concentration du jet permit une pénétration profonde de ce jet froid dans la chambre de combustion élargie. (La figure montre la section "a" du tuyau en condition barrée)

5. PULSO-REACTEURS SANS CLAPETS POUR HAUTES VITESSES (D'AUTRES RECHERCHES DE COMPOSANTS)

En fin des recherches partielles sur l'optimalisation de la chambre de combustion et sur stabilisation de la flamme dans la chambre de combustion en cas des hautes vitesses d'écoulement, les rampes d'étranglement sont été démontes des zones chaudes afin de tenir la consommation de carburant aux valeurs basses.

5.1. Optimisation de l'entrée du réacteur

Une vérification et réarrangement de l'entrée étaient nécessaires. Le dispositif d'essai représenté dans la partie haute de la Planche de Fig. 8 confirmait la validité de ce mesure; ce dispositif qui n'était pas favorable au point de vue d'aérodynamique était utilisé pour des essais de fonctionnement qui étaient exécutés jusqu'aux nombres de Mach élevés (M $\leq$ 1) et qui, en même temps, consommaient peu de carburant. Dans cette phase d'essais, le cône représenté dans la partie basse de la pl. 8 était monté avant l'entrée d'air et ajusté tel qu'il était pourvu dans des projets précédents. Des cônes au diamètre de 80 à 140 mm, qui, partie d'eux, étaient aplatis en avant et qui étaient des longueurs différentes, ainsi que des cônes dont les angles étaient constant à 30°, ont été expérimentés. En utilisant cette famille de cônes, des essais pouvaient être exécutés et mesurés jusqu'à la limite de la capacité du compresseur de la soufflerie, de manière simple, où l'entrée d'air était d'une part pratiquement axiale et d'autre part variable et oblique. La distance entre le cône et l'entrée était optimalisée pour chaque essai. Ce dispositif d'essai n'était donc pas axialement variable pendant qu'un essai était en cours.

5.2. Intervalles commandées par aérodynamique à l'entrée du réacteur

En somme, le principe de procédé achevé pour le clapet d'intervalle aérodynamique peut être décrit comme suit. Le dispositif consiste d'une entrée sans pièces mécaniques et d'une chambre de combustion dont l'entrée est une soupape réelle. Dans l'entrée d'air qui est d'une longueur constante ou télescopique sont montées un ou plusieurs injecteurs de combustible fixes ou variables et en ou sans giration. Après allumage du carburant éjecté de ces injecteurs qui en partie est d'une pression forte (éprouvée jusqu'à 100 ata), les gaz brûlés périodiquement de la chambre de combustion à l'entrée sont concentrés à un jet de flammes d'intervalle (flamme pilote) qui a pour but de regler la quantité d'air subséquente et intermittente. En même temps, le cône fixe ou axialement variable avant le collet d'entrée assiste en outre l'action du soupape dans les deux sens des courants alternants.

La limite abrupte de la chambre de combustion à un angle d'environ 45° forme une rampe pour le courant d'air partiel qui entre dans la chambre de combustion ainsi que pour le mélange inflammable dans la phase de dépression pendant un cycle.

La pression d'arrêt (pression dynamique) du mélange frais conjointement avec les gaz résiduels de la phase de combustion précédente ainsi que la température de la paroi qui correspond à la température d'allumage ou qui est plus haute que celle-ci cause un allumage instantané.

Le nombre des allumages par seconde dépend de la vitesse d'écoulement et de la distance du collet d'entrée à la rampe d'étranglement. De cette manière, la rampe est utilisée comme soupape de reglage, et la flamme pilote élastique dans toute sa longueur a l'effet d'un organe de reglage pour le procès d'entrée intermittent.

5.3 Allumage par pression d'arrêt et des gaz résiduels

Pour des réacteurs à clapet, de différentes études ont traitées entre autres (10/7/6) l'effet d'une onde de choc recirculant de la sortie du réacteur, onde qui entame l'allumage en combinaison avec des particules de carbone restés actifs provenants de la phase de combustion précédente, et à laquelle a été conféré la désignation "allumage des gaz résiduels avec l'effet d'une onde de choc". L'oscillogramme contenu dans l'étude de Schmidt (10), page 396, confirme ces constatations, c'est-à-dire que l'allumage dans les pulso-réacteurs est fortement influencé par des phénomènes de ces ondes. Les études françaises de Bertin et Salmon (11), page 128, contiennent un oscillogramme enregistré pendant des essais sur un tuyau sans clapets (voir aussi (12) dont on n'a pas pu retirer les effets d'onde de choc décrits en haut. Le diagramme du rapport pression temps (voir Planche de fig. 9) qui fut établi dans le cadre du programme d'expérimentation effectué avec une consommation de carburant diminuée ne présente, dans la région de la chambre de combustion (plans de référence Q_2 et Q_3 et tracé de ligne 3), aucun phénomène d'onde de la sortie du réacteur qui ait été décisif pour l'allumage.

Les valeurs contenues dans le diagramme ne pouvaient pas être enregistrées en même temps. L'état de fonctionnement fut maintenu constant pendant toute la série des mesures. De cette analyse, il s'ensuivait que, sur des tuyaux ouverts dans les deux sens, l'effet de l'onde de choc n'est pas requis en ce qui concerne le déclenchement de l'allumage. Pour une autre raison, il paraissait opportun de négliger cet effet. A des vitesses d'écoulement élevées, les vitesses du jet dans le tuyau augmentaient, tout en diminuant les vitesses de propagation des ondes de choc qui pouvaient causer un retard de l'allumage. La configuration de la nouvelle rampe d'étranglement en combinaison avec les gaz résiduels dans les niches de couche-limite dans la région des parois de la chambre de combustion assure un allumage net et direct pour chaque cycle jusqu'aux vitesses maximales d'écoulement.

5.4. Fonctionnements des réacteurs au point fixe et séries de mesures sur deux réacteurs d'essai à des vitesses jusqu'à M = 0.94 (limite des compresseurs du banc d'essai)

Les résultats obtenus et décrits dans les études partielles aboutaient à la fabrication de deux réacteurs d'essai dont la section longitudinale est montrée dans la Planche de fig. 10.

Les dimensions choisies correspondaient aux dimensions favorables en ce qui concerne le nombre de Mach intérieur, la vitesse de flamme et la longueur nécessaire du mélange air-combustible, étant donné que ces grandeurs ne peuvent pas être converties à cause des phénomènes d'écoulement instationnaires. JP4, en qualité de carburant liquide, se prêtait pour cette raison à l'usage tandis que pour des réacteurs aux dimensions inférieures il faut se servir du gaz d'éclairage ou gaz propane dont les caractéristiques d'allumage sont meilleures. La longueur hors tout et la fréquence des réacteurs pourraient être conservées pour des ensembles réacteurs de dimensions plus grandes. Une augmentation des performances pourrait être obtenue par l'accroissement des diamètres.

Vis-à-vis des réacteurs d'essai précédents, le modèle montré en haut de la Planche de fig. 10 présentait un accroissement considérable de la section derrière la chambre de combustion, le cône sur la partie arrière de la chambre de combustion ayant été raccourci à un minimum. Cette conception avait pour but d'obtenir, vis-à-vis de la combustion atteinte jusqu'alors, une pénétration plus profonde de l'air frais dans le tuyau, de manière que, pendant l'allumage, une partie du mélange frais admis circulait déjà immédiatement dans le tourbillon derrière les gaz échappant du cycle précédent.

Le douxième dispositif d'essai se distinguait par un accroissement de la partie centrale du réacteur et avait pour résultat un modèle qui avait déjà été essayé par l'auteur au point fixe à Córdoba (5). Pour le démarrage des réacteurs d'assai l'on se servit de l'air comprimé et des bougies d'allumage montées dans une distance d'environ 200 mm de la paroi avant de la chambre de combustion. Les essais fournissaient des valeurs qui peuvent être réproduites. La force différentielle entre le réacteur à froid et le réacteur en fonctionnement était mesurée. Dans les différentes séries de mesures en fonction du nombre de Mach d'écoulement les valeurs mesurées étaient le comportement de la poussée, la consommation de carburant, ainsi que les températures statiques moyennes sur toute la longueur du réacteur. Les réacteurs n'offraient pas de problèmes au point de vue de fonctionnement. La fréquence mesurée au point fixe était de 83 1/s. Surtout le réacteur d'essai à chambre de combustion unique présentait un fonctionnement sensiblement stabilisé, en fonction d'écoulement. Le fonctionnement du deuxième réacteur d'essai était moins stabilisé, la poussée maximale était pourtant de 15% plus élevée, et les valeurs de consommation étaient très favorables. Dans la Planche de fig. 11 les poussées brut corrigées de même que les valeurs de consommation spécifique du combustible, par rapport aux valeurs corrigées, pour des configurations du cône d'un diamètre de 100 mm et de 140 mm, ont été relevées sur le nombre de Mach d'écoulement. La correction des valeurs de poussée brut fut réalisée en tenant compte de la résistance interne. Dans une recherche partielle séparée, les distributions de pression dans le tuyau eurent été mesurées en fonction de la vitesse d'écoulement, sur deux sections et à 4 points de mesure supplémentaires.

Pour calculer des vitesses, les valeurs mesurées furent pris comme valeurs de base. En se servant de la loi d'énergie de Bernouilli pour des courants compressibles, à savoir:

$$\frac{\varkappa}{\varkappa-1}\cdot\frac{P}{\varrho}+\frac{W^2}{2}=\text{const.} \qquad (1)$$

on obtient pour le cas en question l'équation suivante:

$$\frac{\varkappa}{\varkappa-1}\cdot\frac{P}{\varrho}+\frac{W^2}{2}=\frac{\varkappa}{\varkappa-1}\cdot\frac{P^0}{\varrho^0} \qquad (2)$$

En outre

$$\frac{\rho^o}{\rho} = \left(\frac{P^o}{P}\right)^{\frac{1}{\varkappa}} \tag{3}$$

Lorsque l'équation (3) est substituée dans l'equation (2), W^2 est formé comme suit:

$$W^2 = \frac{2\varkappa \cdot P}{(\varkappa-1)\cdot \rho}\left[\left(\frac{P^o}{P}\right)^{\frac{\varkappa-1}{\varkappa}} - 1\right] \tag{4}$$

En outre,

$$W_o^2 = \frac{\varkappa \cdot P}{\rho}$$

et

$$\frac{P^o}{P} = \frac{P^o - P}{P} + 1 = \frac{q}{P} + 1 \quad ,$$

de manière que l'équation s'exprime maintenant comme suit:

$$M^2 = \frac{2}{\varkappa-1}\left[\left(\frac{P^o-P}{P}+1\right)^{\frac{\varkappa-1}{\varkappa}} - 1\right] \tag{5}$$

Etant donné que les grandeurs P^o et P furent mesurées pendant l'essai, M et W peuvent être déterminés par respectivement les equations (5) et (6)

$$W^2 = \frac{2W_o^2}{\varkappa-1}\left[\left(\frac{P^o-P}{P}+1\right)^{\frac{\varkappa-1}{\varkappa}} - 1\right] \tag{6}$$

Par le calcul du débit d'air dans l'entrée pendant des écoulements continus et par une poursuite du calcul sur la base d'une répartition en 50% pour des écoulements pulsatoires dans un premier calcul approximatif, la courbe de correction qui est requis pour déterminer les poussées brut fut relevée tenant compte des vitesses différentielles respectives (Planche de fig. 12).

Afin de permettre une comparaison, les valeurs des consommations spécifiques du réacteur As-014 out été relevées dans le diagramme à côté des valeurs obtenues. Ces valeurs furent enregistrées en son temps dans la soufflerie. Ce diagramme démontrait qu'il y avait un déplacement des valeurs obtenues dans une plage plus élevée de Mach. Un autre relevé montrait les densités des poussées ($\frac{kp}{m^2}$) pour des poussées brut, toujours en fonction des diamètres de la chambre de combustion des réacteurs d'essai. Ce relevé démontrait que, grâce au nouveau développement, il était possible de surmonter la vitesse-limite jusqu'à M = 0.94. De cette façon, une densité maximale de la poussée pourrait être obtenue avec des valeurs d'ênviron 1900 kp/m² et des consommations spécifiques à l'ordre de 2 kg/kph.

En supposant qu'avec des configurations favorables de l'entrée et de la nacelle du réacteur au point de vue aérodynamique, des résistances pourraient être obtenues qui correspondraient à un coefficient de résistance de 0,2 environ, la portion restante de la densité de poussée d'environ 1000 kp/m² suffirait d'abord pour la propulsion d'un missile. Il est à supposer que le coefficient de résistance du réacteur AS-014 mesuré en son temps de l'ordre de 0,4 - suivant Staab (2) - n'est pas compatible avec les vitesses d'écoulement de M = 0.9 employées à présent.

La configuration optimale d'un capot d'entrée d'air et du contour de nacelle ne fit aucune partie du programme en cours.

A cet égard, il est intéressant de signaler que Gosslau (1) eut en son temps, de très grandes diffucultés en projetant l'aérodynamique du capot d'entrée d'air et de la nacelle du réacteur As-014 (1, p. 415-418, discussion Gosslau, Blenk, E. Schmidt).

Les résultats indiqués plus haut, ont démontré qu'en principe la conception de base qui n'eut pas pourvu des clapets et qui fut réalisée en son temps en France, pouvait être conservé jusqu'à un nombre de Mach d'écoulement de M = 0,94. A cet égard, le réacteur d'essai 016-132 fut soumis à des essais de fonctionnement avec des écoulements correspondant à M = 0.9, sur le banc d'essai de l'ILA à Stuttgart avec des résultats satisfaisants.

Le réacteur d'essai mis au point fournit la preuve qu'il pourra être pris en considération pour des missions à vitesses de l'ordre de M = 0,85 à M = 0,95, pourvue que l'accélération du missile sera réalisée à l'aide des fusées d'accélération. Au cours des essais, on a trouvé que, sous l'effet de l'écoulement, le réacteur fonctionne avec un surplus d'air, environ jusqu'à λ = 3,5; par ailleurs, un calcul pour le réacteur As-014 a donné le même résultat.

Pendant le fonctionnement au sol, c'est-à-dire sur le banc d'essai, la combustion était quasi-stoechiométrique. En outre, on a constaté qu'à des vitesses d'écoulement élevées, il n'y avait pas de coussin d'air et de gaz résiduels dans la partie arrière de la sortie tandis que, sur le point fixe et à des vitesses d'écoulement moins élevées, le coussin d'air fournit une partie de la poussée à froid.

Pendant des essais sur les réacteurs, on a toujours trouvé qu'à partir de Mach = 0.85 il y a une fréquence d'admission d'air plus élevée, causée par les vitesses croissantes. La conclusion en fut qu'il y a plusieurs allumages qui, en combinaison avec la flamme-pilote, produirent un effet freinant aux masses d'air d'admission, tout en évitant une extinction du réacteur.

C'était aussi la confirmation que la rampe était bien dimensionée en ce qui concerne sa position, son angle de cône (demi-angle), sa longueur et sa section de passage. Un chargement des conditions d'essai nécessite l'ajustage de la géométrie de l'entrée, ci-inclus des dimensions de la rampe. Une expérience analogue se fit déjà, en son temps, par un groupe d'experts dans le domaine du réacteur AS-014 (1, pages 415-418).

Le programme d'expérimentation comprit une période de 5 ans qui était interrompue par des intervalles différentes.

Un autre diagramme de la Planche de fig. 12 démontre, pour des fonctionnements du réacteur à M = 0.88, les effets provoqués par une admission d'air oblique; au cours des essais, quatre cônes de diamètres différents ont été mesurés. La Planche de fig. 13 donne une vue du réacteur d'assai, monté sur son berceau de support; la figure au centre donne une vue éclatée de celui-ci. En outre, cette planche montre un fonctionnement du réacteur sous l'écoulement, sur le terrain d'essai de la DFVLR à Trauen (Luftzentrale - Centrale d'Air).

6. ESSAIS REALISES SUR UN PULSO-REACTEUR AVEC PASSAGE AU FONCTIONNEMENT EN STATO-REACTEUR

La réalisation d'un passage du fonctionnement intermittent en fonctionnement continu fut obtenue par les essais de réacteur, afin de déterminer les effets de stabilisation des flammes pendant les deux modes de combustion. Là il s'agissait d'essais préliminaires.

Le passage se présenta à un nombre de Mach d'environ M = 0.7, avec une température moyenne dans la chambre de combustion de 1300°C (la courbe supérieure représente les températures mesurées immédiatement après l'accroche-flamme).

Après le passage du mode pulsatoire au fonctionnement en stato-réacteur, le processus d'une combustion continu put être observé même sous l'effet d'une variation graduelle de la quantité du carburant injecté. Le passage d'un mode de combustion à l'autre put être observé sur le réacteur non-caréné par le déplacement de l'aspect des couleurs provoquées par les températures (chaleur au rouge) dans le sens de la partie arrière de la chambre de combustion. Le passage ayant été accompli, c'est à ce point là où se stabilisa la flamme. Par ailleurs, le passage de combustion put aussi être observé par l'effet acoustique et le changement de la poussée était visible sur l'oscilloscope au moment du passage. Le montage d'essai est montré en Planche de fig. 14. Les valeurs mesurées, telles que les températures moyennes, comportement de poussée et pression d'injection de carburant, out été relevées sur la vitesse d'écoulement. Pour des détails, voir le diagramme gauche de cette planche.

L'expérience gagnée par ces essais démontre qu'il serait avantageux, du point de vue d'un rendement plus grand de poussée, de réaliser le passage à des vitesses d'écoulement plus élevées. Puisque le réacteur d'essai n'est qu'un prototype de définition, d'autres essais des composants devraient être réalisés en connexion avec un réacteur à deux phases de combustion pour pouvoir arriver à des conclusions définitives. En fin de compte, les résultats de ces essais préliminaires à deux phases de combustion ont démontrés que le passage d'un mode de combustion à l'autre est réalisable avec les mêmes composants du réacteur. Pour donner une idée claire du processus, cette planche présente un schéma du passage d'une phase à l'autre.

Comme suite à ces essais, la mise au point d'un accroche-flamme à section variable et à déplacement axial sous forme d'un dispositif polygonal constitua la solution du problème (voir Planche de fig. 16). Il était clair que, pour des modèles futurs de ces réacteurs, l'accroche-flamme devra avoir une section variable à la partie arrière de la chambre de combustion afin de forcer le passage par une pression dynamique accrue. Les conditions survenues au cours des fonctionnements du réacteur furent examinées à l'aide de calculs.

Les résultats de ces calculs relevés en Planche de fig. 15 permettent de voir, à l'aide de courbes, les gammes accomplis. Il a été relevé le rapport critique des surfaces en fonction des températures d'échappement de la chambre de combustion pour de différents nombres de Mach pendant une combustion stoechiométrique. Pour les trois sections critiques examinées et les valeurs de température T_2, correspondant aux diamètres de 70 à 90 mm, mesurées au cours des essais, le comportement du nombre de Mach M_2 fut constaté dans la chambre de combustion. Par le relevé du rapport du chauffage sur M_2, il était possible de déterminer les nombres de Mach d'entrée M_1. Ces valeurs rangeaient entre 35 et 50 m/sec. Il s'en suit que l'air froid s'écoulant dans la chambre de combustion devait avoir cette vitesse. Ce résultat était tout d'abord suffisant pour les essais sur un prototype de définition.

7. EMPLOI DES PULSO-REACTEURS

En raison de leur construction simple et de leur coût de revient réduit, les pulso-réacteurs peuvent être pris en considération comme des propulseurs (de bon marché) pour des missiles guidés. Ce sont, en particulier, les frais de fabrication bas de même que les frais d'exploitation modérés qui justifient leur emploi pour des vitesses de vol jusqu'au M = 0.9; pour cet emploi, le rayon d'action est limité car la consommation se trouve au-dessus des valeurs de consommation des turboréacteurs. De point de vue d'entretien, d'usure et de la déformation, ces réacteurs n'offrent que d'avantages. En outre, leur puissance massique est favorable; la plupart des composants du réacteur peut être réalisée en tôles d'une épaisseur de 0,5 jusqu'à 0,7 milimètre. Le programme d'expérimentation a démontré qu'il faut des "boosters" (fusées d'accélération) pour le lancement des engins munis de tels réacteurs. Quand, pour certain raisons, il faut renoncer à cette accélération, la sortie à grande volume doit être télescopique ou éjectable, comme montre la figure haute de Planche de fig. 17.

En se basant sur le programme d'expérimentation réalisé, on a considéré de s'appuyer, entre autres choses, sur les résultats de mesures faites pour le projet de turboréacteurs. Pour cela, il semble d'abord important de projeter une soupape pour une admission d'air approprié. Le but en est de permettre une alimentation en air intermittente qui est suffisante pour des chambres de combustion courtes. Le modèle de réacteur, répresenté en coup longitudinal dans la figure basse de la planche de fig. 17 est doté d'une soupape rotative à vitesse de rotation variable dans la chambre de combustion. Cette soupape comprend un stator et un rotor qui s'allongent sur toute la chambre de combustion. En rotation, le rotor couvre et affranchit les fentes du stator. En prévoyant un certain nombre de fentes, on cherche, en combustion pulsatoire, à assurer la formation impeccable du mélange pendant la phase d'admission très courte. Comme constaté dans une note précédente (13), il est important de prévoir une chambre de stabilisation entre la chambre de combustion et la turbine. Du côté de compresseur, de l'air additionnel est admis à la chambre de stabilisation afin de provoquer le refroidissement approprié des gaz et leur mélange avec de l'air comprimée et d'affaiblie les oscillations de pression avant l'entrée des gaz dans la turbine (à une ou plusieures étages). De cette manière, la combustion pulsatoire dans la chambre de combustion peut s'effectuer sur des températures très élevées ce qui n'était pas possible auparavant. La proposition envisage un processus de combustion à pulsation commandée; il n'y en a pas encore des résultats d'essais.

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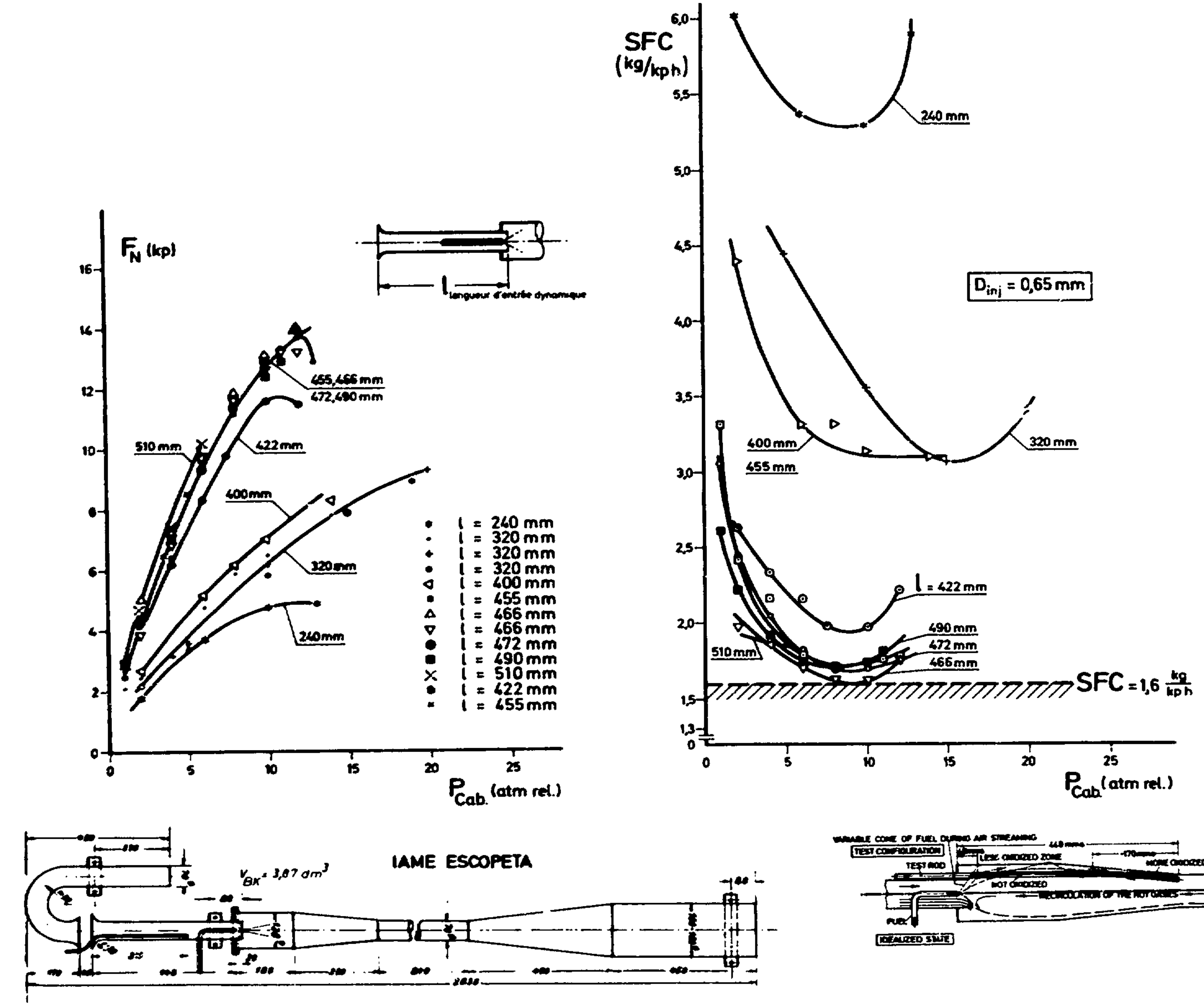


F_N (kp)
l longueur d'entrée dynamique
455,466 mm
472,490 mm
510 mm
422 mm
400 mm
320 mm
240 mm
l = 240 mm
l = 320 mm
l = 320 mm
l = 320 mm
l = 400 mm
l = 455 mm
l = 466 mm
l = 466 mm
l = 472 mm
l = 490 mm
l = 510 mm
l = 422 mm
l = 455 mm
$P_{Cab.}$ (atm rel.)
SFC (kg/kph)
240 mm
D_{inj} = 0,65 mm
400 mm
455 mm
320 mm
l = 422 mm
490 mm
472 mm
466 mm
510 mm
SFC = 1,6 kg/kph
$P_{Cab.}$ (atm rel.)
IAME ESCOPETA
V_{BK} = 3,87 dm^3
VARIABLE CONE OF FUEL DURING AIR STREAMING
TEST CONFIGURATION
TEST ROD
FUEL
IDEALIZED STATE

Figure 1

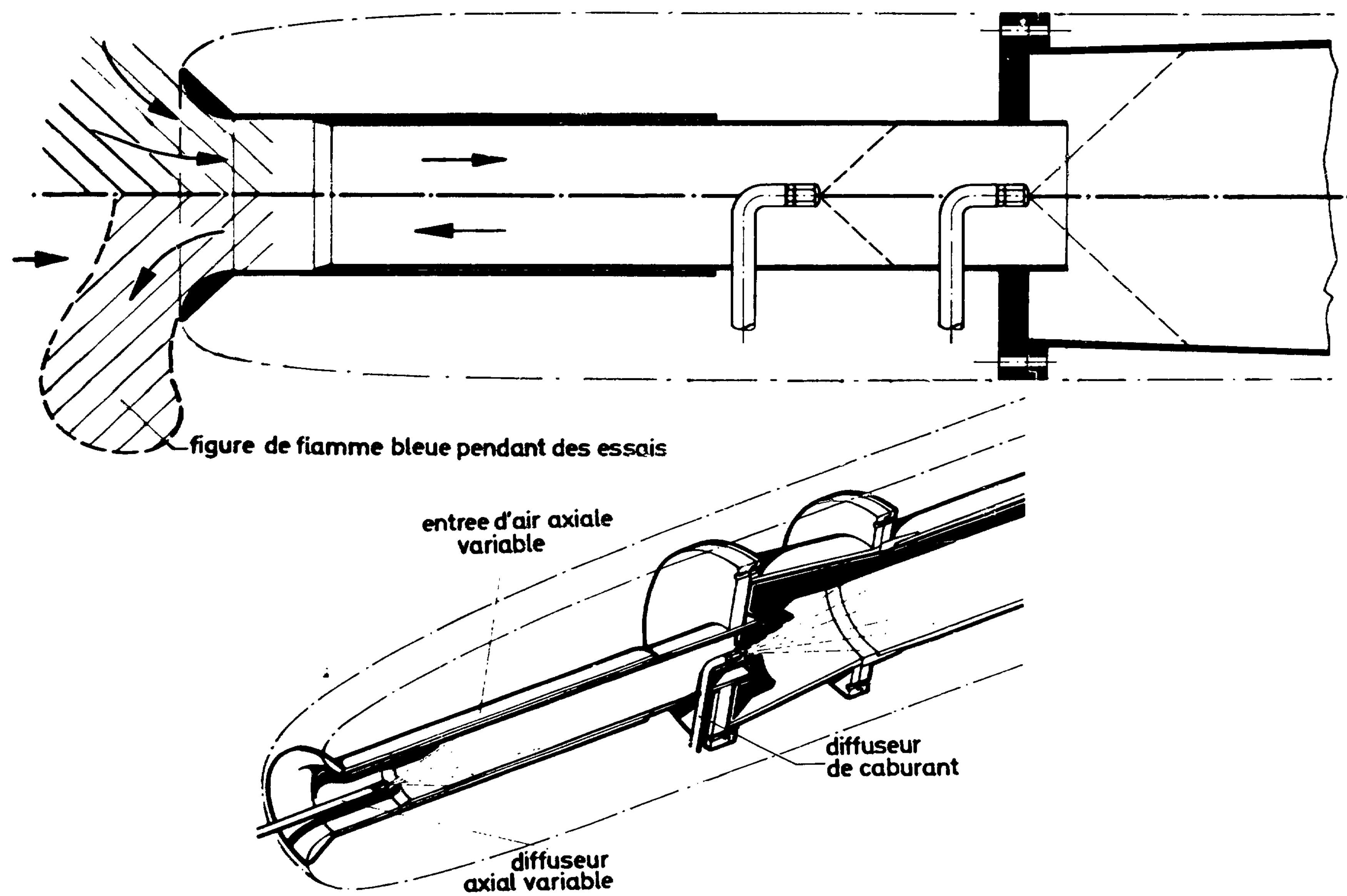
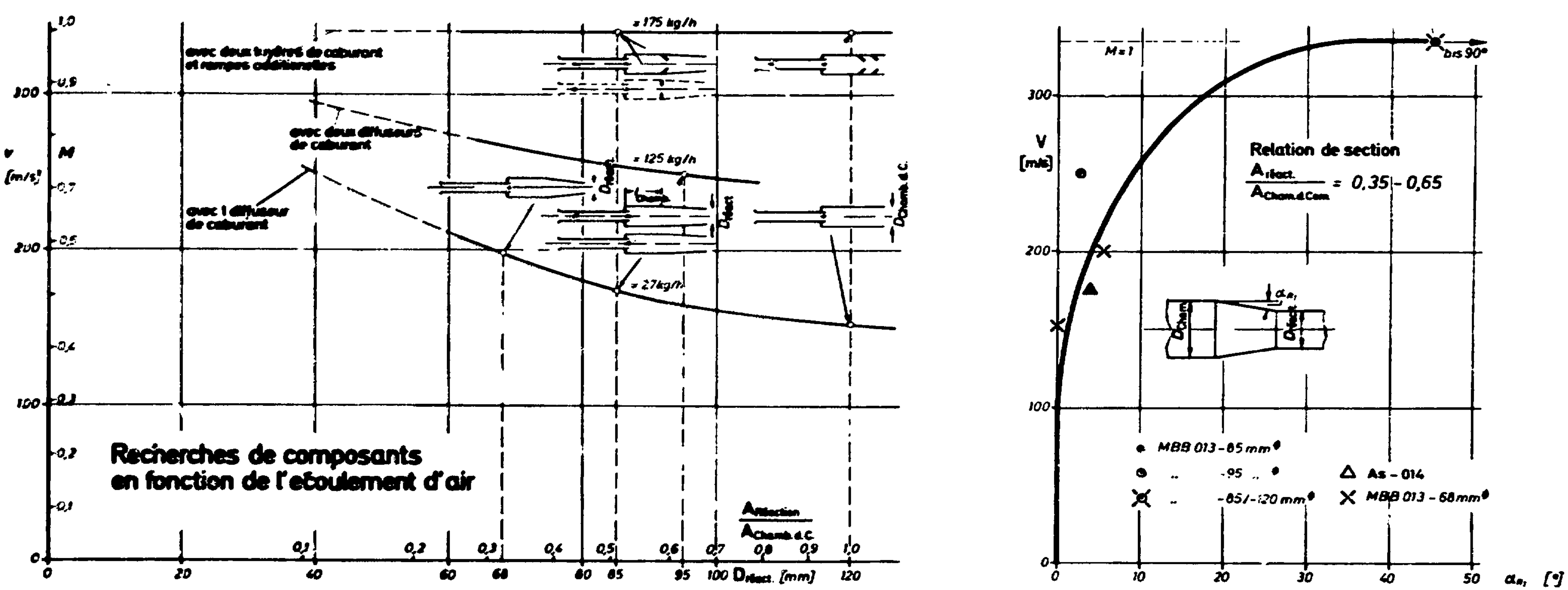


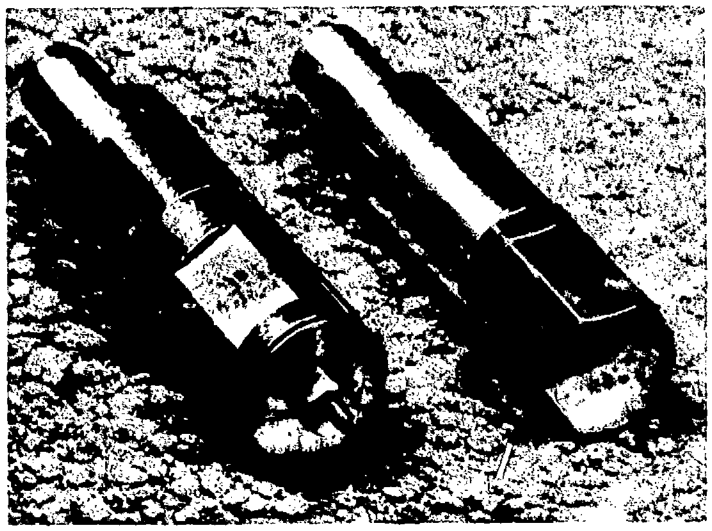
figure de flamme bleue pendant des essais
entree d'air axiale
variable
diffuseur
de caburant
diffuseur
axial variable

Figure 2



Recherches de composants
en fonction de l'écoulement d'air
avec deux tuyères de carburant et rampes additionnelles
avec deux diffuseurs de carburant
avec 1 diffuseur de carburant
≈ 175 kg/h
≈ 125 kg/h
≈ 27 kg/h
v [m/s]
M
D [mm]
Relation de section
= 0,35 - 0,65
M = 1
bis 90°
V [m/s]
MBB 013 - 85 mm
As - 014
MBB 013 - 68 mm

Figure 3



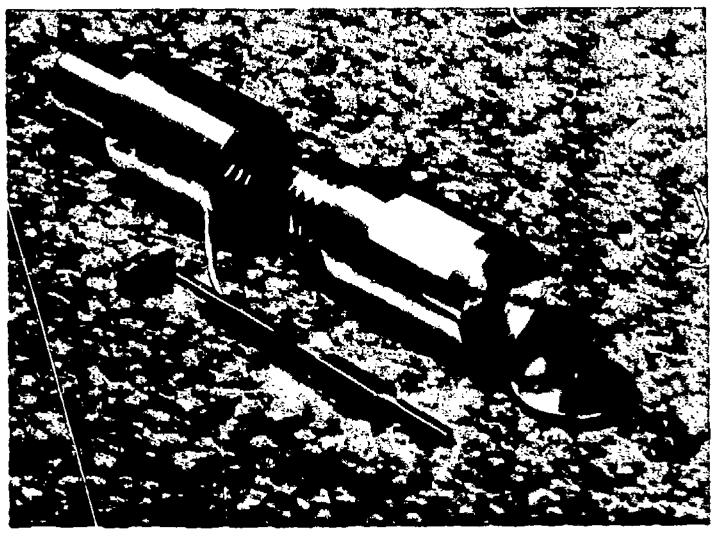


Figure 4

- Projet -

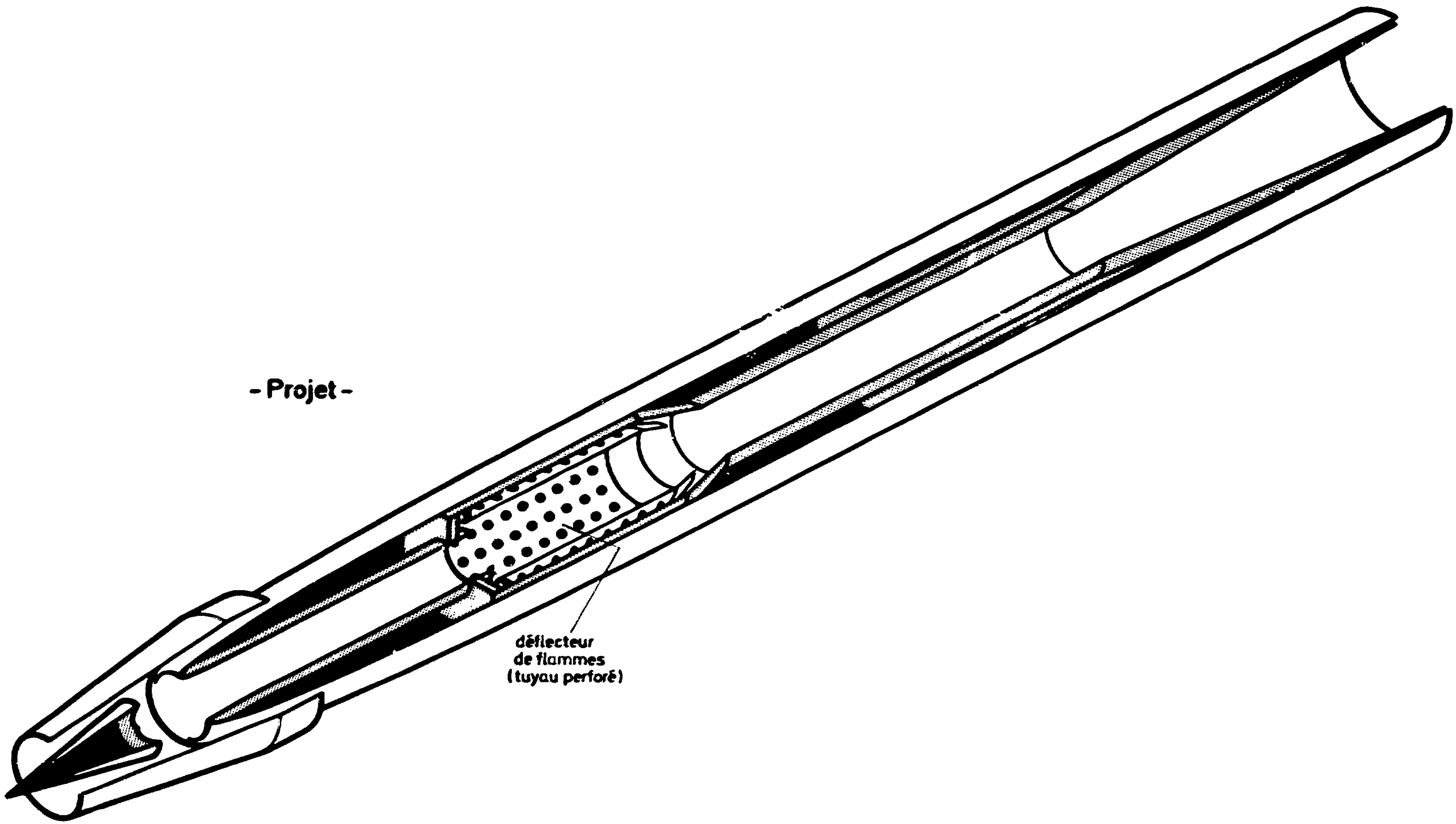
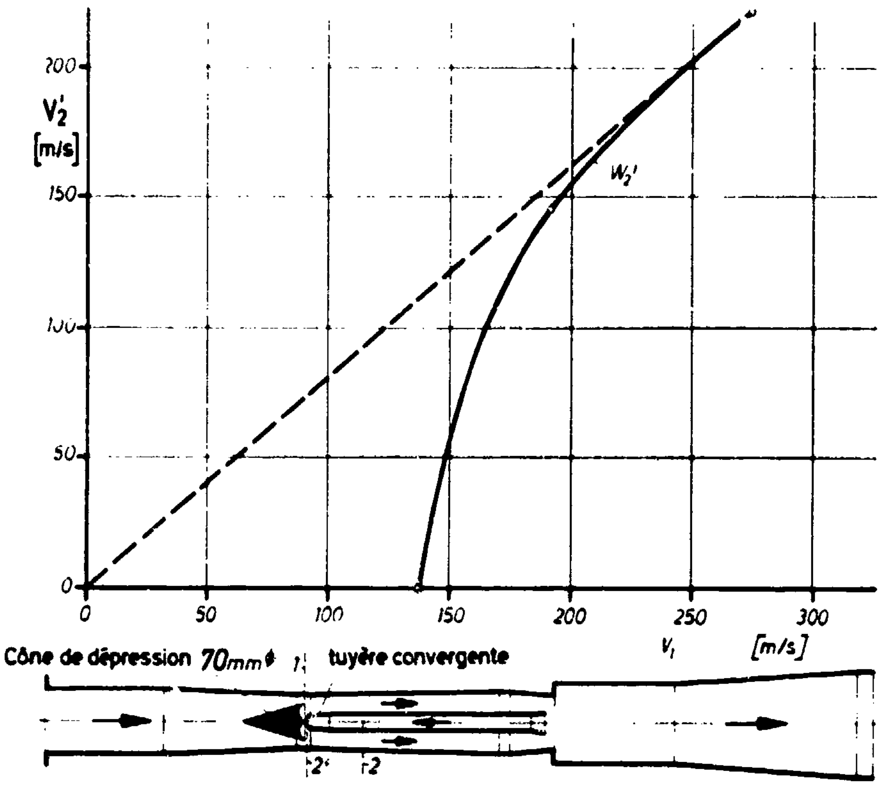


Figure 5


V_2' [m/s]
W_2'
V_1 [m/s]
Cône de dépression 70mm ⌀
tuyère convergente

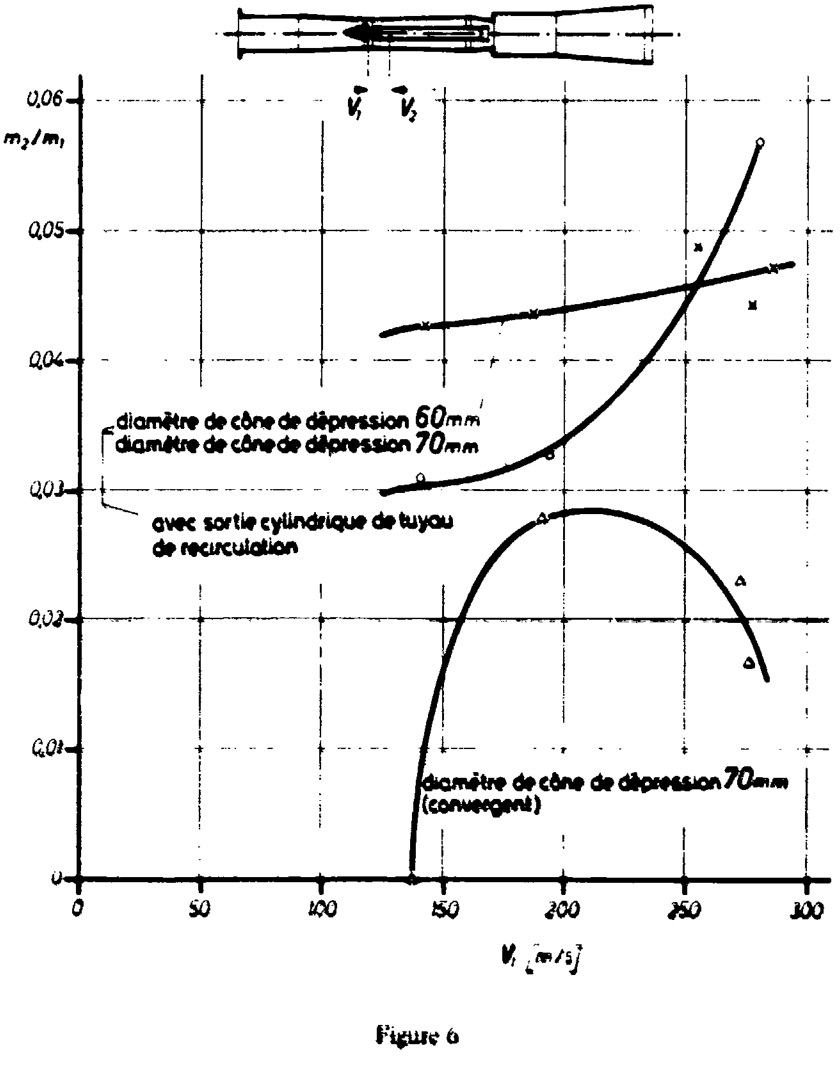
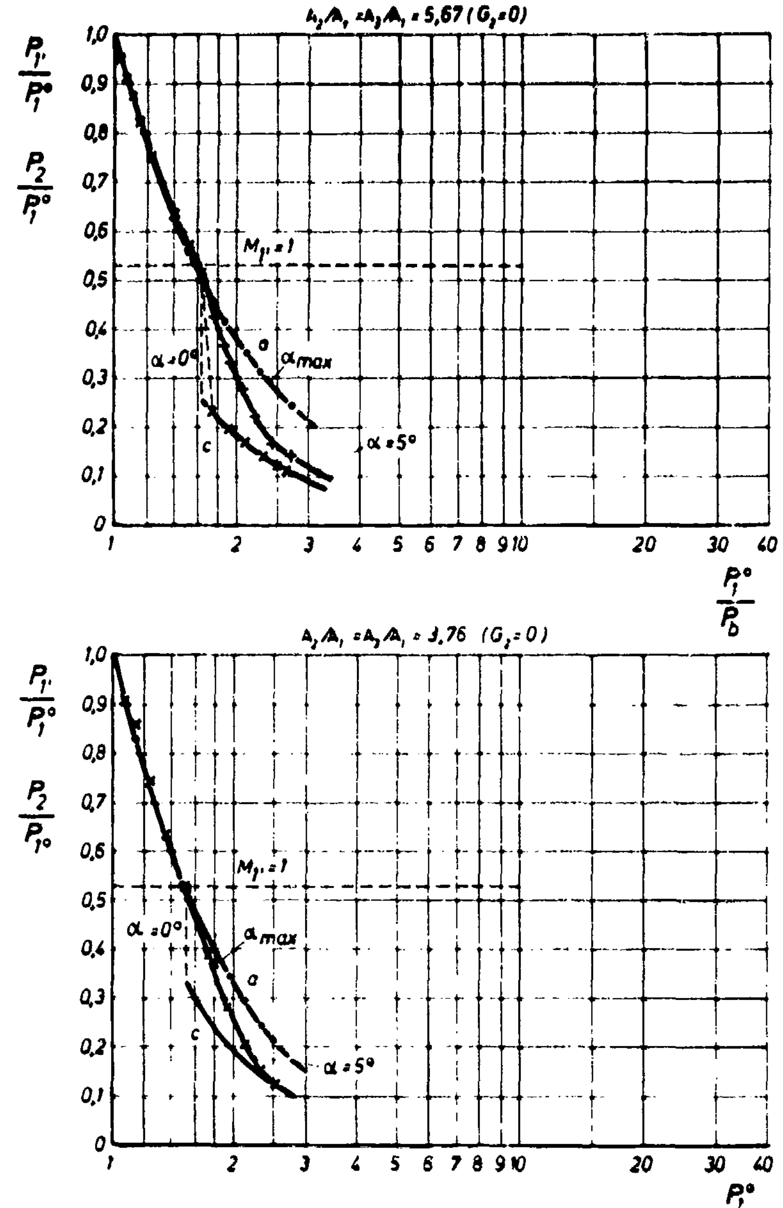

V_1
V_2
m_2/m_1
diamètre de cône de dépression 60mm
diamètre de cône de dépression 70mm
avec sortie cylindrique de tuyau de recirculation
diamètre de cône de dépression 70mm (convergent)
V_1 [m/s]

Figure 6



Elargissement abrupte de tuyau
a – écoulement secondaire
c – " primaire

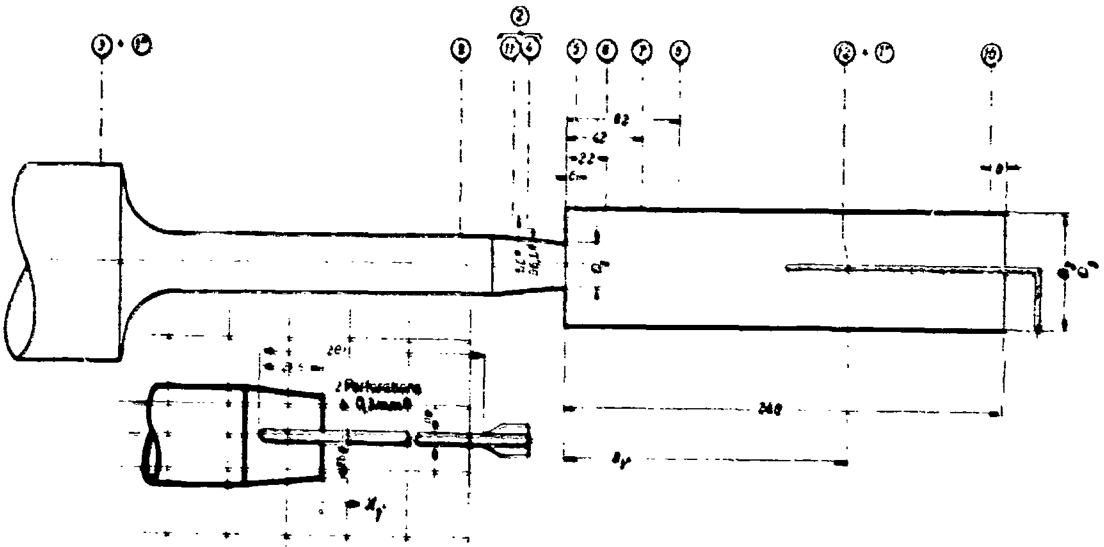


Figure 7

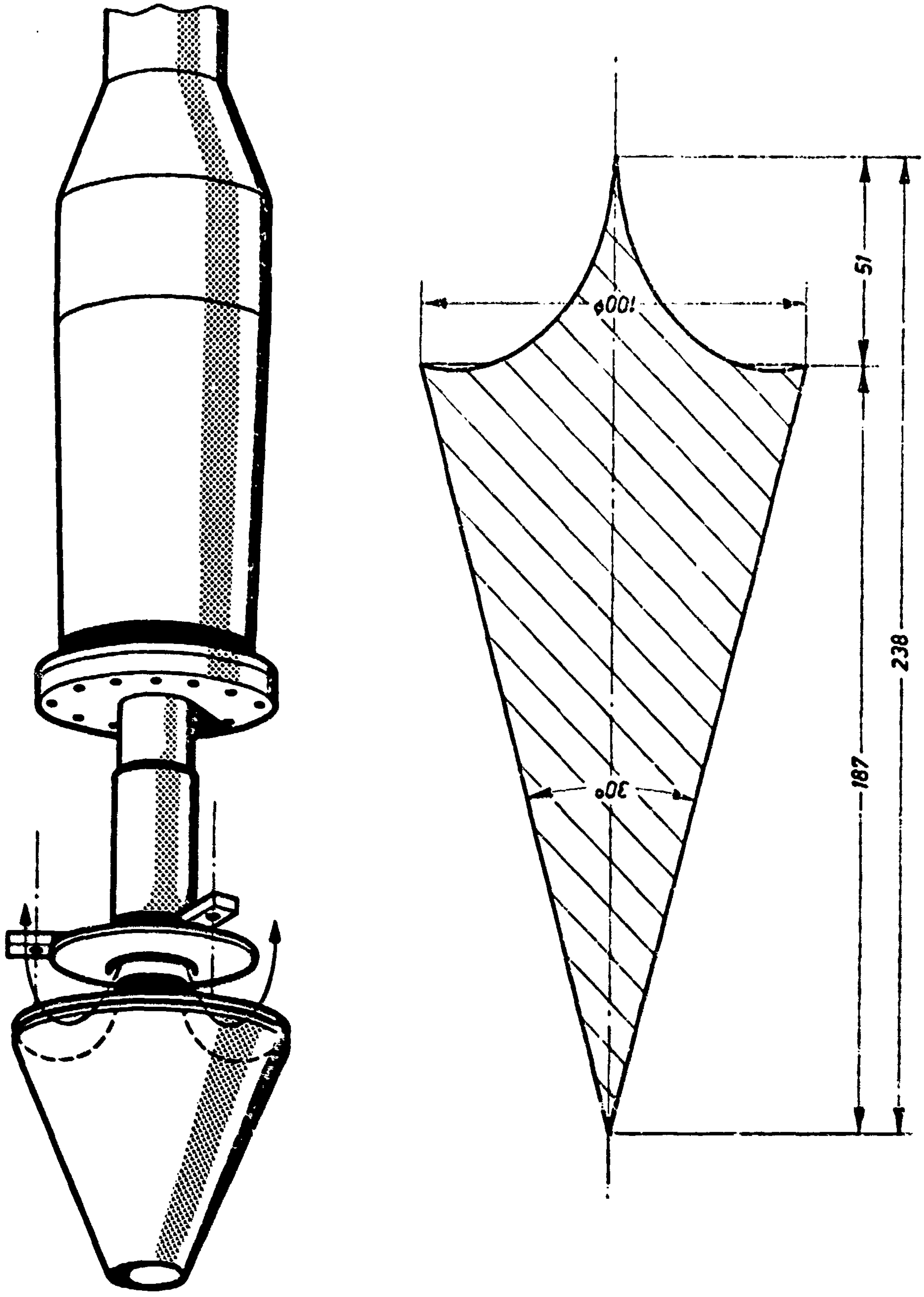


Figure 8

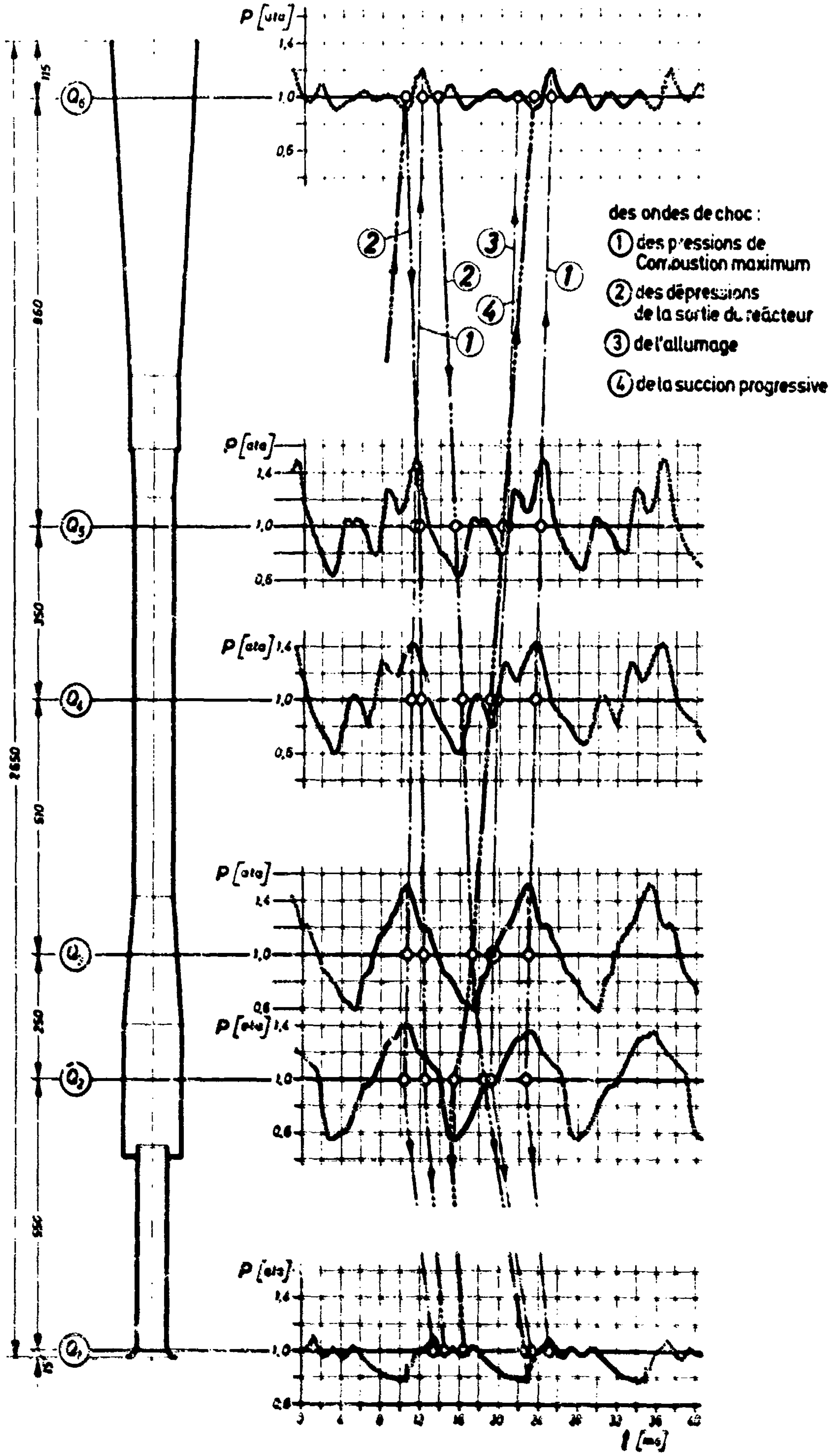

des ondes de choc :
1 des pressions de Combustion maximum
2 des dépressions de la sortie du réacteur
3 de l'allumage
4 de la succion progressive
p [ata]
t [ms]

Figure 9

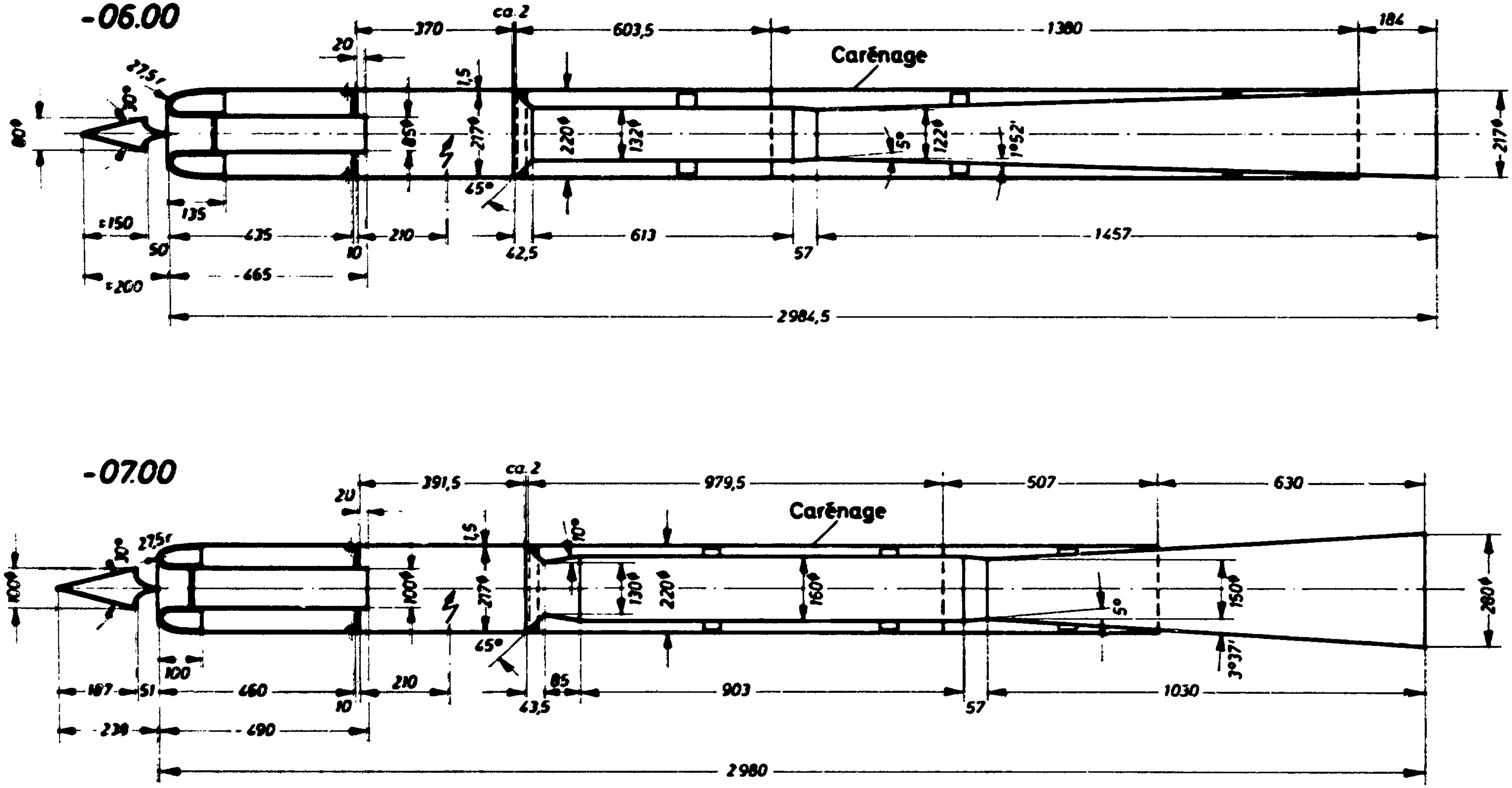
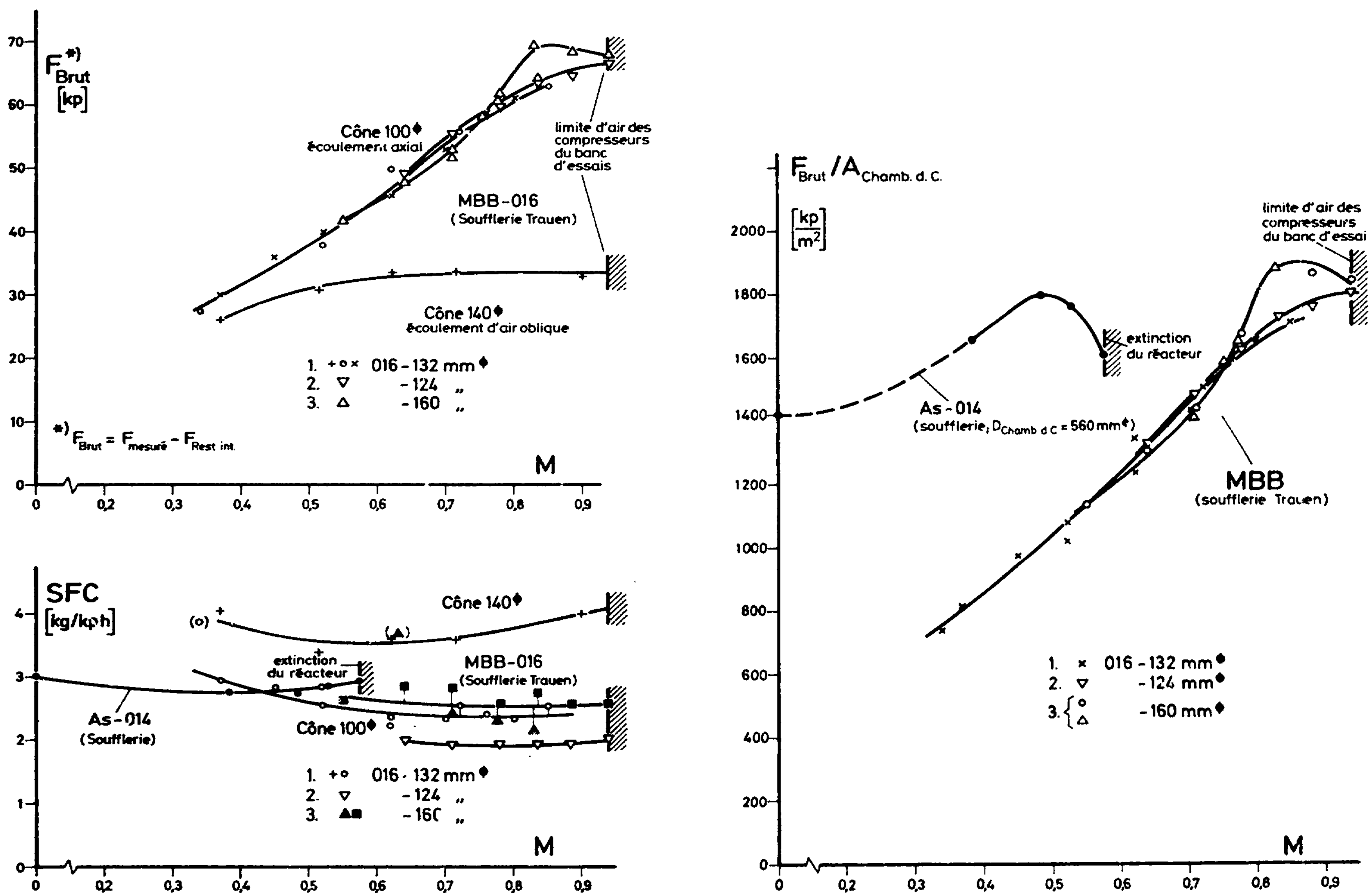
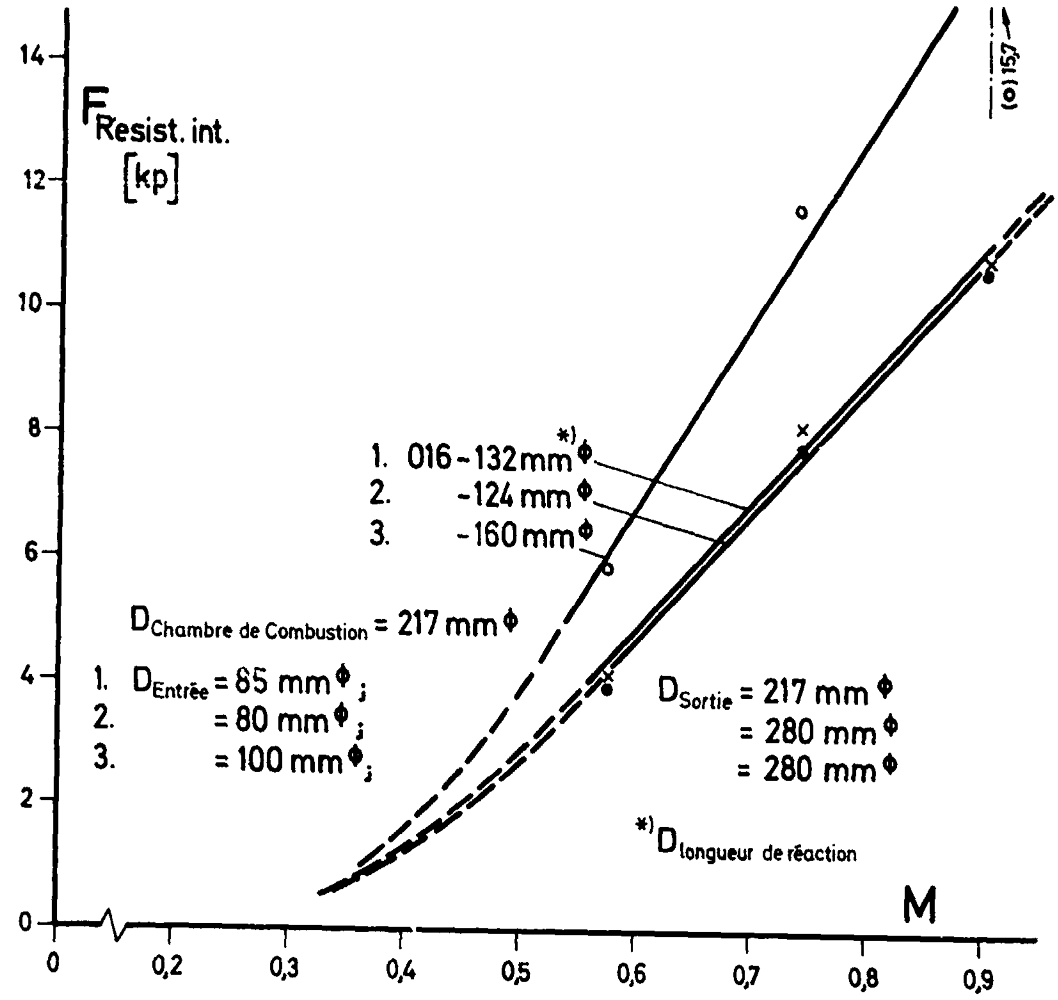

-06.00
Carénage
370
ca. 2
603,5
1380
184
20
27,5 r
30°
80 ϕ
1,5
85 ϕ
217 ϕ
220 ϕ
132 ϕ
5°
122 ϕ
1°52'
217 ϕ
45°
135
≈150
50
435
10
210
42,5
613
57
1457
≈200
465
2984,5
-07.00
Carénage
391,5
ca. 2
979,5
507
630
20
27,5 r
30°
100 ϕ
1,5
10°
100 ϕ
217 ϕ
130 ϕ
220 ϕ
160 ϕ
5°
150 ϕ
280 ϕ
3°37'
45°
100
187
51
460
10
210
85
43,5
903
57
1030
238
490
2980

Figure 10

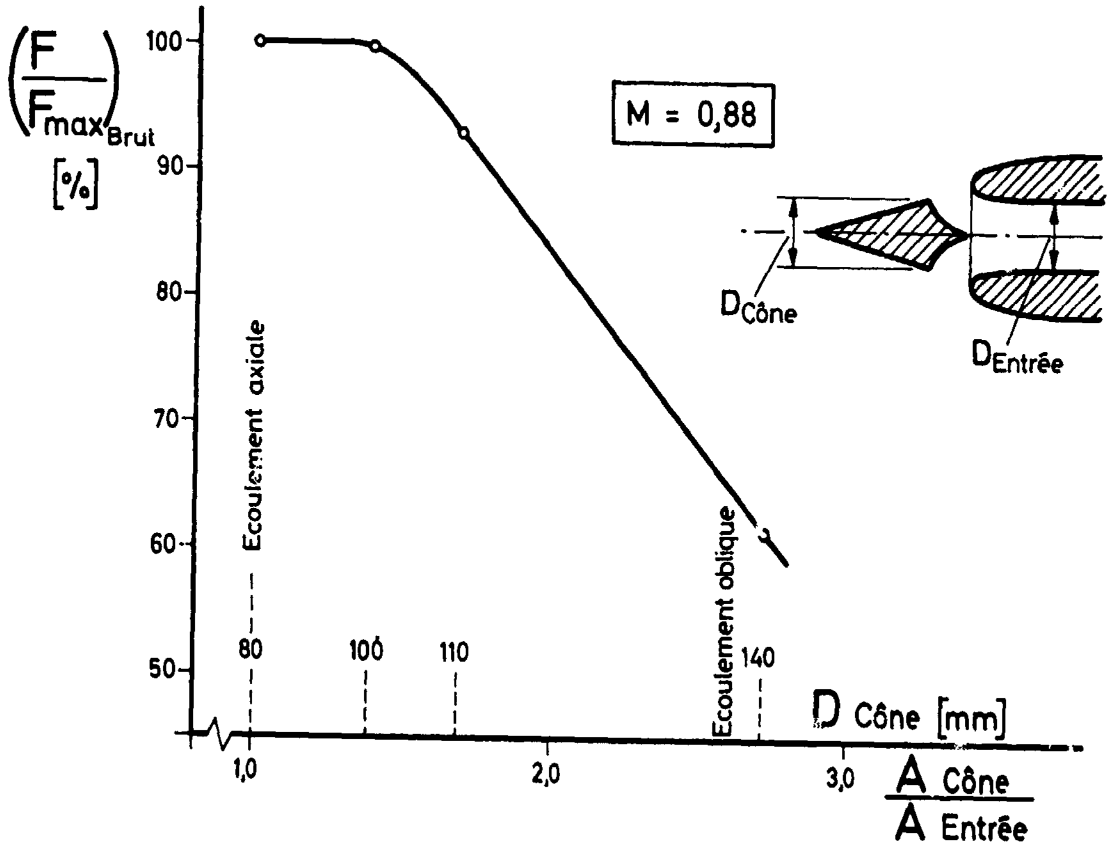


$F_{Brut}^{*)}$ [kp]
Cône 100⌀
écoulement axial
limite d'air des compresseurs du banc d'essais
MBB-016
(Soufflerie Trauen)
Cône 140⌀
écoulement d'air oblique
1. +○× 016-132 mm⌀
2. ▽ -124 „
3. △ -160 „
*) $F_{Brut} = F_{mesuré} - F_{Rest\ int.}$
M
SFC [kg/kph]
Cône 140⌀
extinction du réacteur
MBB-016
(Soufflerie Trauen)
As-014
(Soufflerie)
Cône 100⌀
1. +○ 016-132 mm⌀
2. ▽ -124 „
3. ▲■ -160 „
M
$F_{Brut}/A_{Chamb.\ d.\ C.}$ [kp/m²]
limite d'air des compresseurs du banc d'essai
extinction du réacteur
As-014
(soufflerie; $D_{Chamb\ d\ C}$ = 560 mm⌀)
MBB
(soufflerie Trauen)
1. × 016-132 mm⌀
2. ▽ -124 mm⌀
3. ○ △ -160 mm⌀
M

Figure 11

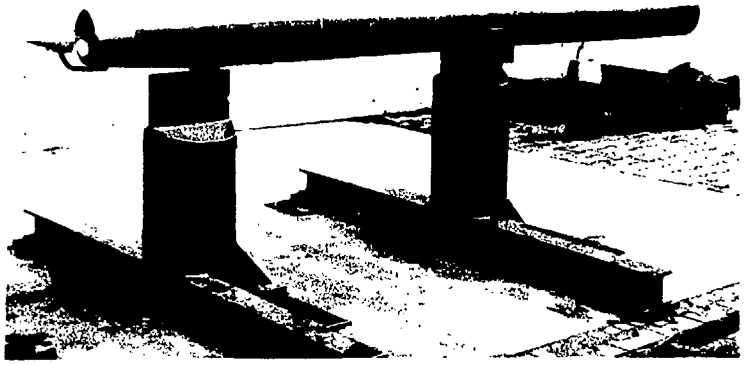


F Resist. int.
[kp]
14
12
10
8
6
4
2
0
1. 016-132 mm Φ *)
2. -124 mm Φ
3. -160 mm Φ
D Chambre de Combustion = 217 mm Φ
1. D Entrée = 85 mm Φ
2. = 80 mm Φ
3. = 100 mm Φ
D Sortie = 217 mm Φ
= 280 mm Φ
= 280 mm Φ
*) D longueur de réaction
M
0
0,2
0,3
0,4
0,5
0,6
0,7
0,8
0,9
(o) 157



(F / F max) Brut
[%]
100
90
80
70
60
50
M = 0,88
D Cône
D Entrée
Ecoulement axiale
Ecoulement oblique
80
100
110
140
D Cône [mm]
1,0
2,0
3,0
A Cône / A Entrée

Figure 12



Vue eclatée

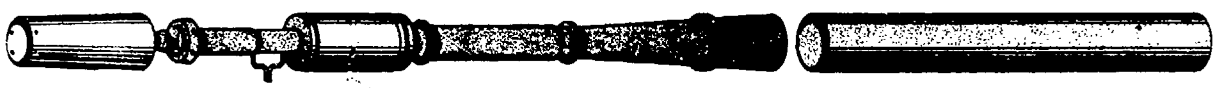
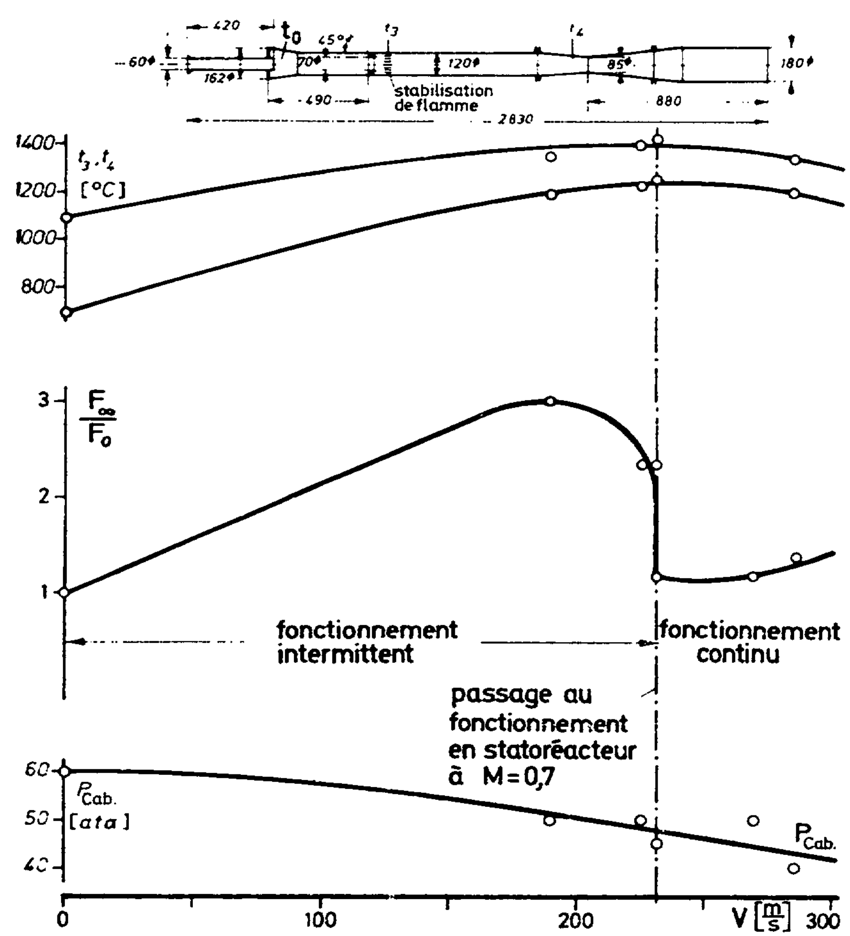
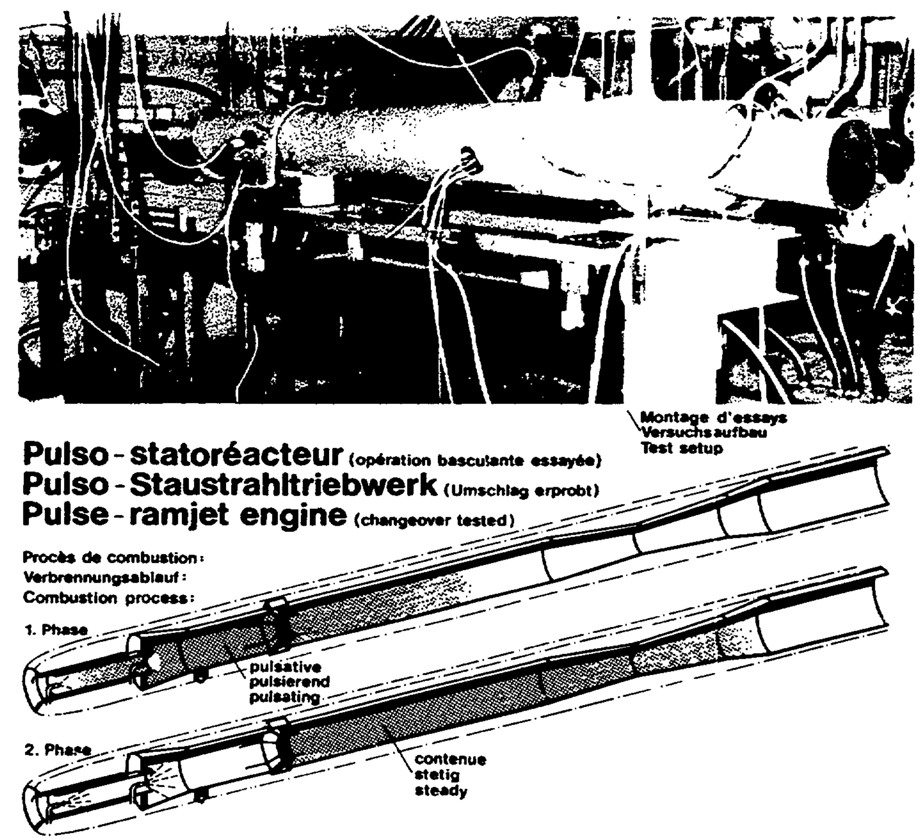


Figure 13

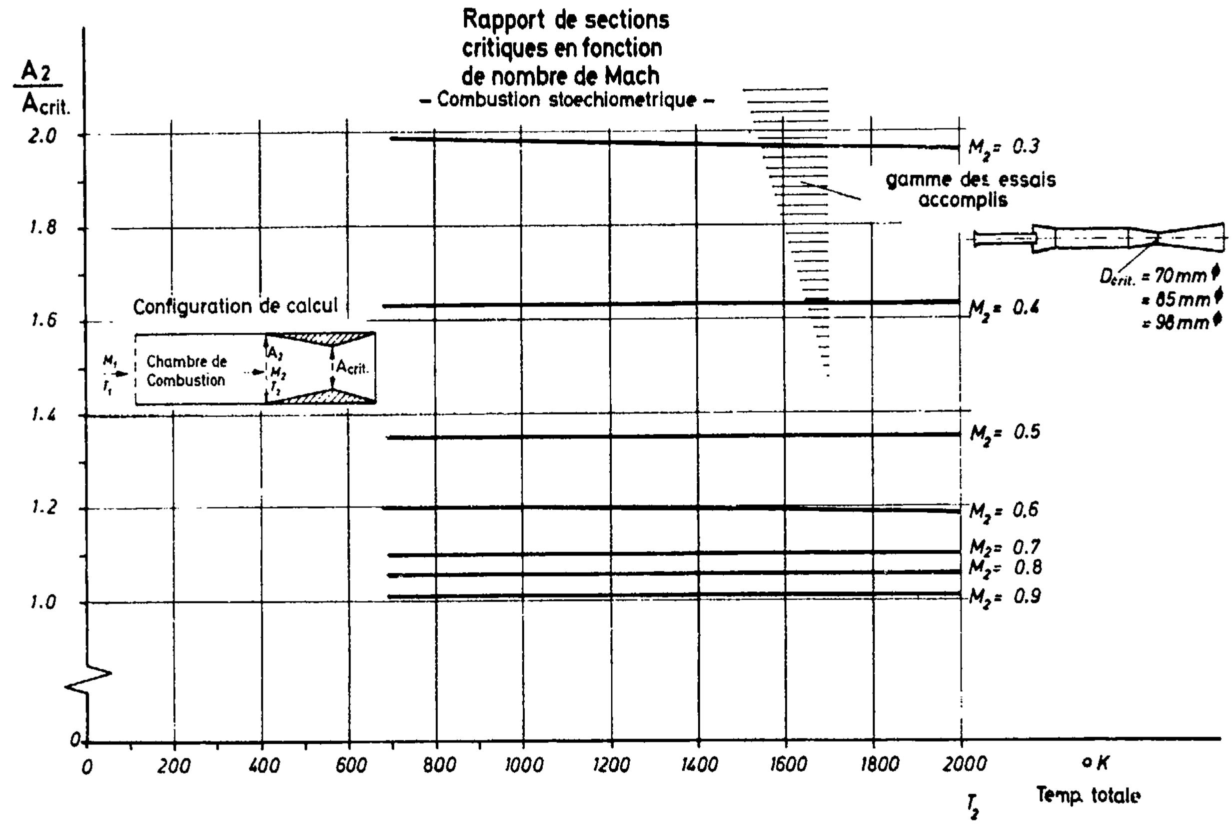


420
t_0
45°
t_3
t_4
60ø
162ø
70ø
120ø
85ø
180ø
490
stabilisation de flamme
880
2830
t_3, t_4 [°C]
1400
1200
1000
800
$\frac{F_\infty}{F_o}$
3
2
1
fonctionnement intermittent
fonctionnement continu
passage au fonctionnement en statoréacteur à M=0,7
$P_{Cab.}$ [ata]
60
50
40
$P_{Cab.}$
0
100
200
V [m/s]
300

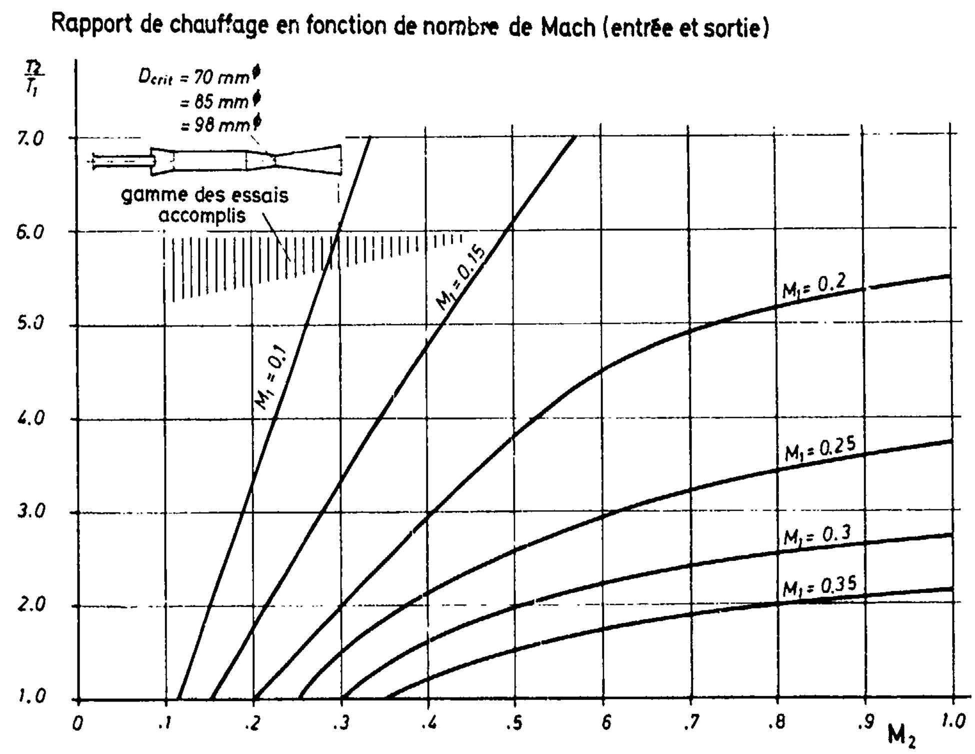


Montage d'essays
Versuchsaufbau
Test setup
Pulso - statoréacteur (opération basculante essayée)
Pulso - Staustrahltriebwerk (Umschlag erprobt)
Pulse - ramjet engine (changeover tested)
Procès de combustion:
Verbrennungsablauf:
Combustion process:
1. Phase
pulsative
pulsierend
pulsating
2. Phase
contenue
stetig
steady

Figure 14



Rapport de sections
critiques en fonction
de nombre de Mach
– Combustion stoechiometrique –
A2 / Acrit.
2.0
1.8
1.6
1.4
1.2
1.0
0
M2 = 0.3
gamme des essais
accomplis
Dcrit. = 70 mm
= 85 mm
= 98 mm
Configuration de calcul
Chambre de
Combustion
Acrit.
M2 = 0.4
M2 = 0.5
M2 = 0.6
M2 = 0.7
M2 = 0.8
M2 = 0.9
0
200
400
600
800
1000
1200
1400
1600
1800
2000
°K
T2
Temp. totale



Rapport de chauffage en fonction de nombre de Mach (entrée et sortie)
T2/T1
Dcrit = 70 mm
= 85 mm
= 98 mm
gamme des essais
accomplis
7.0
6.0
5.0
4.0
3.0
2.0
1.0
M1 = 0.1
M1 = 0.15
M1 = 0.2
M1 = 0.25
M1 = 0.3
M1 = 0.35
0
.1
.2
.3
.4
.5
6
.7
.8
.9
1.0
M2

Figure 15

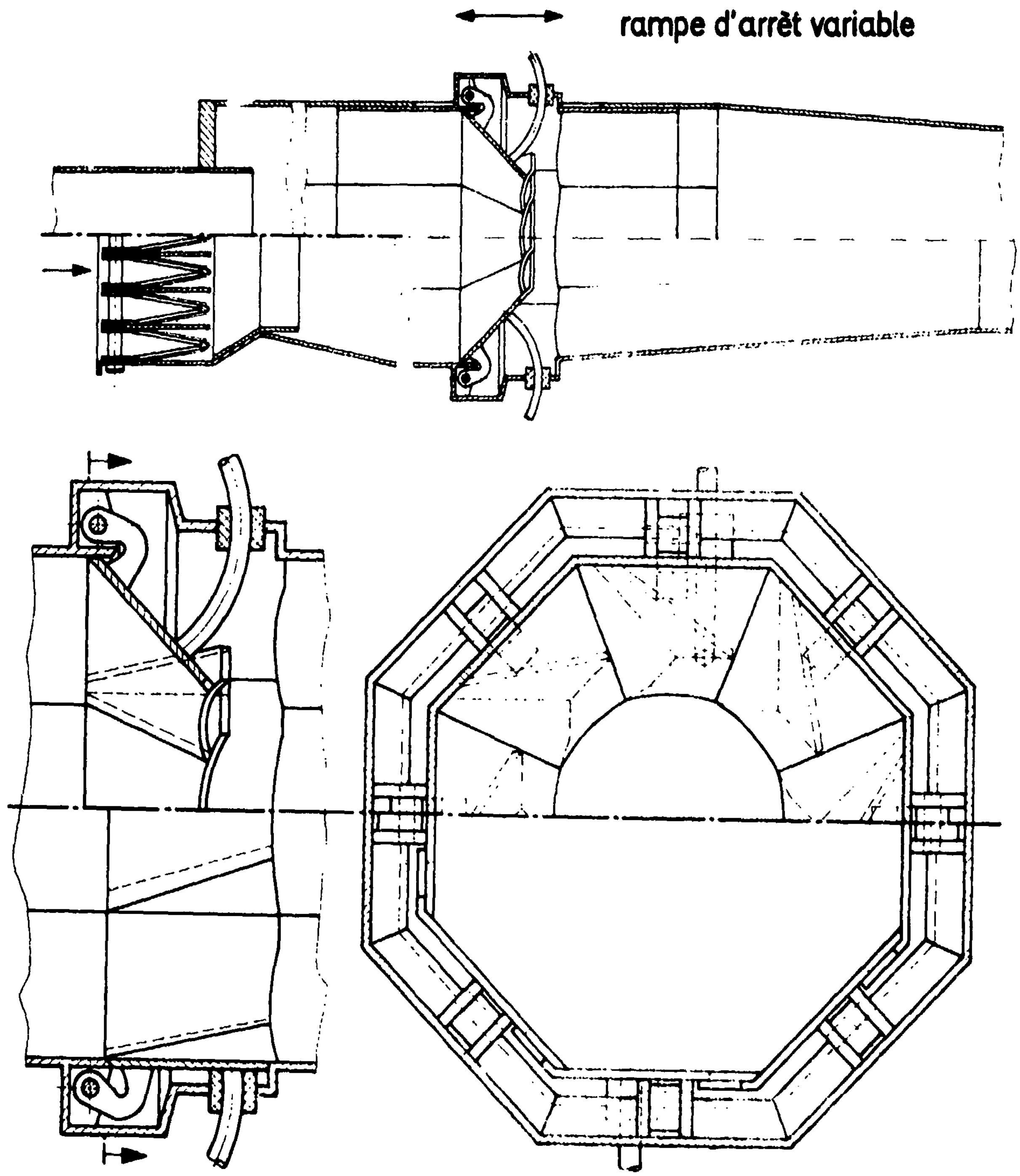
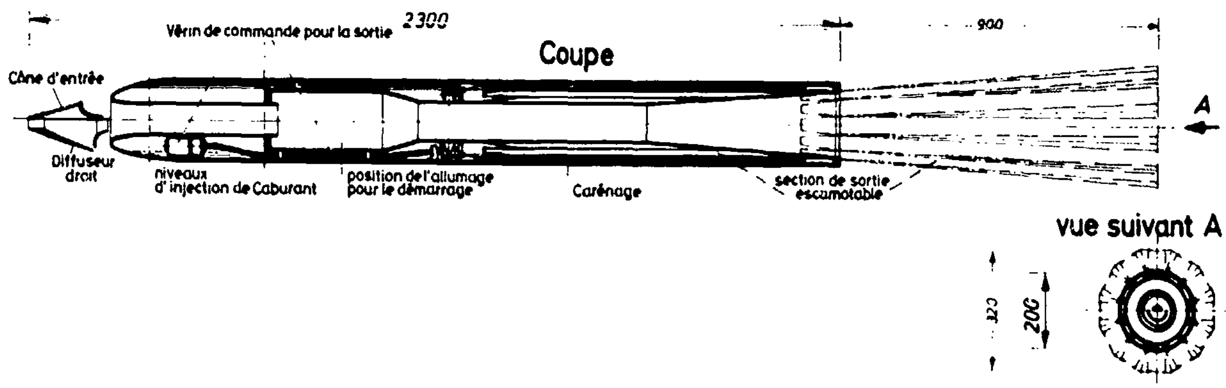


Figure 16


2300
Vérin de commande pour la sortie
Coupe
900
Cône d'entrée
Diffuseur droit
niveaux d'injection de Caburant
position de l'allumage pour le démarrage
Carénage
section de sortie escamotable
A
vue suivant A
320
200

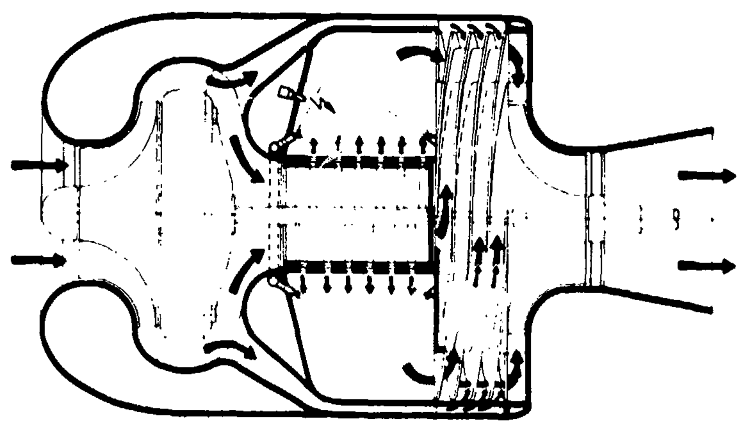


Figure 17

PERFORMANCE CHARACTERISTICS OF TURBO-ROCKETS AND TURBO-RAMJETS USING HIGH ENERGY FUEL

by
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56100 Pisa, ITALY

SUMMARY

Aerodynamic and thermodynamic behaviour of turbo-rockets and turbo-ramjets is considered. By means of variable engine geometry, multi-cycle engines meet aircraft requirements for take off, climb, cruise, maneuver, loiter and landing.
In these conditions, performance characteristics are carried out, taking into account variable geometry, in some intake and exhaust configurations.
Problems arising from future high energy fuels, particularly hydrogen, impose changes in interface components, geometry and control.
Preliminary designs of turbo-rockets and turbo-ramjets are conducted, both for military and civil applications.

LIST OF SYMBOLS

q = specific fuel consumption
M = flight Mach number
M/q = propulsion efficiency
L/D = aircraft lift-to-drag (aerodynamic efficiency)
$M/q \cdot L/D$ = flight efficiency
C_D = aircraft drag coefficient
C_F = engine thrust coefficient
D = aerodynamic drag
F = thrust
W = weight
W_i = all-up weight
F_o = sea level static thrust
V = flight and gas speed
R = range
A_e = engine exhaust cross sectional area
C_L = aircraft lift coefficient
ρ = density
p = static pressure
T = static temperature
$\dot{m}$ = mass flow
H = entalpy
h = altitude
g = gravity acceleration
C_p = specific heat
N = revolution per minute
ω = angular velocity
λ = compression ratio
θ = temperature ratio
u = peripheral velocity
k = specific heat ratio
Δc = peripheral gas velocity variation
P = power
f = fuel-air ratio

SUBSCRIPTS

t = stagnation
g = gas
a = air
o = ambient condition
1 = post-skock
2 = compressor face and by-pass valve
3 = compressor exit
4 = turbine face
5 = turbine exit
5' = ramjet combustion
6 = after-burning
7 = turbojet exhaust
7' = ramjet exhaust
8 = gas mixture

INTRODUCTION

The prospects for advanced low and high supersonic airplanes suggest configurations incorporating advanced materials and control concepts and utilizing variable-cycle engines, as turbofan-turbojets, turbo-rockets and turbo-ramjets. Hydrogen fuel offers great promise, even though formidable problems associated with the use of such fuel probably precludes its use on any near term second-generation supersonic airplane.

The flight efficiency $\frac{M}{q}$ L/D involves the aerodynamic efficiency, as expressed by the aircraft lift-to-drag L/D ratio, and the propulsive efficiency, as determined by the flight speed M to the engine specific fuel consumption q ratio. Variable-sweep designs, which utilize full-time stability augmentation systems, offer some interesting possibilities for aerodynamic efficiency. Variable-cycle engines, which try to meet the requirements of given flight programs, offer high propulsion system efficiency.

The propulsion system requirements of the long range low supersonic airplane dictate the use of an engine which operates as a turbojet for supersonic flight and as a turbofan for take-off and subsonic flight. No such engine exists at the present time. The hypothetical variable-cycle engine, for wich the specific fuel consumption (lb-fuel/lb thrust/hr) may gradually rise from 0.6 to 1.2 as the flight speed is increasing from rest to Mach 3.0, operates as a bypass ratio engine at subsonic speeds and as a turbojet in the supersonic speed range. The concept involves an engine in which the front-mounted fan utilized in subsonic operation becomes the first stage of compression in the turbojet. There are a number of component matching problems, however, which limit the variable bypass ratio concept.

The propulsion system requirements of the long range high supersonic airplane bring out the interest and the potentialities of composite airbreathing engines, as the combination of ramjet and turbine engines, on which some practical experience has been gained. The first example was the turbo-ramjet of the well-known Griffon, the only turbo-ramjet-powered aircraft that has ever flown supersonically, figure 24.
A difficult requirement facing turbo-ramjet designers is that of matching the operation of individual engine components to provide the proper amount of thrust to meet aircraft requirements for takeoff, climb, cruise, maneuver, loiter, approach and landing, at minimum cost.
The turbojet (or turbofan) has to incorporate variable fan vanes, variable compressor vanes, variable high and low spool turbine nozzle areas, and variable primary and fan duct exhaust configuration. The amount of variable geometry utilized in this engine provides improved flexibility in controlling engine operating pressures, thrust-turbine temperature-airflow relationships, engine bypass ratio, and transient response.
The ramjet needs variable geometry, from the external compression intake to the exhaust nozzle. In selecting a shock geometry, a choice has to be made between all external compression and combined external/internal compression designs, both permitting incorporation of the throat area variation necessary for engine matching over a range of flight speeds and engine flow requirements. Two major intake geometry components are incorporated to obtain the necessary geometry variations, needed for engine airflow matching at transonic speeds and for the proper flow area contraction at supersonic speeds. In choosing variable intakes and nozzles it is often found that the separate development of these components has led to the ultimate in individual performance but, in combination, it is by no means assured that these items will maintain that high performance, even with the aid of a complex control system.
The development of transient schedules for a turbo-ramjet usually involves a compromise between the need for both rapid thrust response and adequate compression system stability, especially at the change interface from an engine to the other. The variable primary stream nozzle of the enginemay be used to control airflow including adjustment for cycle variation due to altitude, bleed or horsepower extraction, and flight speed.
The turbo-ramjet is operating as a bypass turbojet when the flight speed is low, primary and secondary airflow going, respectively, through the low and high pressure compressors and through the low pressure compressor and the bypass duct. As the flight speed has reached Mach 3.0, the intake compression is sufficient for an adequate propulsion cycle without mechanically driven compressor, the flap bypass is closed to eliminate the turbomachine, and ramjet combustion is operating by means of afterburners. Variable ramps and shock system at the intake, and variable exhaust nozzle, may help the composite engine to solve matching problems during the whole flight program, as the air intake supplies either the turbo or the ramjet. Its high specific impulse makes possible to increase the range of the turbojet, but technology for a boost plus cruise engine is complicated.

The propulsion system requirements of the medium range supersonic boost airplane bring out the interest for the combination of turbo and rocket engine, with its average specific impulse, high thrust but complicated technology, and possibility as a heavy missile launcher. Variable geometry is necessary for matching operating conditions over the required performance range, through adjustable intake ramps and variable exhaust nozzle.
Combustion gases for the turbojet come out from the rocket, to drive the turbine connected to the compressor through reduction gear. Cruise, full power with afterburning, acceleration and deceleration phases need rocket as prime mover. Turbine exhaust gases mix with hot (at high Mach number) air flowing from the compressor, go through after combustion and expansion to exhaust. Rich mixture in the rocket chamber avoids high temperature on the turbine blades. Even if propellant consumption may be 50% more than the equivalent turbojet with afterburning (because of the oxygen aboard), the turbo-rocket may spend three times less than rocket.
On airbreathing engines it is important to have maximum thrust when the vehicle has going through sound speed. This usually happens at 33,000 feet altitude and Mach = 0.9, with 70°F air intake temperature, reaching a 3650°F possible maximum temperature on the duct preceding the exhaust nozzle.
The turbo-rocket should be essentially reserved to missile launch aeroplanes. One of these could be for instance equipped with two turbo-rocket engines, weighing 100,000 lb at take-off, including a two stage 25,000 lb solid rocket. The missile should be launched at Mach 4.5 and 65,000 feet of altitude, putting on terrestrial orbit a 500 lb satellite.

The propulsion system requirements of the low range boost vehicle bring out the interest for the air augmented rocket. Suitable for high thrust and average specific impulse, such engine could be used at flight speed less than Mach 4.0, with variable geometry.
The air augmented rocket, for the propulsion of boost vehicles, is known as an engine which improves on the propulsion efficiency of the rocket at low flight speeds, and which requires a smaller weight of manufactured structure than other airbreathin engines.
Particular attention is paid to features affecting installation and use in an aircraft designed as the first stage booster for a manned spacecraft. Simple engines with fixed geometry and variable airflow have realized at best about one third of the potential gain in specific impulse due to air augmentation. The failure to gain more is very largely due to the difficulty of arranging a fixed geometry which allows for expansion of the exhaust stream, after mixing is completed. With a fixed outer cowl, the specific impulse can be improved by arranging for the air intake diffuser to vary internally. A beneficial variation in air duct area could be arranged with comparatively simple moving parts operating in the cooler part of the engine. This has a double effect in improving performance; reduction in diffuser outlet area at high flight Mach number assists in intake matching.

Ducted turbulent mixing of coaxial streams occurs in the composite propulsion systems such as the above mentioned engines. Quite difficult it is to develop an adequate engineering theory to describe the ducted mixing process, including chemical reactions. It is a strong viscous interaction problem in which the mixing layer may extend over most or all of the duct cross section at the exit plane. The variable-area, variable pressure ejector can be used to augment the net thrust.

It is purpose of the present paper to make approximate comparisons of the performances characterizing such airbreathing, composite and combined cycle, engines, considering hydrogen as fuel.

ADVANTAGES AND PROBLEMS WITH HYDROGEN FUEL

Another approach for achieving improved flight efficiency is provided by the use of fuel having a heating value greater than that of current hydrocarbon fuels. Table 1 lists properties of some candidates for replacing petroleum products, with JP (Jet A-1) shown for comparison, Ref. 1.
Many factors condition a choice among them. The use of liquid methane as a fuel for the supersonic transport has been studied in some detail. The heating value of this cryogenic fuel is only about 15% greater than that of conventional fuel. Liquid hydrogen, however, has a heating value about 2.8 time that of hydrocarbon fuel. The higher heating value of hydrogen results in a flight efficiency factor which is nearly three times greater than that for an aircraft utilizing conventional fuel. Spectacular reduction in the fuel weight required for a given range would therefore appear to be potentially possible. In addition, higher turbine inlet temperatures and thus more efficient engines would be possible through the use of the very cold hydrogen fuel for turbine blade cooling. Finally, the product of combustion of hydrogen and oxygen is water vapor. Thus, a hydrogen burning engine would have a clean exhaust except, when using air, for the NO_x compounds.
Hydrogen can significantly improve aircraft

Table 1

	JP Fuel	Methane	Ethyl Alcohol	Methyl Alcohol	Ammonia	Hydrogen
Nominal composition	$CH_{1.94}$	CH_4	C_2H_5OH	CH_3OH	NH_3	H_2
Molecular weight	≈120	16.04	46.06	32.04	17.03	2.016
Heat of combustion BTU/pound	18,400	21,120	12,800	8,600	8,000	51,590
Liquid density Lb/ft^3 at 50°F	47	26.5●	51	49.7	42.6●	4.43●
Boiling point (°F) at 1 atmosphere	400 to 500	-258	174	148	-28	-423
Freezing point (°F)	-58	-296	-175	-144	-108	-434
Specific heat BTU/Lb °F	0.48	0,822	0.618	0.61	1.047	2.22
Heat of vaporization BTU/Lb	105-110	250	367	474	589	193

● At boiling point

performance. Only weight-sensitive applications such as aircraft would benefit so much from the very high heat of combustion available with hydrogen.

For use in aircraft, hydrogen would be tanked in liquid form, considerably affecting airplane design because:

- hydrogen weight for given mission is less than comparable JP design by a factor of 2.8;
- high specific heat affects airplane design because hydrogen can be used to cool engine hot parts and provide heat sink for aircraft cooling requirement;
- lesser weight of fuel required leads to low wing loading, high cruise altitude, higher power loading;
- low density affects airplane design because even though fuel weight is less, volume required is greater by a factor of about 3.78, resulting in lower aerodynamic efficiency, i.e. lower L/D.

The storage of hydrogen in cryogenic tanks is under consideration. Hydrogen is cooled to -422.9°F to reach a state of liquefaction, and stored in that form by well insulated tanks and tank pressure scheduling to minimize boil-off losses. The insulation is sufficient to maintain the liquid temperatures for several days. When small amounts of hydrogen are vaporized by heat that does enter the tank, the gas is vented through a catalytic exhaust tube where the escaping hydrogen is flamelessy converted to water vapor.
Several aluminium alloys and stainless steels are applied in lightweight structures for cryogenic liquids at extremely low temperatures. Improved materials with higher specific strenghts and increased toughness will be required in the future: weldable aluminium alloys, titanium alloys and fibre reinforced composite materials.
The main requirements to be fullfilled should be: high ratio of strength/density, resistance to stress corrosion cracking especially in the region of welds, fatigue life at high stress levels and compatibility with the cryogenic liquid, adequate toughness, weldable and easily formed. Materials in sheet and plate form, to reduce the mass of the propellant tank, need to have a notched tensile stress/ultimate tensile stress ratio (NTS/UTS) greater than 0.9 for adequate toughness in structure applications, Ref. 2.

All cryogenic propellant tank structures that have been unpressurized and self-supporting have been con-structured from aluminium alloys. There has been a gradual improvement in the proof stress/density ratio of materials from 0.22×10^6 in for the aluminium-magnesium alloy 5052-H32, which was selected in 1951 for the liquid oxygen tank of the Redstone missile, to 0.56x10 in for the aluminium-copper alloy 2014-T6, which was chosen in 1962 for both the liquid oxygen and liquid hydrogen tanks for the S-II and S-IV B stages of the Saturn V rocket. The more recently developed aluminium-zinc-magnesium alloys have specific strengths of approximately 0.71×10^6 in with NTS/UTS ratio 0.98÷1.07.

Austenitic stainless steels of 18% chromium, 8% nickel content, in the annealed condition have low specific strengths of approximately 0.3×10^6 in. Improvements in this value can be achieved by cold working the material. AISI Type 301 cold rolled 60% to a minimum tensile strength of 200,000 lb/in^2 is used in the liquid oxygen tanks for Atlas and the liquid oxygen and liquid hydrogen tanks of Centaur. The liquid oxygen tanks of stainless steel, FSMI, which is cold rolled to a minimum tensile strenght of 170,000 lb/in^2.
The manufacturing processes used in all the above tank structures are similar. The wall thicknesses vary from 0.01 in to 0.04 in, and the tanks are fabricated by butt welding cylindrical segments.

For the future, the object is to find materials that have improved specific strength aluminium alloys with an ultimate strength of 75×10^3 lb/in^2 at 68°F, a NTS/UTS ratio of 1.0 at 68°F and 0.9 at -422,0°F, an "as-welded" NTS/UTS ratio of 0.85 at-422,9°F. Some of the aluminium-zinc-magnesium alloys already fullfill certain of the above conditions, and in addition have the following characteristics: insensitive to quenching rates, resistance to hot cracking if correct welding procedures are employed, and age hardened naturally or artificially after welding to yield high joint efficiences.

A more intensive effort on the properties of fibre reinforced composities is necessary before they can be applied with confidence in light weight cryogenic structures.

Liquid-hydrogen production is nowadays based, in the main, on natural-gas feed-stock. On the other hand, water electrolysis for hydrogen (and oxygen) production is well developed for specialized applications. However, it uses expensive electrical energy. So the price of hydrogen principally reflects power costs. But future developments in water-splitting systems will make hydrogen cost competitive. Over the long haul, water may be our only large-quantity source of hydrogen.
At the moment, the only well-developed water-splitting process we have is electrolysis. Solar energy may some day power hydrogen generation. And hydrogen may prove a better medium than electricity.
Projecting to the last decade of this century and beyond hydrogen may be, as a consequence of recent events on the energy front, the "economy fuel" of the future.
However, the required growth of the liquid hydrogen industry to support a fleet of hydrogen-fueled supersonic transports will be tremendous, Ref. 3. The fact that the single largest liquid hydrogen plant, placed into operation in support of the U.S. Space Program at about 60 ton/day, would support about one flight per day by a single supersonic transport (range mile 4,200 n.mi, cruise speed Mach 2.7, payload 250 passengers) places this fact in striking perspective.
Ref. 4 examines a production rate of 2500 ton/day of liquid hydrogen using nuclear-electrolysis. This equates to an electrical output requirement of 6,200 MW, or about six 1,000 MW nuclear units of the size-class currently being installed into US electrical utility system. This would service about 50 aircraft flights dayly on the same basis as described above. A production level of 2,500 ton/day by an ocean-based solar-to-hydrogen facility would require a collector area of 14x14 Km array, corresponding to 0.00316 watt/cm^2.

Hydrogen-fueled aircraft offer worthwhile environmental gains: emissions of carbon monoxide, carbon dioxide, hydrocarbons, and smoke are virtually eliminated.
Hydrogen provides for a much wider fuel/air ratio operating range than conventional jet-fuel. Because of this, hydrogen fuel brings with it the potential for theoretical reductions in nitric oxide formation of about 2 orders of magnitude for the same primary zone dwell time. Dwell time is a measure of the time the combustion products remain at full temperature prior to air dilution and/or expansion in the turbine, usually the order of millisecond or so. A second significant route to lower oxides of nitrogen production is much higher "space heating rates" of the hydrogen; this giving the possibility to considerably shorten engine combustor length and associated dwell times.
The relative impact of oxides of nitrogen and water vapor on the stratosphere by a hydrogen airplane is not yet known. The effect of nominal increase in water vapor exhausted at high altitude may be too high with supersonic flight; subsonic aircraft are limited to the upper atmosphere with its much more active circulation and "turnover times". Weather and climate modifications, as a result of injection of water and nitric oxide, should derive from the possible reduction of the naturally occurring ozone in the lower stratosphere with the result that atmospheric ozone in the lower atmosphere is turned to oxigen, thus reducing the shielding capability of the ozone layer and increasing the amount of ultraviolet radiation reaching the earth's surface.

HYDROGEN FUELED SUBSONIC-SUPERSONIC AIRCRAFT

The convenience of hydrogen as fuel for commercial aviation appears from the flight efficiency values $M/q \cdot L/D$, figure 1.

Specific fuel consumption q with JP fueled variable cycle engines is increasing with flight speed from 0.65 lb fuel/lb thrust/hr (low speed) to 0.9 (M = 0,9) and to 1.6 (M = 3.0). Aerodynamic efficiency L/D with JP fueled aircraft has a value of 12 ÷ 16 for subsonic transport and of 9 for supersonic transport

like Concorde and Boeing 2707.
Therefore, maximum flight efficiency is 12 ÷ 16 for subsonic transport and may be so high for supersonic transport at the highest speed, both with JP fuel.

Specific fuel consumption q with hydrogen fueled engines may be about 0.21 ÷ 0.30 in subsonic flight and 0.45 ÷ 0.60 at M = 2.7 ÷ 3.0. Aerodynamic efficiency L/D with hydrogen fueled aircraft has a value of 12 ÷ 15 for subsonic transport and 7.5 ÷ 10 for supersonic transport.
Therefore, maximum flight efficiency may be more than three time using hydrogen fuel instead of JP fuel, both for subsonic and supersonic transport, considering the possibility to adequate aerodynamic configuration and propulsion to the hydrogen higher specific volume and heating value.

Now, in terms of range/payload related to subsonic and supersonic transport using JP fuel, it is known that: a subsonic aircraft carries a passenger payload of about 10% of its gross weight for a distance of 5600 n. miles, an amount corresponding to 45% of the gross weight being reserved to the fuel; in a supersonic aircraft similar to the Boeing 2707, for the same gross weight and structural weight a range of approximately 3500 n. miles, the payload fraction is reduced to 6.5% of the gross weight. This percentage increases to 12.5% if the lift-to-drag ratio increases from 7.5 to 10, assuming that the off-loaded fuel weight could be directly replaced by payload weight (fuel percentage from 55% to 49% of the gross weight).

As numerical example about the feasibility of using liquid hydrogen for both subsonic and supersonic transports, we refer to the study performed by Lockheed-California for NASA-Ames, Ref. 1.

The subsonic study analyzed the potential of a hydrogen-fueled equivalent of an L-1011 Tristar. The LH_2 version was required to match the mission capability of the original: payload 56,000 lb, range 3,400 n. miles, cruise speed M = 0.82, engines of equal thrust.
Performance characteristics resulted, for the JP fueled version and (the LH_2 fueled version):

- gross weight 430,000 lb (318,000 lb)
- payload/gross weight 13% (17.6%)
- fuel weight 137,000 lb (46,650 lb)
- fuel/gross weight 31.85% (14.65%)
- fuel volume 2,920 cu ft (11,050 cu ft)
- aerodynamic efficiency L/D = 15.5 (12.6)
- specific fuel consumption q = 0.677 lb/lb/hr (0.216 lb/lb/hr)

The LH_2 version showed an increase of 278% of fuel volume, requiring two tanks mounted on the wing to carry the hydrogen. A reduction in L/D of over 18% for the wing tanks introduced a penalty in aerodynamic performance. Engine performance with hydrogen was assumed to be only 12% lower in specific fuel consumption. Such subsonic study showed: a flight efficiency $\frac{M}{q}$ L/D = $\frac{0.82}{0.677}$ 15.5 = 18.8 for the JP fueled original version, and $\frac{0.82}{0.216}$ 12.6 = 47.8 for the LH_2 fueled version; a potential saving in gross wright of 26%, in fuel weight of 66%, and in wing area of 18.2%.

The Lockheed supersonic study analyzed a hydrogen-fueled arrow-wing supersonic transport design, in comparison with an equivalent JP-fueled design. Both vehicles were designed to the same ground rules: 1981 state of the art; the same FAA standard for noise, for takeoff and landing, and for reserve fuel. The LH_2 version was required to match the mission capability of the JP version: payload 49,000 lb (234 passengers), range 4,200 n. miles, cruise speed M = 2.7.
Performance characteristics resulted, for the JP fueled SST and (the LH_2 fueled SST):

- gross weight 750,000 lb (379,000 lb)
- payload/gross weight 6.5% (13%)
- fuel weight 391,300 lb (98,000 lb)
- fuel/gross weight 52% (25.85%)
- fuel volume 8,290 cu ft (22,100 cu ft)
- aerodynamic efficiency L/D = 7.5 (8.0)
- specific fuel consumption q = 1.51 lb/lb/hr (0.561 lb/lb/hr)
- thrust per each of the 4 engines 89,500 lb (47,350 lb)
- energy per seat mile 6102 BTU/seat n. mi. (4375 BTU/seat n. mi.)

The LH_2 version showed an increase of 267% of fuel volume, requiring two large fuel containement areas on the fuselage in front and back of the passenger cabin (with upper and lower seating areas), 35 ft longer than the JP version. Such supersonic study showed: a flight efficiency $\frac{M}{q}$ L/D = $\frac{2.7}{1.51}$ 7.5 = 13.4 for the JP fueled version, and $\frac{2.7}{0.561}$ · 8 = 38.5 for the LH_2 fueled version; a potential saving in gross weight of 50%, in fuel weight of 75%, and in wing area of 30%.
It is clear there are important gains by using hydrogen for fuel in commercial transport aircraft. This because it leads to lower operating cost for the airlines: maintenance items due to low gross weight, smaller engines, lower airframe weight, and energy expended per available seat mile.

The relatively large fuel fraction for the LH_2 fueled supersonic transport is primarily responsible for the poor payload fraction. The fuel is utilized as it follows: take-off, climb and acceleration about 25%, cruising phase of flight 60%, descending and landing 1.5%, reserve fuel 13.5%.
The variable-cycle engine, for which the specific fuel consumption increases gradually from subsonic flight to low supersonic Mach, operates as a bypass ratio engine at subsonic speeds and as a turbojet in the supersonic speed range. The large potential benefits of the variable cycle engine make it an attractive candidate for intensive future research and development.

The flight efficiency $\frac{M}{q}$ L/D, for JP and LH_2 fueled commercial aircraft, is shown in figure 1 as a function of Mach number for a present-generation supersonic transport, and for advanced arrow wing and variable-sweep concepts equipped with both dry turbojets and variable bypass ratio engines.
The following points are immediately apparent from an examination of the curves presented in figure 1 for both JP and LH_2 fueled transport aircraft:

- large improvements in the range/payload characteristics of future supersonic transport aircraft are made possible by the higher flight efficiency of the arrow wing aircraft employing a non-afterburning engine, in the range of Mach numbers corresponding to climb and acceleration and to cruise flight Mach number of 2.7, especially using liquid hydrogen;
- the use of a variable cycle engine results in a higher flight efficiency at subsonic and supersonic speeds.

We have seen that, referring to subsonic and supersonic commercial transport aircraft, potential gains in performance are associated with the use of advanced arrow wing and variable-sweep concepts (for subsonic speeds) together with light variable-cycle engines. The use of hydrogen fuel provides another approach for achieving improved flight efficiency.
For other applications and for higher cruise supersonic Mach number, hydrogen fuel and variable-cycle engines may be a good way to improve performance characteristics.

VARIABLE GEOMETRY CONFIGURATIONS FOR TURBO-RAMJET AIRCRAFT

The high specific impulse of the turbo-ramjet, a variable geometry and multicycle engine as in the sketch of figure 2, makes possible to increase range and cruise speed of the turbojet, maintaining high performances with the aid of a complex control system.
For low flight speed, the engine is operating as a normal turbojet, with the primary airflow going through both compressors of low and high pressure, and the secondary airflow going through the low pressure compressor and the bypass duct. When the flight speed has reached Mach 3, the intake compression is sufficiently high for a propulsion cycle and the mechanical driven compressor is no more necessary. The flap bypass (B) is thus closed to eliminate the turbomachine, and the plant is starting to operate as a ramjet, using existing afterburners for combustion. The air intake cross section is adjusted through a variable collar (A) actuator; the exhaust nozzle throat and exit areas are adjusted through two retractable coaxial ducts (C), (D). The flap bypass (B) in front of the second compressor closes the high pressure section of the engine.

Engine operation phases are as it follows:

- <u>takeoff</u>: intake collar (A) fully open; flap bypass (B) open and high pressure section operating; external nozzle duct (D) retracted and internal nozzle duct (C) partially extended to give nozzle cross section area sufficient for a low heating degree through afterburners (E). After takeoff and during subsonic climb, flaps (A) and (B) remain unchanged, as well as the maximum power rating of the high pressure section. The nozzle exit cross section area is gradually increased, retracting the internal duct (C) to adequate the rising gas flow temperature;
- <u>transonic acceleration</u>: as at takeoff; but the ramjet afterburners are completely operating, and the internal and external nozzle ducts (C), (D), are fully retracted to give maximum nozzle exit area;
- <u>supersonic cruise for M = 5</u>: during supersonic climb, the intake air cross section area is gradually reduced through flap (A) and variable ramps (F). At about M = 3, the high pressure section is excluded. For higher Mach, the low pressure compressor is freely rotating, and the whole airflow has going to the ramjet heating system. Both the nozzle ducts (C), (D), are extended to reduce the throat area and to increase the exit area for full ramjet heating;
- <u>subsonic cruise</u>: air intake area fully open for subsonic operation; flap bypass (B) open and fuel flow in the combustion chamber for turbojet operation; external nozzle duct (D) retracted and internal duct (C) adjusted for the required nozzle exit area.

These operation phases require a lot of control and variable geometry.

To design a composite engine like the one in figure 2, suitable for a speed field from low subsonic to high supersonic, we have to re-consider the variable geometry inlets and nozzles, their interfaces with the ramjet and turbojet engines, and the flow separation and mixing conditions.
The main function of the intake consists of picking up efficiently the atmospheric air requested by the engine and delivering it as a high quality flow. To meet the very stringent specifications imposed on the intake because of its interference with the propulsion units, several controlled elements, movable or expansible, are necessary.
Figures 3 and 4 show sketches of two types of variable geometry supersonic inlets, two-dimensional and axisymmetric respectively. Both two-dimensional and axisymmetric designs are used for mixed compression inlet, figures 4 and 5. Some of the different mixed compression inlet design types are illustrated in figure 6. In order to produce high thermodynamic performance, a mixed compression inlet must approximate critical operation (terminal shock just aft of the throat) requiring precise, rapid response control of throat flow conditions; as the inlet in figure 7, corresponding to the ramjet starting condition of the turbo-ramjet in figure 2.
In the schematic view of the intake shown in figure 8, Ref. 5, two major geometry components are incorporated to obtain the necessary geometry variations needed for engine airflow matching at transonic speeds and for proper flow area contraction at supersonic speeds. The flow area required for transonic engine

airflow matching is shown as the minimum flow area just aft of the intake cowl lip station. This area allows a flow Mach number of 0.85 at full engine power and includes allowance to pass the engine secondary air cooling flow. To obtain the flow area variation required between the curves labeled "takeoff" and "transonic", a set of four cowl throat doors is used. These are 90° segments of the inner cowl, hinged at the forward end and rotated cutward between parallel-sided (90°) structural beams in the intake. For supersonic airflow matching, the centerbody retracts from its fully extended position. The intake internal flow area variation provided by this translation is shown in the area progression curves of figure 8 as the area decreases between the curves labeled "takeoff" and cruise".
A schematic of the secondary air valves and overboard bypass and takeoff doors is shown in figure 9. There are eight butterfly-type secondary air valves, two per 90° segment in the intake secondary flow channel. They are used to regulate cooling air to the engine secondary nozzle, to serve as a fine control for positioning the normal shock during started intake operation. There are four overboard by-pass doors, one per 90° segment. These doors perform the well known function of spilling the large amounts of intake excess airflow that occur during airplane descent and inoperative engine conditions.
The intake/landing ($M < 0.5$) configuration is shown in figure 9 b, where, the centerbody is extended, the throat and overboard bypass doors are closed, and the secondary air valves are open. The minimum duct flow area now occurs well aft in the intake as shown in figure 8. With the throat doors in the closed position, the auxiliary takeoff doors are allowed to float free of the overboard bypass door frames. On figure 10 it is shown a generalized schematic view of the buzz suppression mode, with the centerbody fixed at an extended position of 90% of full travel and the throat doors are placed at 90% of the full open position. The buzz suppression control mode is manually initiated during climb at approximately Mach 1.2 (to Mach 2.9 in the figure).

For higher Mach number, from 3 to 5, the starting conditions of the mixed supersonic compression inlet of the turbo-ramjet of figure 2 are theoretically to be found in a configuration such that a normal shock occurs right at the entrance of the internal convergent section,because of variable geometry. The air intake needs a centerbody retracting and extending to obtain a minimum flow area variation. A sketch of a propulsive assembly is shown in figure 11, where it is also indicated a possible supersonic chamber of combustion. The air intake, destined to feed a chamber of combustion operating on supersonic airflow at maximum flight Mach number, comports a conical nose and an annular supersonic convergent nozzle, figure 12 a). The only adjustment of this air intake is the displacements of the centerbody to modify the variable geometry as function of the flight speed. When the annular nozzle becomes convergent-divergent with a throat area, as in figures 12 b) and 12 c), the airflow at the exit and in the chamber goes through subsonic mixing and combustion. In the case of supersonic turbulent stream, the important factor is the time needed to mix and burn the fuel/air. The enormous velocities involved require extremely short overall times in order that combustion chamber lenghts will not be excessive. The longest process is generally the mixing and hence the combustion must take place in a much shorter time to minimize the overall time. The combustion chamber length would then correspond closely to the length required for mixing only.

The air intake of a turbo-ramjet like the one in figure 2 is largely designed to decelerate the air flow approaching at the flight Mach number, with the low ssible loss, to the subsonic velocity which is determined by the requirements of the subsequent : compressor or combustion chamber. This deceleration is carried out in a wholly or partly closed or or partly variable geometry, but usually without using rotating parts.
The pre-entry flow of the air intake is characterized sharp rise in pressure. At supersonic speed, the pressure rise corresponds to the deceleration of ai through a shock wave system, which may or may not be followed by a further subsonic deceleration before the entry.
For subsonic and possibly low supersonic speeds, the pre-entry pressure rise poses no particular problem in relation to design for good pressure recovery and flow distribution. For supersonic speed, the pre-entry pressure rise has an important influence on the design, on account of its interaction with boundary-layers in the flow field approaching the entry plane (figure 13). The pre-entry pressure rise causes distortion of the boundary-layer profile and, in many cases, flow separation. It is accepted pratice to minimize the interaction effect by use of a boundary-layer bleed or diverter. To the extent to which the boundary-layer is thereby by-passed from the main intake, the viscous interaction effects are aliminated and design values of pressure recovery and flow uniformity are achieved.
In fact, intakes designed for efficient operation at Mach number above 2 have at least a small amount of boundary layer bleed extracted in the region of the terminal shock wave of the supersonic compression system. For example, in the two-dimensional shock system shown in figure 14, the bleed mass flow is controlled by an orifice at the exit to the bleed chamber extending through the engine bay with the controlling orifice formed by the expansion boundary of the primary jet and the nozzle divergence wall in the nozzle secondary system. In figure 15 the two-dimensional all-external compression design with three external compression surfaces has the initial compression surface and the inlet cowling fixed, yielding a convetional constant capture area induction system design. The second and the third external compression surfaces toghether with the subsonic diffuser ramp may be rotated indipendently about their respective hinge axes. Inboard and outboard side walls are provided in control lateral flow spillage from the inlet. Finally, the design embodies a bleed/bypass slot located just downstream from the cowl closure station, for the purpose of boundary layer and excess flow management. The throat slot flow is connected with a plenum region to a fully articulated convergent exit door located at the aircraft wing top surface.
For instance, the optimum efficiency of the propulsion system throughout the entire flight envelope of a fighter aircraft with high supersonic speed capability requires a variable geometry air intake, as in figure 16. The intake is of the two-dimensional, horizontal, double wedge, external compression type, with

an approximate square cross-section at the intake lip plane and is comprised of a fixed first ramp, a second ramp hinged at its front edge and a third ramp hinged at its rear edge forming the upper surface of the intake. Variation of the intake geometry is accomplished by the movement of the second and third ramps by means of a hydraulic actuation system through a linkage. Between the two movable ramps a wide slot is provided by which bleed air is exhausted overboard through fixed bleed exits on top of the intake. It is purpose of the air intake control system to position the variable surfaces of the intake in order to match intake airflow to engine demand at acceptable distortion levels to provide the maximum possible installed thrust of the powerplant system under steady state and transient operating conditions over the whole flight envelope.
The basic choice for inlet placement may be guided by several operational factors of aircraft utilization such as missile and weapon placement, and foreign object ingestion from unprepared runways. A common design feature of recent tactical aircraft is the placement of the inlet on the side of the fuselage where each inlet is supplying air to one engine, as in figure 17. In such a way, the fuselage cross-sectional shape effects the performance and operational stability.
In the Concorde powerplant, figure 18, the intake is mounted off the wing with a vertical separation calculated to eliminate wing boundary layer ingestion at 1 g' conditions at 2.0 M. The intake compression geometry is predominantly external compression, designed for first shock on lip at a local Mach number of 2.1 M. Intake geometry is varied by means of the movable front compression surface (front ramp) and a spill door mounted in the floor of the main diffuser (dump door). The bleed air is used to ventilate the engine bay before being exhausted via the dual stream secondary nozzle.

The intake of the turbo-ramjet of figure 2 is of axisymmetric design as in figure 19. This may be followed by a multistage split flow fan, freely or power rotating, in conjunction with an indipendent multistage core compression, as in figure 20. The modern conventional engine technology put at disposal for this, the duct-burning turbofan, as the SCAR engine shown in figure 21, associated with variable stator vanes operated by individual stage support rings and vane actuating levers. The pin in each lever engages in its respective slot in one of the actuating rings which fit around the compressor casing and move circumferentially on bearings located in the support rings. The actuating rings are positioned by the actuator assembly which pivots the vanes to the desired angle.

The structural assembly of the ramjet, following in figure 2 the split fan, may consist of a Hastelloy skin backed by an integral offset-fin ring-stiffened heat exchanger through which the hydrogen fuel is circulated to provide cooling of the structure before injection in the combustor. From Mach 3 to Mach 5 the performance is considered for subsonic operation; at Mach 6 it should be possible subsonic and supersonic combustion operation, and at Mach 7 supersonic combustion.

The variable geometry nozzle can only be chosen in the light of its combined effect with the intake on the overall aircraft affected. The throat bleed flow from the intake may be taken, figure 22, to the exterior and exhausted separately. From practical considerations it is found to be essential to supply the nozzle with an amount of air extra to the primary system in order to give good performance and flexibility. However, as the bleed air passes from the intake throat to the secondary system of the nozzle it will incur losses which, if large, will seriously affect the balance of the system, resulting in losses in overall performance. The position and size of the annulus through which the secondary air flow, figure 22, is introduced into the divergence of the nozzle has a profound effect on the quantity and quality of the flow that the nozzle will accept.
A large variety of nozzle systems exists nowadays, Ref. 6, each having its particular features. Figure 23 gives a survey of various designs. The fixed convergent nozzle is used with airplanes for subsonic flight, only without thrust augmentation by afterburning, such as with civil transports. Early jet fighter aircraft used ejector nozzles. Turbojets needed a flow of secondary cooling air which could be obtained from the ejector action of the primary jet. This it is still true for turbojet engines for supersonic transports. A later development was the introduction of extra tertiary air intakes at the nozzle location. These blow-in-door ejector nozzle gave performance gaines at transonic flight.
A good intake/nozzle matching may be reached in the case of the turbo-ramjet engine, as in figure 2 where it is possible to vary both the nozzle throat and exit areas.

The first airframe/turbo-ramjet integration was realized with the Griffon, figure 24, a delta-winged aircraft designed in 1953 to meet the requirements of a light interceptor program. It carried out 300 experimental flights, reaching a Mach number of 2.2. Only structural limitations on the airframe prevented it from flying at even higher Mach numbers.
It consisted essentially of a coaxial arrangement of ramjet and turbojet with common air inlet and exit nozzle. Part of the captured air was spilled around the turbojet to feed the ramjet combustion chamber which surrounded the turbojet. This was a SNECMA Atar 101 E3 dry turbojet with 7,700 lb static thrust, Mach limited to 1.85. No need for variable geometry, either for the inlet or the nozzle, arose. In fact, the inlet efficiency resulting from the combination of the oblique shock from the fuselage spike and the normal shock at the inlet face was quite acceptable. Also along the climb path, air inlet and exit nozzle remained well matched. The ramjet was ignited at about Mach 0.5. Considerable acceleration resulted, permitting rapid passage through the transonic range and supersonic climb.
Satisfactory results are made clear by looking at figure 25, Ref. 7. It shows the drag of the Griffon along a typical climb acceleration profile. The lowest curve gives the thrust coefficient of the turbojet and the upper curve the thrust coefficient of the turbo-ramjet. It shows that considerable excess thrust was available. As a reference is also plotted the thrust coefficient of the ATAR G, which was the afterburning version of the ATAR 101 E3, used on the Griffon.

Interference between the turbojet exit and the ramjet was negligible in flight.

Some aerodynamical aircraft configurations may be chosen to incorporate hydrogen fueled turbo-ramjets. For example, to meet the requirements of a high supersonic military interceptor of fighter airplane, a modified shape of the original Griffon may be a good solution, as in figure 26. The axisymmetric inlet and the composite engine must be selected to be compatible with each other and with the mission and to achieve improved performances during maneuvers, thrust changes, and atmospheric disturbances. An electronic control must be designed to ensure efficiency and tolerate the transients. The importance of the inlet design may be understood considering that, for example at Mach 2.5, the inlet generates half of the thrust, the engine supplying the remainder.
The three inlet performance characteristics which have the largest effect on aircraft range are total pressure recovery, boundary layer bleed, and cowl drag. Fighter-type aircraft use, at supersonic speed, all-external compression inlet (area change that occurs in front of the cowl lip). Long range cruise vehicles use a mixed-compression inlet, figure 27. The cowl angle determines the strength of the shock generated by turning suddenly the supersonic flow. Since drag and pressure recovery both affect aircraft range, it is necessary to select an optimum cowl angle.
Boundary layer bleed flow is needed to avoid separation and to provide good performance. The reason for that is because, at a flight Mach number of 2.5, the static pressure ratio across the supersonic section of an inlet is about 11, pressure gradient that acts against the flow and causing it to separate from the walls. This problem is controlled by removing some of the low energy air (boundary layer bleed) near the walls at the points where the shock waves reflect. Facing the flow an additional pressure rise in the subsonic section of the inlet, enough low-energy air is removed to help the flow through this section too.
The inlet control is realized with bypass doors. A signal that represents measured shock position is fed to the control for comparison with requested shock position. The difference causes the control action, positioning the bypass door to maintain a nearly constant-airflow.

The controllability of a mixed-compression inlet can be improved by integration of the inlet and engine control systems; this provides an alternate approach for handling the turbojet light-off and the ramjet light-in and viceversa.
In the turbo-ramjet propulsion system, the control inputs include inlet shock position, engine rotor speeds, ramjet operation, and other appropriate propulsion system pressures and temperatures. The control outputs are inlet bypass door position, spike position, fuel flows (both turbojet combustion chamber and ramjet burners), compressor stator vanes or bleeds, and exhaust nozzle area. In the automatic integrated control, each of these control inputs directly affects each of the control outputs, for their best use.

The inlet's role is extremely important with high supersonic cruise transport aircraft. At Mach 3, the inlet generates about 70 percent of the thrust. If the inlet unstarts, its pressure recovery drops drastically, which can upset the flight stability. To solve this problem, bleed happens from the inlet throat, where a porous skin is installed.
For supersonic transport aircraft, the location of the turbo-ramjet engines is important to the perfomance and safety. Problems regarding various arrangements of the propulsion packages relative to one another arise from the fact that unstarts of mixed compression inlets cannot presently be completely eliminated, and that the unstart of one inlet must not initiate the unstart of an adjacent one, since the two unstarted adjacent inlets would precipitate extreme rolling and yawing moments. For that, it was proposed the twin inlet design, as in figure 28. This arrangement places two propulsion packages in a single nacelle and attempts to insure independent operation by employing a splitter plate which divides the nacelle and extends forward, isolating two inlets. This arrangement allows placement of the engines more nearly on the centerline of the transport aircraft, which is desiderable for structural and aerodynamic reasons.

Other possible arrangements of turbo-ramjet engines with two-dimensional inlets and variable centerbodies to change the geometry for off-design operation, usually for flight Mach number no more than 3, are shown in figures 29, 30, 31 and 32.

HYDROGEN FUELED TURBO-RAMJET/AIRCRAFT BASELINE DESIGN

To assure realistic frameworks for baseline designs, representative advanced supersonic strike and transport configurations were established, figures 33 and 34, to meet the requirements of typical mission scenarios.
The defined missions and their ranges, figures 35 and 36, served as references for the induction system/ mission effectiveness analyses and the preliminary propulsion system development, while the aircraft configurations theirselves provided the basic geometries for overall designs.

The multi mission combat aircraft of figure 33 is a twin-turbo/ramjet engined configuration, rated at 24,000 lb sea level static thrust for each engine. Basic operating weight and maximum takeoff weight are respectively 40,000 and 63,575 lb. Maximum level speed at height is Mach 3.5. Additional liquid hydrogen for first climbing and cruising to the combat area is contained in tanks mounted on the wing. The baseline mission, figure 35, comprises: take off and climb to best cruise altitude; cruise out at subsonic speed to the forward edge of the battle area; penetration at sea level altitude and transonic speed; "pop-up" in target area, followed by interception engagement; climb and acceleration to supersonic speed and return dash; lapse to subsonic cruise, followed by loiter and landing.
The engine inlet is a rectangular, two-dimensional design three external compression surfaces.

The arrow-wing supersonic transport aircraft of figure 34 is a twin-axisymmetrical inlet configuration with four turbo-ramjet engines, for a maximum speed at height of Mach 4.5. The design departs from previously shown configurations by electing all-fuselage storage of the liquid hydrogen. The double-deck arrangement shown in the figure was adopted for the passenger cabin, in conjunction with full cross-section cryogenic fuel tanks fore and aft, in order to provide: fuel tanks with high structural and volumetric efficiency; minimum c.g. travel as fuel is consumed; and maximum separation of passengers from fuel. Gross takeoff weight, total fuel weight, and payload weight are respectively 432,390 lb, 194,577 and 54,050 lb. The baseline mission, figure 36, comprises: takeoff and subsonic climb; transonic acceleration; supersonic climb; supersonic cruise; deceleration to descent speed; transonic deceleration; initial and final descent; ground approach and landing.
Thrust per engine is 73,230 lb.

Multimission combat aircraft - The schematic outline of the turbo-ramjet chosen for the present computation is shown in figure 37. The air intake is designed to decelerate the air flow approaching at the flight Mach number, with the lowest possible loss, to the subsonic velocity which is determined by the requirements of the subsequent component: turbojet compressor, ramjet combustion chamber or both. The necessary performance is well provided by external and internal compression, designed in order to avoid the unstart problem characterizing a mixed-compression inlet. (With internal compression, the normal shock is located inside the inlet and is stable if located downstream of the throat, where the walls diverge. However, the highest total pressure recovery occurs when the shock is at the throat. If the shock is moved ahead of the throat, perhaps by some disturbance, it becomes unstable and pops out ahead of the cowl lip, i.e. unstart condition as in figure 38. When unstart happens, the pressure recovery drops and flow is spilled over the cowl, producing high drag and aircraft control problems, expecially with high internal compression).

The diverse capabilities of the multi-mission military fighter require a wide range of propulsion system thrust levels to provide excess energy needed for combat maneuvers, in addition to low fuel consumption at reduced thrust levels for cruise operation. As a consequence of the broad flight scenario, including loiter, transonic combat and supersonic intercept, the design of the propulsion system has to be highly integrated into the total system. Operational requirements define several thrust sizing points in addition to more mission profiles, determining the basic engine characteristics.
Now, a multi-cycle engine like the turbo-ramjet may have the combined requirement of high performance and mission flexibility. At high Mach number, the turbo-ramjet may deliver about three times the internal (figure 37) afterburning turbojet. Ramjet combustion chamber is generally less sensitive to flow distortion. At take-off, an engine like the one in figure 2 is operated as a turbofan, with possible reheat on both flows. At supersonic or subsonic Mach number, depending on the altitude, the engine may go in the mixed flow mode, by that gaining in thrust. When approaching the Mach limit, the progressively closing of the external flow determines the r.p.m. slowing down to idle, starting the pure ramjet operation.
Of course, trade-off studies have been performed on an overall propulsion system basis with proper consideration of inlet/engine airflow matching, afterbody/exhaust nozzle interactions, environmental conditions, bleed and power extraction, and design parameters (such as bypass ratio, turbine temperature and pressure ratio) effecting the overall performance.
Since inlet capture area was sized at the maximum turbojet flight speed (M = 2.5), the inlet flow is in excess of engine requirements for all lower speeds and low altitudes. Combining engine and inlet characteristics for maximum installed thrust results in significant losses on net thrust and specific fuel consumption. The thrust-to-weight ratio is approaching 1.0. Subsonic cruise occurs at low levels of drag, due to improved aerodynamic efficiency L/D. These two characteristics require the engines to operate at low thrust settings and low airflows, resulting in high spillage drag due to the mismatch of engine/inlet flow.
In this combining the thrust sizing and range requirement, for the purpose of high levels of performance, the aircraft has engines oversized for optimum range.

The tactical mission defined in figure 39, showing the flight Mach number as function of the altitude according the baseline scenario in figure 35, was chosen as basic for computation of the engine/aircraft performances summarized on figure 40.
In figure 39, the descent with transonic acceleration occurs between the subsonic cruise (out) and the low supersonic dash, not regarding to limitations imposed by sonic boom overpressures reaching the ground. Another descent with transonic deceleration occurs between the dash (return) and the cruise (return) phases, before loiter.
Specific fuel consumption q and thrust F values along such typical flight profile were established through appropriate flight efficiencies $\frac{M}{q}$ L/D, aircraft drag and thrust coefficients C_D, C_T, figure 41, altitude, amount of reheat, and previous considerations.

Considering the aerodynamic drag expressed as

$$D = \frac{1}{2}\, \rho\, V^2 \cdot S \cdot C_D$$

where S is the wing plan form area, the aircraft maximum level speeds (at sea level and at height) and ceiling are obtained when specific excess power (S.E.P.)

$$S.E.P. = (F - D)\, \frac{V}{W} \qquad (1)$$

is equal zero (V and W corresponding, respectively, to flight speed and weight).
This means that, imposing F = D, for instance to the fighter Dassault Mirage III-S, we have for the maximum level speed at S/L (Mach 1.14)

$$F_o = 13{,}670 \text{ lb} = D = \frac{1}{2}\,\frac{0.07651}{32.2}\,(1266.4)^2 \cdot 375\ C_D$$

i.e

$$C_D \approx 0.02$$

and, for the maximum level speed M = 2.0 at 39,375 ft, where $C_D = 0.015$

$$D = \frac{1}{2}\,\frac{0.019}{32.2}\,(2{,}130)^2 \cdot 375 \cdot 0.015 = F = 7{,}529 \text{ lb}$$

S.E.P. is therefore regarded as one misure of combat capability, i.e. an excess of thrust for maneuvering. One value of it is also specified for steady rate of climb. In both cases, the aircraft is accelerating, as in general in figure 25. In other equilibrium (F = D) flight conditions, as at cruising speed, thrust is reduced through variable geometry of inlet, nozzle or compressor vanes. Thrust may be then expressed as

$$F = \frac{1}{2}\,\rho\,V^2 \cdot A_e \cdot C_F \qquad (2)$$

being A_e and C_F (figures 25 and 41) the engine exhaust cross sectional area and thrust coefficient. From the values of the maximum level speeds at S/L, we carry out, respectively for the fighters F-111A, F-14A, Mig 21, Mig 23, aicraft drag coefficients 0.035, 0.03, 0.035, 0.02.

In our case, with a wing gross area of 576 ft^2 and a drag coefficient $C_D = 0.025$, we have a maximum level speed at S/L dash of 1,675 ft/sec (M = 1.5), a maximum level speed at 65,000 ft dash of 3,388 ft/sec (M = 3.5), respectively ramjet, and the other cruise level speeds considered in the mission of figure 35, as shown in Table 2. The values of C_D are connected to the drag polars (C_L, C_D) and the aerodynamic efficiencies for operation at subsonic and supersonic speeds, figures 42 and 43.

Table 2

Altitude, ft	S/L	65,000	36,000	12,000
Maximum level speed, ft/sec	1,675	3,388	--	--
Level speed, ft/sec	--	--	871	906
Mach number (maximum)	1.5	3.5	--	--
Mach number	--	--	0.9	0.85
C_D	0.025	0.02	0.03	0.028
Max. Thrust (2 engines), lb	48,000	11,648	--	--
Reduced Thrust (2 eng.), lb	--	--	4,610	10,900

These cruise and dash phases are governed by the Breguet range equation, approximated in the form

$$R \approx \frac{V}{q} \cdot \frac{L}{D}\ \ln \frac{W_2}{W_1} \qquad (3)$$

expressing range R as function of weight W.

Climb acceleration, dive transonic acceleration and deceleration, and combat sorties are made up from individual elements of specific excess power (1) and corresponding range, according to

$$\frac{dW}{W} = \left(\frac{dh}{V} \pm \frac{1}{g}\,dV\right) q \qquad (4)$$

$$q' = q\left(\frac{1}{1 - S \cdot C_D / A_e\ C_F}\right) \qquad (5)$$

Equation (4) gives the fraction of weight decreasing due to fuel consumption for uniform climbing, in the hypothesis of F/W ≈ 1, and for acceleration on the same trajectory; climb should preferably take place at the highest velocity at a given altitude. Specific fuel consumption q' in eq. 6 refers to the excess thrust, compared to q as carried out from local engine operation, aerodynamic efficiency, figure 42, and flight efficiency, figure 1.

Ramjet - The capture area is sized for a flight Mach number of 2.5, at which the inlet is yet correctly operating. To provide an uniform intake flow field at high angles of attack and supersonic speed, the inlet was selected with horizontal ramps (three compression ramps and a diffuser ramp, with the first ramp rotating), to have three oblique and one normal shocks ahead of the cowl. (Mixed-compression is occurring, through variable geometry, at higher Mach numbers).

In two-dimensional oblique and normal shock approximation for perfect gas, starting from ambient conditions $M_o = 2.5$, h = 45,000 ft, $p_o = 2.141$ psia, $T_o = 392.7°R$, $V_o = 2{,}420$ ft/sec, $\rho_o = 0.01474$ lb/ft³, we obtain, after shock and dynamic compression, respectively, $M_1 = 0.963$, $p_1/p_o = 5.82$, $\rho_1/\rho_o = 3.04$, $T_1/T_o = 1.916$, $V_1 = 1{,}297$ ft/sec, and $p_2 = 15.35$ psia, $V_2 = 328$ ft/sec, $T_2 = 797.5°R$, $\rho_2 = 0.059$ lb/ft³. Now, free turbulent mixing (between hydrogen jets and air stream) and combustion occur in the chamber, insofar that a prefixed maximum temperature of 4,500°R is reached. The stoichiometric reaction for combustion of air with hydrogen fuel is

$$1/2\ O_2 + H_2 \longrightarrow H_2O$$

This gives a stoichiometric fuel/air ratio by weight of $f_{st} = 0.029156$.

Using an equivalent ratio 1.0, such that $f = f_{st}$, we have as consequence of combustion, a temperature rise

$$\Delta T = f/(f + 1) \cdot \varepsilon Q/C_p = 3{,}705.5 \tag{6}$$

where Q = 51,590 BTU/lb, $\varepsilon = 0.96$ and $C_p = 0.378$ BTU/lb°R are hydrogen heat of combustion, combustion efficiency and specific heat at constant pressure. The C_p value is carried out by interpolation from the tables of Ref. 8, assuming that gas composition remains constant within the combustion chamber (frozen flow); where density and temperature result $\rho_{5'} = 0.00727$ lb/ft³ and $T_{5'} = T_2 + \Delta T = 4{,}500°R$. Assuming chemical equilibrium at very point of the expansion, as an upper limit to the obtainable performance, and with K = 1.269, the theoretical nozzle exhaust velocity and density become, figure 44,

$$V_{7'} = \sqrt{2g\ H_{5'}(1 - (p_0/p_2)^{(K-1)/K})} = 6{,}157 \text{ ft/sec} \tag{7}$$

and $\rho_{7'} = 0.00154$ lb/ft³, (H = entalpy).

A thrust $F = D = 1/2 \cdot \rho\ V_o^2\ S\ C_D = 21{,}618$ lb is necessary to obtain a maximum level flight $V_o = 2{,}420$ ft/sec at h = 45,000 ft.

The corresponding gas and air mass flows (with a correction factor 0.9), exhaust and capture cross sectional areas, specific fuel consumption, and overall propulsion efficiency (propulsive and thermal) are:

$$\dot{m}_a = F/|(1 + f)\ V_{7'} - V_0|0.9 = 6{,}133 \text{ lb sec/ft} \tag{8}$$

$$\dot{m}_g = \dot{m}_a(1 + f) = 6.312 \text{ lb sec/ft} \tag{9}$$

$$A_g = \dot{m}_g\ g/\rho_{7'}\ V_{7'} = 21.43 \text{ ft}^2 \tag{10}$$

$$A_a = \dot{m}_a\ g/\rho_0\ V_0 = 5.53 \text{ ft}^2 \tag{11}$$

$$q = (\dot{m}_g - \dot{m}_a)\ g\ 3600/F = 0.959 \text{ lb/lb/hr} \tag{12}$$

$$\eta = \frac{|V_{7'}(1 + f)/f - V_0/f|\ V_0/g}{778\ Q + V_0^2/2g} = 0.251 \tag{13}$$

For a flight Mach number of 3.5 (maximum) at h = 65,000 ft, we get a thrust F = 11,648 lb, through small area changes. Specific fuel consumption results still too high; even though a compression ratio $p_1/p_0 = 11.573$ is obtained behind inclined mixed shock waves, and a further dynamic compression happens in a diffuser behind throat area. A comparison with performances obtainable using JP fuel is shown in Table 3. JP fuel becomes probably unpracticable from the point of view of weight increase in a ramjet engine for a fighter aircraft, and hydrogen probably requires too much volume in the aircraft.

Gross thrust F, specific fuel consumption q and fuel-air ratio f, are largely changing with flight Mach number and altitude, in a ramjet engine designed for h = 45,000 ft, $M_o = 2.5$, and $T_{5'} = 4{,}500°R$. So, in our study, where gas mixtures in chemical equilibrium and unvariable temperature $T_{5'}$ are considered, gross thrust at h = 45,000 ft is reduced about 50% at h = 65,000 ft and M = 3.5, while specific fuel consumption is practically unchanged. A proficuous extention of flight speed field is made possible by the variable geometry inlet and exhaust nozzle, imposed by a fixed capture area. At given altitude and Mach number, a

Table 3

RAMJET	H_2	JP fuel	H_2	JP fuel
M_0	2.5	2.5	3.5	3.5
F (lb)	21,618	21,618	11,648	11,648
h (ft)	45,000	45,000	65,000	65,000
p_2 (psia)	15.35	15.35	12.756	12.756
T_2 (°R)	797.5	797.5	935	935
Equiv. ratio	1.0	1.1	1.0	1.08
f	0.02916	0.07439	0.02916	0.073
$T_{5'}$ (°R) max	4,500	4,500	4,500	4,500
$V_{7'}$ (ft/sec)	6,157	5,760	7,023	5,998
$\dot{m}_a$ (lb sec/ft)	6.133	6.373	3.370	4.246
$\dot{m}_g$ (lb sec/ft)	6.312	6.847	3.469	4.556
A_g (ft^2)	21.43	21.56	23.68	36.04
A_a (ft^2)	5.53	5.73	5.65	7.12
q (lb/lb/hr)	0.959	2.54	0.978	3.08
η	0.251	0.261	0.344	0.303

desired thrust value would require an optimum temperature $T_{5'}$ without geometry changing. Hydrogen/air mixtures give about 20% less thrust than JP fuel/air, using tha same engine size and combustion temperature. However, it is possible to think about a future employment of hydrogen in ramjet powered fighters as a result of a still acceptable specific fuel consumption, whereas JP fuel may be prohibitive because of high initial fuel weight.

Turbojet - Approaching the mach limit 2.5, the turbojet r.p.m. is slowing down to start the pure ramjet operation. In such matching condition, it is necessary to increase, by afterburning, the dry turbojet thrust. Thus, performance characteristics at M_o = 2.5 are carried out, as maximum level speed limit at h = 45,000 ft made possible by turbojet with afterburning, figure 45.
Airflow is entering an axial compressor after inlet shock and dynamic compression, through which: $p_2 = 15.35$ psia, $\rho_2 = 0.057$ lb/ft^3, $T_2 = 797.5$°R and $V_2 = 328$ ft/sec. It is necessary a compression ratio $p_3/p_2 = 7$ to obtain, with an hydrogen equivalence ratio 0.2 (i.e. leanest fuel/air mixture f' = 0.005832), a maximum temperature T_4=2,500°R in combustion chamber. In fact, (with $T_{3'} = T_2\ (7)^{K-1/K} = 1{,}390$°R, $p_3 = 7\,p_2 = 107.45$ psia, $T_3 = (T_{3'} - T_1)/\eta_c + T_2 = 697 + T_2 = 1{,}495$°R, $\rho_3 = \rho_2\ (7)^{1/K} = 0.229$ lb/ft^3, $p_4 = 0.95\ p_3 = 102.08$ psia), Eq. (6) gives, with $C_p = 0.285$

$$\Delta T = 1{,}005 \qquad , \qquad T_4 = T_2 + \Delta T = 2{,}500°R$$

Moreover, from the work balance

$$C_{pa}\ (T_{3'} - T_2)/\eta_c = C_{pg} \cdot \eta_g (1 + f')\ \Delta T' \tag{14}$$

$$(C_{pa} = 0.24, \quad C_{pg} = 0.30, \quad \eta_c = 0.85, \quad \eta_g = 0.9)$$

we obtain

$$\Delta T' = 342, \quad T_5 = T_4 - \Delta T' = 1{,}881°R, \quad p_5 = p_4/(T_4/T_5)^{K-1/K} = 31.15 \text{ psia}$$

Through afterburning, we get the maximum temperature $T_6 = 4{,}500$°R. In fact, with $f'' = f_{st} \cdot 0.45 = 0.013122$, Eq. (6) gives ($C_p = 0.318$)

$$\Delta T'' = 2{,}000 \qquad , \qquad T_6 = T_4 + \Delta T'' = 4{,}500°R$$

Eq. (7) gives the exhaust velocity V_7= 6,330 ft/sec, and Eqs. (8) to (13) provide, with $f = f' + f''$, $\dot{m}_a$ = 5.981 lb sec/ft, $\dot{m}_g$ = 6.097 lb sec/ft, A_g = 16.58 ft^2, A_a = 5.4 ft^2, q = 0.626 lb/lb/hr, η = 0.412. Without afterburning, figure 46, the expansion in the exhaust nozzle should start from the condition p_5 = 31.15 psia, T_5 = 1,881°R, ρ_5 = 0.0385 lb/ft^3, to reach an exhaust velocity V_7 = 4,206 ft/sec, from which specific fuel consumption and total propulsion efficiency become q = 0.415 lb/lb/hr and η = 0.58. Engine airflow and inlet cross-sectional area require to be, however, two times more than the one with afterburning. In Table 4, performances of dry turbojet and turbojet with afterburning are compared, taking into account hydrogen and JP-fuel.

Dry turbojet operation during subsonic cruise speed (M = 0.85) at h = 36,000 ft altitude is shown in Table

4, both for hydrogen and JP-fuel. A compression ratio of 20 : 1 is made possible by an eight stage axial compressor (stage pressure ratio from 1.82 to 1.24, and blade tip velocity of 1,500 ft/sec); temperature in the combustion chamber is considered to be 1,935°K. High compression ratio and temperature allow quite low specific fuel consumption, 0.345 lb/lb/hr with hydrogen and 1.192 lb/lb/hr with JP-fuel.
It is however possible to obtain a better specific fuel consumption. For instance: the Rolls-Royce RB.211 turbofan, 3-shaft, by-pass ratio 5.04 : 1, has an overall pressure ratio of 26 : 1, a turbine inlet temperature of 1,535°K and a specific fuel consumption of 0.6[illegible] hr; the General Electric CF.6 turbofan, two-

Table 4

	TURBOJET AFTERBURNING		DRY TURBOJET		DRY TURBOJET	
	H_2	JP fuel	H_2	JP fuel	H_2	JP fuel
M_0	2.5	2.5	2.5	2.5	0.85	0.85
F (lb)	21,618	21,618	21,618	21,618	3,842	3,842
h (ft)	45,000	45,000	45,000	45,000	36,000	36,000
p_2 (psia)	15.35	15.35	15.35	15.35	5.105	5.105
T_2 (°R)	797.5	797.5	797.5	797.5	445	445
p_3 (psia)	107.45	107.45	107.45	107.45	102.1	102.1
T_3 (°R)	1,495	1,495	1,495	1,495	1,153	1,153
Equiv. ratio	0.2	0.25	0.2	0.25	0.4	0.6
f'	0.00583	0.0169	0.00583	0.0169	0.0117	0.0406
T_4 (°R) max	2,500	2,500	2,500	2,500	3,485	3,411
p_4 (psia)	102.08	102.08	102.08	102.08	97.00	97.00
T_5 (°R)	1,881	1,881	1,881	1,881	2,863	2,717
p_5 (psia)	31.15	31,15	31,15	31,15	41.27	34.74
Equiv. ratio	0.45	0.53	- -	- -	- -	- -
f''	0.01312	0.0358	- -	- -	- -	- -
T_6 (°R) max	4,500	4,500	- -	- -	- -	- -
p_6 (psia)	31.15	31,15	- -	- -	- -	- -
V_7 (ft/sec)	6,330	5,276	4,206	4,055	4,912	4,811
f	0.01895	0.0527	0.00583	0.0169	0.0117	0.0406
$\dot{m}_a$ (lb sec/ft)	5.981	7.666	13.271	14.102	0.982	0.974
$\dot{m}_g$ (lb sec/ft)	6.097	8.070	13.348	14.340	0.994	1.013
A_g (ft^2)	16.58	26.33	20.07	20.87	1.39	1.13
A_a (ft^2)	5.40	6.92	11.91	12.71	2.24	2.22
q (lb/lb/hr)	0.626	2.147	0.415	1.278	0.345	1.192
η	0.412	0.31	0.580	0.526	0.18	0.146

shaft, has an overall pressure ratio of 30 : 1, and an estimated specific fuel consumption of 0.6 lb/lb/hr.

Turbojet - ramjet matching - The take-off program of the aircraft corresponding to the mission scenario in figure 35 requires uniform acceleration to $M_0 = 0.85$ during climb from sea level to 16,000 ft, followed by a constant M climb to 36,000 ft, where a subsonic cruising capability is prescribed. After a dive transonic acceleration to low altitude, followed by low supersonic flight and maneuvers, it is prevised a fast acceleration, by turbojets with afterburning, to an high supersonic dash at M = 3.5 by ramjets.
Now, from the compressor performance map of a typical turbojet engine, figure 47, it is possible to understand the main reason for chosing a turbo-ramjet combination also for the present application. The map shows the dependence of compressor total pressure ratio p_{3t}/p_{2t} on the referred air mass flow $\dot{m}_a \sqrt{T_{2t}}/p_{2t}$ for constant values of the referred rotational speed $N/\sqrt{T_{2t}}$. At subsonic cruise conditions (M = 0.85, h = 36,000 ft) the compressor operates near peak pressure ratio and maximum rotational speed N. In the figure, this operating point is indicated by A. As the ram effect at the air intake increases in magnitude with increasing flight speeds, the referred rotational speed diminishes, reducing thereby the compressor pressure ratio as illustrated by point B which represents supersonic dash (M = 3.5, h = 65,000 ft). From the point of view of the maximum cycle temperature, the decrease in compressor pressure ratio is desiderable because of the large temperature increase occurring in the inlet diffuser. But, due to the decrease of the referred speed, there is a simultaneous reduction in the rate air flow that the compressor can swallow compared to supersonic conditions. On the other hand, the inlet is capable of handling at B an

air flow which exceeds the mass flow at A. This higher rate of flow is desiderable for high thrust. It can be realized with the engine operating in the ramjet mode with the flow-by-passing the turbojet section of the engine.
A matching device between the free stream upstream of the engine and the flow entering the engine is the inlet tube, in as much as it channels the proper amount of air to the entrance of the engine entrance section. The quality of an inlet is measured at a given rate of flow and free stream Mach number by the internal energy losses encountered by the flow passing through it and by the external drag force due to the intake. Expecially at high supersonic flight speeds, inlets of variable geometry are required to handle the air needed by the engine with acceptable efficiency.
The design of the inlet to the airframe-engine combination consists of selecting the inlet configuration required by the mission profile, figure 35, and of equipping it with variable geometry and other features that are required to make it operate with acceptable efficiencies over the entire range of flight conditions. Slight deviations from the optimum shock configuration can result in a significant increase in internal pressure losses due to shock-boundary layer interactions and separations and in an appreciable increase in inlet pre-entry drag due to spillage. At $M_0 = 3.5$, even the optimum shock configuration produces usually a total pressure ratio of less than 0.80. Modification and separation of the shock pattern can be minimized with the aid of boundary layer bleed-off through flush scoops, figures 15 and 16, preventing the low energy boundary layer air from entering the inlet. With the aid of a series of vortex generators, it may be alleviated separation of the flow in the subsonic diffuser (with geometry limited by weight and length considerations) downstream of the inlet passage throat. By-pass doors prevents the inlet shock system from being expelled when the inlet idles.
Along the operating line of the compressor, largely effected by the inlet characteristics, matched conditions exits at the interfaces of all engine components. An operating line may be dangerously close to stall during transonic flight speeds. This would require an appropriate modification of the inlet throat area. For multiple engine installations, the articulated, two-dimensional inlet geometry, figure 15, using a movable ramp to vary the throat area, may be more suitable than axisymmetric inlet, aside from higher aerodynamic losses, Ref. (9).
The inlet of a turbo-ramjet must satisfy a great variety of requirements, through a sophisticated variable geometry involving complicated assemblies of winged ramps, bleed slots, dump doors, actuators, etc.. The increasing to $M = 2.5$ for the Mach number limit at the compressor face represents the way of easing the inlet problems of the turbo ramjet, in as much as, at higher Mach number, the amount of deceleration in the diffuser would be reduced, limiting the rise in static pressure, temperature and losses.
As flight conditions change from sea level static to $M_0 = 3.5$ in the stratosphere, the referred r.p.m. $N/\sqrt{T_{2T}}$ is quite decreased, point B in figure 47. This change in operating conditions presents difficulties in matching of the stages of the compressor tending to force the low pressure stages toward stall and the high pressure stages toward choking. Thus, the equilibrium operating line should be located in the region of highest compressor efficiency, as possible near the surge line with a minimum stall margin.
Variable stator vanes in the front and rear of the compressor are effective measures to meet the extreme demands of low and high Mach number flights, particularly to tune the compressor to the inlet conditions, and to move the operating points of individual compressor stages away from the surge line. The problem is much less complicated by the use of a compressor having two or three mechanically independent rotors.
Matching of the combustion system with the rest of the engine is very complicated in the case of an engine designed for high speed flight with its variable geometry and wide range of operating conditions, involving a wide field of temperature. Severe spatial limitations are particularly imposed when hydrogen is used instead of JP-fuel.
The matching of the turbine to the combustion chamber is required to maintain an optimum radial temperature distribution of the gas, over the broad entire operating range of this kind of engine.
The largest effect on turbo-ramjet performance is due to the jet nozzle. In fact, at subsonic flight speed, the nozzle exhaust pressure ratio lies in the range of 7 : 1, increasing 4 ÷ 5 times with flight Mach number when the engine operates in the ramjet mode, and the discharge to throat area ratios vary many times.

Preliminary design data - According to the turbo-ramjet engine design data contained in the Tables 3 and 4 and the mission scenario of figure 35, we carry out the total fuel weight needed for the entire mission of the twin-engined long range attack fighter.
As a maximum thrust at disposal for S/L emergency and low altitude combat maneuvers, it is considered that one equivalent to the aircraft sea level drag for an uniform flight speed corresponding to $M_0 = 1.5$, well over the need for climb and transonic acceleration (Eq. 4), i.e. $F = D = 0.07651\ (1{,}673)^2 \cdot 576 \cdot 0.025/2 \cdot 32.2 = 48{,}000$ lb. However, the dash phase at $M_0 = 1.5$ during the mission occurs at higher altitude with a thrust $F = 26{,}000$ lb (corresponding to $M_0 = 1.5$ at $h = 36{,}000$ ft).
With a maximum thrust (at disposal)/maximum take-off weight ratio equal to 0.755, as the Mikoyan MiG 25 fighter (0.52 the Dassault Mirage III, 0.5 the F-14 A, 0.58 the F-4D, 0.4 the F-5A, 0.62 the MiG 23), the maximum take-off weight results 63,575 lb (maximum operating take-off weight 40,000 lb).
Take-off, with an average acceleration to h = 16,000 ft, and climb, without acceleration to h = 36,000 ft, require 802 sec (65 nm) with an average thrust of 18,093 lb. Considering average thrust and specific fuel consumptions during all the mission phases (climb 65 nm, 802 sec; cruise (out) 300 nm, 2222 sec; dash at low altitude 135 nm, 490 sec; climb 50 nm, 132 sec; dash (return) 200 nm, 359 sec; cruise (return) 150 nm, 1048 sec; loiter 110 nm, 1200 sec), the total fuel weight at take-off should result 5,440 lb = 1,227 ft^3, using hydrogen, and 19,120 lb = 405 ft^3 with JP-fuel, obtained by adding together the phase fuel weights $q \cdot F \cdot sec/3600$.
The fuel weight for such long range fighter should be prohibitive for both hydrogen and JP-fuel. Taking

into account the fact that specific fuel consumptions as previously computed, Table 3 and 4, are largely in excess to the ones corresponding to engines with higher compression ratio, it should be possible to carry the fuel weights and volumes.

As comparison term we take Phantom II-4 N twin-engined long range attack fighter, quoted for the following characteristics: power plant 34,000 lb (two turbojets with afterburning), maximum take-off weight 54,600 lb, maximum take-off weight (clean) 46,000 lb, maximum level speed with external stores $M_0 = 2.0$, combat radius 781 nm (interceptor) and 868 nm (ground attack), wing gross area 530 ft^2, maximum wing loading 103 lb/ft^2 (max T.O. weight) and 87 lb/ft^2 (max T.O. weight, clean), total fuel capacity (tankage in wings and fuselage) 267 ft^3 plus eventual (one external tank under fuselage and two underwing tanks) 130 ft^3 capacity corresponding to 18,735 lb (12,600 + 6,135).

In our design, we get a maximum wing loading of 63,575/576 = 110 lb/ft^2(normal for fighter aircraft) and a fuel weight (volume) of 19,120 lb (405 ft^3) for JP-fuel and of 5,440 lb (1,227 ft^3) for hydrogen. This is acceptable for JP-fuel, in consideration of the possibility to use engines with less specific fuel consumption and of the heavier and more powerful aircraft. From the other hand, the hydrogen fueled aircraft may use supplementary external wing tanks, figure 33, to take hydrogen needed to reach the combat area, corresponding to climb and cruise (out), i.e. 2,292 lb = 517 ft^3. It is possible to carry in normal tankage the remaining hydrogen volume 1,227 - 517 = 710 ft^3; actually reducible with specific fuel consumptions better than those in Tables 3 and 4.

<u>High supersonic transport aircraft</u> - The schematic outline of the turbo-ramjet chosen for the present computation is shown schematically in figure 48. The complex propulsion cycle has several more independent variables than have faced the control system even before. The number of variables which the control system must coordinate is up to eleven for the engine and another five for the inlet. For maximum performance it is imperative that all of these variables be coordinated to maintain engine performance for steady-state operation, stable and safe operation for transient, as well as to provide the fastest possible engine response to power change commands. The variables which must be coordinated include fan geometry, compressor geometry, gas generator and ramjet fuel flows, turbine geometry, core-stream exhaust nozzle, and a variable divergent nozzle behind the convergent nozzle. Going back up front to the inlet, one sees a system of three ramps and bypassing door arrangements.
As in the previous computations, the total fuel weight needed for the mission scenario in figure 36 is carried out on the basis of specific fuel consumptions, average thrusts and flight times, relative to each phase, taking into account only hydrogen as fuel of an improved and uptodate turbo-ramjet.
After take-off, the aircraft accelerates and reaches transonic and supersonic speeds at altitudes higher than those corresponding to minimum specific fuel consumptions. Climb and acceleration schedule takes into account limitations imposed from sonic boom, reaching the prescribed distance of 385 nm with 20% and 42% more fuel consumption and climb time, in respect to the military aircraft, in order to reduce sonic boom ground overpressure to a value between 1.25 and 1.55 lb/ft^2. Following a supersonic climb, at the end of which operation is changing from turbojet to ramjet engine, supersonic cruise at $M_0 = 4.5$ and h = = 70,000 ft covers the most part of the 4,200 nm range. Descent and transonic deceleration, before approach and landing, follow a schedule imposed by sonic boom limitations and structural resistance. In the mission profile of figure 36, the transitory climb and descent phases to/from h = 50,000 ft (sonic boom altitude) require, respectively, 17 and 23 minutes over an horizontal distance of 160 and 135 nm. Cruising speed at 70,000 ft altitude is covering a range of 3,700 nm. Speed and time for such cruise range are V = 4,357 ft/sec and t = 22,481,627 ft/4,357 = 5,160 sec.

To a supersonic transport, in uniform flight at $M_0 = 4.5$ and h = 70,000 ft, with a wing area S = 7,580 ft^2, it corresponds an overall air drag $D = \frac{1}{2}\rho V^2 \cdot S \cdot C_D = 195{,}285$ lb. This value is also that one of the required thrust F. Therefore, fuel consumption for the cruise range results $q \cdot D \cdot 5{,}160/3{,}600 = 125{,}960$ lb. As specific fuel (hydrogen) consumption, it has been carried out, following the previous method of computing, a value q = 0.45 lb/lb/hr.
Similary, taking into account all the climb, descent, acceleration and deceleration, phases with their appropriate average specific fuel consumption, we need at most another hydrogen charge of 37,117 lb, to which we add a fuel reserve of 31,500 lb.

Total fuel weight and volume for the entire mission should be 194,577 lb and 43,903 lb. The largest amount of hydrogen, 64.8% is utilized in the cruising phase of flight which is conducted at a Mach number of 4.5 in this case. Take-off, climb, acceleration, descend and land, account for 19%, and fuel reserves are 16.2% of the total fuel load. The fuel required for cruise and for take-off, climb, and acceleration, is a function of the flight efficiency $M/q \cdot L/D$ of the aircraft. Also the reserve fuel is governed by the flight efficiency and by rules concerning aircraft holding in bad weather and diversion to an alternate airport. It is, however, still possible to improve the flight efficiency for reducing the fuel required.

Takeoff gross weight and payload weight result (with a fuel percentage 45% and a payload percentage 12.5%) 432,390 lb and 54,050 lb (258 passengers), respectively. Aerodynamic efficiency and maximum wing loading are L/D = 8.5 and 432,390/7,580 = 57 lb/ft^2.

Engine thrust, required for an overall design thrust 50% more than that one corresponding to cruise speed at h = 70,000 ft, becomes 1.5·195,285 = 292,928 lb, i.e 73,230 lb thrust per engine (SLS), with a thrust-to- weight ratio equal to 0.68.
The major problem to solve for an actual design of hydrogen fueled high supersonic aircraft remains that

one of the excessive fuel volume, to be reduced through improvement of specific fuel consumption. A solution like the one indicated in figure 34 (tankage in fuselage) should be possible reducing about 25% the fuel weight needed for such mission range.

HYDROGEN FUELED TURBO-ROCKETS/AIRCRAFT

The mission of a turbo-rocket, figure 49, engined aircraft, figure 50, could be the one in which the payload is a sounding rocket, a ballistic missile, or a satellite launcher. Such composite launch system makes possible satellite or rocket programs directly from normal airports, without necessity of complex ground facilities but with choice of initial launch positions and orbit slopes.

A satisfactory launch speed could be reached by a turbo-rocket propelled aircraft, for instance at an altitude h = 70,000 ft. Such kind of engine is enough flexible to meet a large variety of initial conditions with speed around $M_0 = 4.5$. Like turbo-ramjet, this multi-cycle engine is arranged with a lot of inlet and exhaust variable geometries, with hydrogen-oxygen rocket combustion chamber both for acceleration/deceleration phases and cruise, and with hydrogen afterburning system for thrust increasing. Interfaces between various engine components are combined in a specific turbo-rocket, introducing complex matching problems.
It is possible to power, with an acceptable oxygen and hydrogen tankage, both for rocket and afterburning combustions, an aircraft able to follow a high altitude laungh mission with very high payload and gross weight.

It has been here chosen, as an example, a high speed transport plane with range corresponding to a flight of 20 minute at $M_0 = 4.5$ and altitude h = 70,000 ft. The power plant is a combination of a fanjet and a rocket. High supersonic flight speeds greatly accentuate the structural and gas dynamics problems encountered by the fan and its rocket propelled turbine. Inlet and engine are assembled conveniently in an airframe and subjected to maneuvers, thrust changes, and atmospheric disturbances.

A thrust F = 36,500 lb for each turbo-rocket, figure 49, has been chosen, largely depending on matching of the air intake performance to engine requirements and on total pressure recovery, bleed boundary layer, and cowl drag delivered by the air intake system.
The aerodynamic thrust against the fan is given by, figure 51

$$p_2\ A_2(1 + KM_2^2/2) - |p_0\ \varepsilon\ A_1(1 + KM_0^2/2) + 1/2 \cdot K\ p_0\ M_0^2\ A_1\ C_x| - 1/2 \cdot KM_0^2\ p_0\ A_2\ C_y \tag{15}$$

where, the first term represents the force due to the compressed air at the end of the diffuser (being $p(1 + KM_2^2/2)$ the stagnation pressure), the second and third terms the capture and external drag. Mass flow and internal/external capture coefficients are indicated, respectively, with $\varepsilon = A_0/A_1$, C_x and C_y. Mass flow is equal to

$$\dot{m}_a = \varepsilon\ \rho_0\ V_0\ A_1/g = \varepsilon\ K\ p_0\ A_1/a_0 \tag{16}$$

with $a_0 = \sqrt{K\ g\ p_0/\rho_0}$ corresponding to sound velocity.
We define an inlet efficiency (as pressure recovery coefficient)

$$\eta_0 = p_{2t}/p_{0t} \tag{17}$$

to be intended as ratio between isoentropic stagnation pressures, corresponding, Ref. 10, to 0.20 ÷ 0.25 for $M_0 = 4.5$ and fixed geometry cone wedge angle of 25° ÷ 30° with all external compression. Better pressure recovery coefficients may be obtained with variable geometry inlet during flight, to change the area contraction ratio with flight Mach number and airplane angle of attack. Inlet compresses supersonic flow by changing its direction. The centerbody turns the flow outward, and then the cowl catches it and turns suddenly it back toward the engine, generating a shock wave. The cowl angle determines the strength of this shock, which affects the total pressure that is recovered from the free stream.

Considering in our case three oblique and one normal shock, with a small percentage of internal compression, it is possible to attain in practice for $M_0 = 4.5$ an actual value $p_{2t} = 90\ p_0 \cong p_2$ (with negligible air speed in front of the fan). This means

$$\eta_0\ 90/p_{0t} = 90/p_0(1 + (K - 1)\ M_0^2/2)^{K/K-1} = (K = 1,4) = 90/289 = 0.31$$

Ambient conditions at an altitude h = 70,000 ft correspond to $\rho_0 = 0.0041$ lb/ft³, $p_0 = 0.660$ psia, $T_0 =$ $= 392.7$ °R. Now, because $p_{2t} = 90\ p_0 = \eta_0\ p_{0t} = p_0(1 + 0.65(K - 1)\ M_0^2/2)^{K/K-1} = p_0(1 + (n - 1)M_0^2/2)^{n/n-1} =$ (with n = 1.56) $\cong p_2$, we obtain, as an approximation, for the air conditions in front of the fan

$$T_2 \cong T_{2t} = T_0(1 + 0.65(K - 1)\ M_0^2/2) = 1443\ °R$$

$$\rho_2 = \rho_0(p_2/p_0)^{1/n} = 0.073\ \text{lb/ft}^3 \quad , \quad p_2 = 59.4\ \text{psia}$$

A two-stage high pressure ratio fan is chosen to increase appropriately (during acceleration phases) air pressure and temperature behind the diffuser. A tip speed of 1,450 feet/sec is selected for the fan and the overall pressure rate set at 2.8, with individual stage pressure ratios of 1.74 and 1.61. Peak efficiency at design speed is 85.5%. High compression ratio and efficiency are obtainable through appropriate rotor speed and blade loading, with blade profiles and flow path geometries to minimize shock losses and flow separation in mixed supersonic/subsonic flow.

Pressure, temperature and density behind the fan become

$$p_3 = 2.8\ p_2 = 166.32 \text{ psia} \quad , \quad T_{3'} = 1{,}936.5\ ^\circ R$$

$$T_3 = 2020\ ^\circ R \quad , \quad \rho_3 = 0.152 \text{ lb/ft}^3$$

The powers needed from the turbine and absorbed from the fan are the same, according to the following thermal balance

$$\dot{m}_a C_{pa}(T_3 - T_2)/\eta_m = \dot{m}_g \cdot C_{pg} \cdot \Delta T_g \cdot \eta_t \tag{18}$$

where $\dot{m}_g$, ΔT, C_{pg}, ΔT_g, η_t and η_m, are, respectively: hydrogen oxygen mass flow, ideal temperature decrease, average specific heat, of the hydrogen/oxygen gas mixture operating in the turbine; turbine efficiency, and compressor mechanical efficiency.

The air from the compressor has going through an expansion in order to mix conveniently with the gas flowing from the rocket power turbine. Correspondent air temperature and pressure are

$$T_{5a} = T_3 - (V_{5a}^2 - V_3^2)/2g \cdot C_{pa}\ 778 \quad , \quad p_{5a} = p_3\,(T_{5a}/T_3)^{K/K-1} = p_{5g} \tag{19}$$

being V_3, V_{5a} and p_{5g}, air velocity downstream the compressor and after the expansion, and pressure at the turbine exit.

The gas flow $\dot{m}_g$ has undergoing an expansion in the turbine, from which we carry out

$$T_{4g} = T_{5g} + \dot{m}_a C_{pa}(T_3 - T_2)/\dot{m}_g C_{pg} \eta_t \eta_m \tag{20}$$

$$p_{4g} = p_{5g} + 778\ \dot{m}_a \cdot C_{pa}\ (T_3 - T_2)\ \rho_{mg}/\dot{m}_g\ \eta_m\ 144 \tag{21}$$

$$p_{4g} = p_{5g}\ (T_{4g}/T_{5g})^{n/n-1} \quad , \quad \rho_{4g} = \rho_{5g}\ (p_{4g}/p_{5g})^{1/n} \tag{22}$$

where ρ_{mg}, T_{5g}, n and subscript 4, mean average density of the gas through the turbine, temperature at the turbine exit, polytrophic exponent during the real expansion, and gas condition upstream of the turbine (downstream the rocket nozzle).

The rocket exhaust velocity

$$V_{4g} = \zeta \sqrt{2g\ C_{pg}(T_{\theta g} - T_{4g}) \cdot 778} \tag{23}$$

is also the inlet velocity (with appropriate angle) in the turbine. This velocity has to match conditions imposed from the turbine and the rocket combustion chamber. We take, from Eq. (23), the temperature $T_{\theta g}$ and

$$p_{\theta g} = p_{4g}\ (T_{\theta g}/T_{4g})^{n/n-1} \quad , \quad \rho_{\theta g} = \rho_{4g}\ (p_{\theta g}/p_{4g})^{1/n} \tag{24}$$

From the nozzle throat conditions (th)

$$a_{th} = \sqrt{g\ n\ p_{th}/\rho_{th}} \quad , \quad p_{th} = p_{\theta g}\ (2/n+1)^{n/n-1} \quad , \quad \rho_{th} = \rho_{\theta g}\ (2/n+1)^{1/n-1}$$

(where a indicates sound speed; we get the nozzle throat and exit (A_e) areas

$$A_{th} = \dot{m}_g \cdot g/a_{th}\ \rho_{th} \quad , \quad A_e = A_{th} \cdot a_{th}\ \rho_{th}/V_{4g}\ \rho_{4g} \tag{25}$$

The multistage turbine has to be of reaction type to give an axial thrust equal and contrary to the aerodynamic thrust (Eq. 15) plus the fan axial differental force.

The constant pressure mixing of the two streams, mass flow $\dot{m}_a$ with speed V_{5a} and temperatura T_{5a}, and gas

flow $\dot{m}_g$ with speed V_{5g} and temperature T_{5g}, is a quite complex problem to be solved experimentally, because chemical reactions are involved. Mass flows are three-dimensional and a lot of heat is transferred to the duct wall. According a constant-pressure mixing design method, we have to make use, step by step in various constant-area flow channels, of the equation of state and the conservation of mass, energy and momentum at constant state pressure.
Here we assume, as a first approximation, a simple mixing in which the resultant stream is one-dimensional, approaching the hydrogen injection zone without any intermediate combustion before the one at the hydrogen afterburners. In so doing, the main stream proceeds practically at the same initial velocity $V_{5a} = V_{5g}$, considering a coaxial mixing in which the gas speed is in advance reversed in the sense of the air speed.

Now, for getting an approximate value of the entalpy H_6 after constant pressure combustion, we consider completely daveloped the previous combustion in the rocket. In this way, acceptable for an overall approx imate design, the gas entalpy after the "ramjet" combustion is, according Eq. (6)

$$H_a + H_g + \varepsilon\, Q \cdot f/(f + 1) = H_6 \tag{26}$$

where the fuel-air ratio f depends upon the maximum acceptable temperature T_6.

The theoretical nozzle exhaust velocity, Eq. (7), and the real density (as weight average) become

$$V_7 = \sqrt{2g\, H_6\, (1 - (p_0/p_5)^{K-1/K})\ 778} \tag{27}$$

$$\rho_{7av} = \rho_{5av}\, (p_5/p_0)^{1/n} \tag{28}$$

Finally, we have Eqs. from (8) to (13) in the following modified forms

$$\dot{m}_a = (F - \dot{m}_g \cdot V_7)/|(1 + f)\, V_7 - V_0|\ 0.9 \tag{29}$$

$$\dot{m}_7 = \dot{m}_a\, (1 + f) + \dot{m}_g \tag{30}$$

$$A_7 = \dot{m}_7 \cdot g/\rho_7\, V_7 \tag{31}$$

$$A_a = \dot{m}_a \cdot g/\rho_0\, V_0 = \varepsilon\, A_1 \tag{32}$$

$$q = (\dot{m}_7 - \dot{m}_a)\ g\ 3600/F \tag{33}$$

$$\eta = \frac{V_7(1 + f)/f + \alpha\, V_7\, (1 + f^*)/f - V_0/f|\ V_0/g}{(f + \alpha\, f^*)/f \cdot |778\, Q + V_0^2/2\, g|}$$

where

f^* = hydrogen-oxygen ratio

α = oxygen-air ratio

$\dot{m}_g = \dot{m}_{oxygen}\ (1 + f^*)$

Now, from the air conditions behind the fan (pag. 18), with V_3 = 656 ft/sec and V_{5a} = 1312 ft/sec, we get: T_{5a} = 2,127 °R, $p_{5a} = p_{5g}$ = 199.25 psia, T_{4g} = 3,887 °R, p_{4g} = 758 psia, V_{4g} = 3,000 ft/sec, T_{5g} = 3,052 °R, T_{8g} = 4,285 °R, p_{8g} = 1,303 psia, V_7 = 12,670 ft/sec.
For this, we have chosen $\dot{m}_a/\dot{m}_g$ = 5, f^* = 0.285, $f = f_{st}$ = 0.0292.

Finally, for a thrust F = 36,500 lb at h = 70,000 ft, we obtain from Eqs. 29, 30, 32 33 and 34, being f^* = 0.285 and α = 0.14:

$\dot{m}_a$ = 3.56 lb sec/ft, $\dot{m}_7$ = 4.37 lb sec/ft, $A_a = A_0 = A_1$ = 6.42 ft², q = 2.55 lb/lb/hr, η = 0.53.

Fuel consumption, for a 20 minute cruise at M_0 = 4.5 and h = 70,000 ft, should result, with two engines

$$q \cdot 2F \cdot 1200''/3,600 = 62,050 \text{ lb}$$

Additional fuel consumption for take-off, climb, acceleration, descend and land, and fuel reserves, should result too high for an entire mission from take-off to 70,000 ft height and landing.

Therefore, for having reasonable payload and take-off aircraft gross weight, turbo-rockets may be employed as auxiliary engines to increase considerably the velocity of a conventional turbojet powered aircraft.

when this has already reached an appropriate flight altitude, or as rocket air augmented second stages.

CONCLUSION

This paper has described a method of approximate overall design of turbo-ramjet and turbo-rocket/aircraft. Only as a role of example, numerical computations are carried out. In particular, the author notes that the values of the shock compression ratios applied to flight Mach numbers $M_0 = 2.5$ and $M_0 = 3.5$ on Table 3 are practically too low. In fact, for example at a flight Mach number of 2.5, the static pressure ratio across the supersonic section of an inlet is practically about 11.

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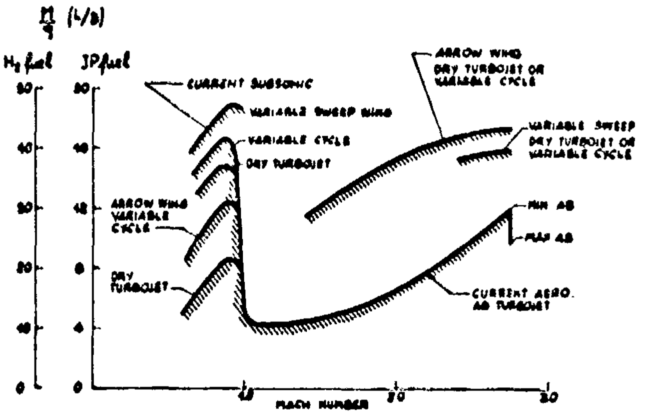
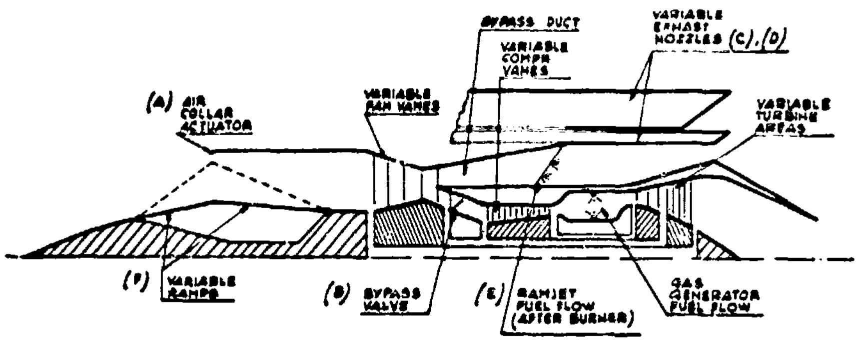


Fig. 1 - Flight efficiencies as function of flight Mach number, for present generation engine/aircraft

Fig. 2 - Variable geometric and multicycle turbo-ramjet engine



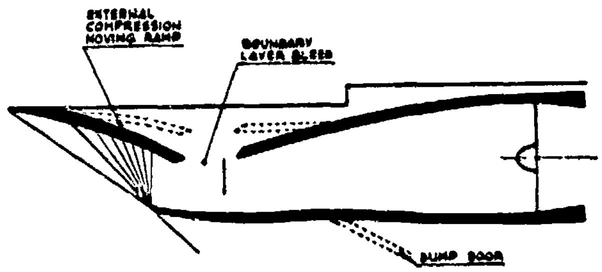
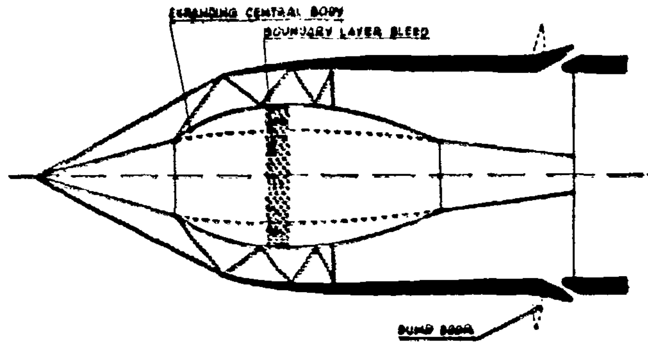


Fig. 3 - Two dimensional inlet

Fig. 4 - Axisymmetric inlet



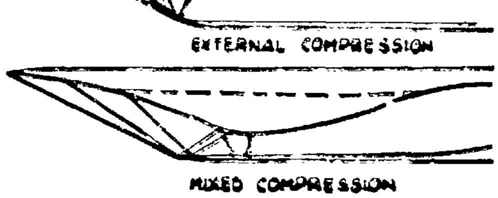


Fig. 5 - Comparison of external and mixed compression inlets

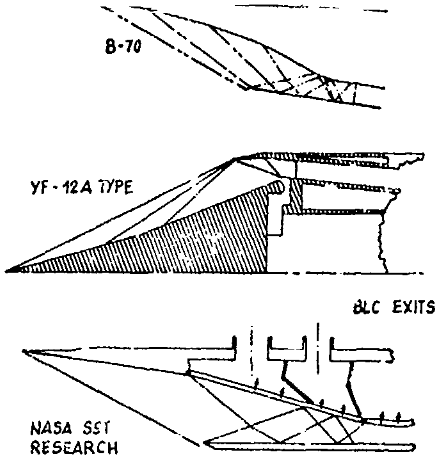


Fig. 6 - Mixed compression inlet design in actual aircraft

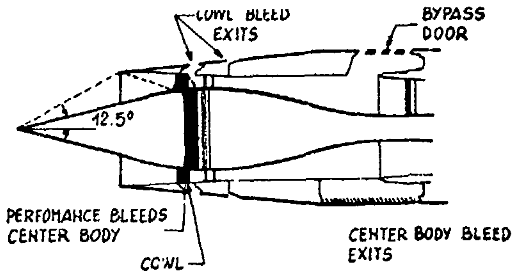


Fig. 7 - Axisymmetric mixed compression for ram jet inlet at $M_0 = 2.5$

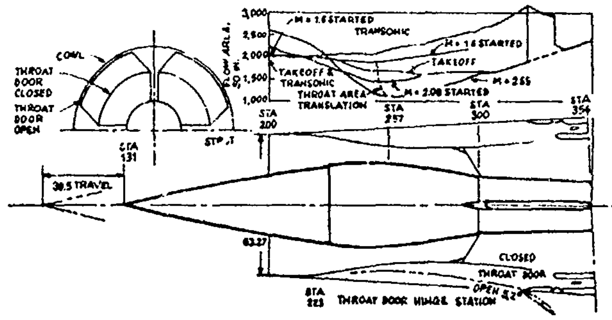


Fig. 8 - Geometry variation needed for engine airflow matching at transonic and supersonic speeds (Ref. 5)

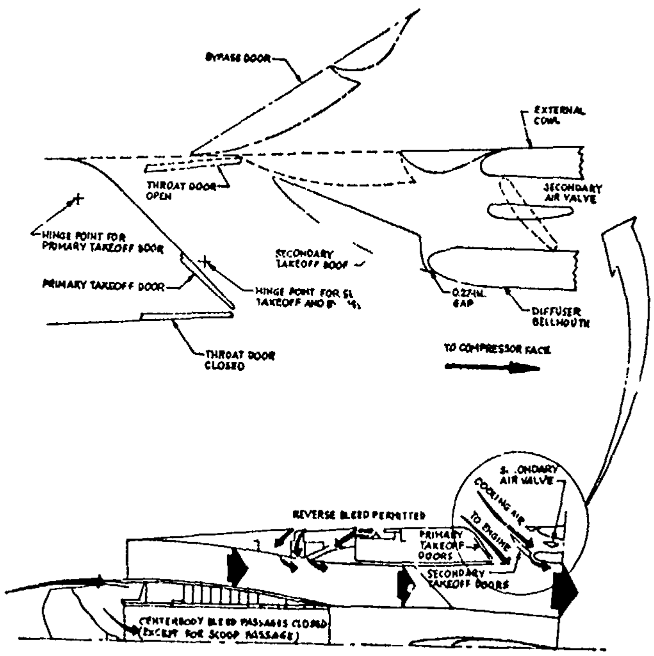


Fig. 9 - Secondary ai‹ valves bypass and take-off doors (Ref. 5)

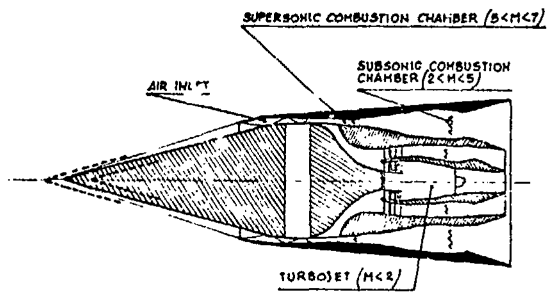


Fig. 11 - Supersonic chamber of combustion for scramjet or turbo-scramjet

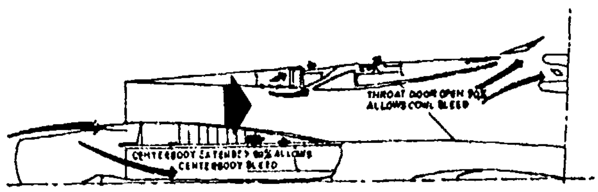


Fig. 10 - Buzz suppression mode inlet operation (Ref. 5)

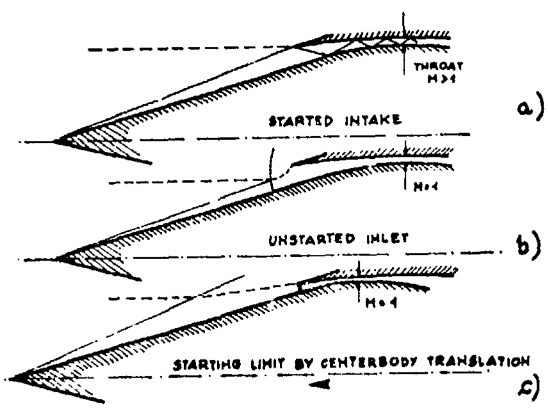


Fig. 12 - Displacement of the centerbody to modify the variable geometry as function of the flight speed

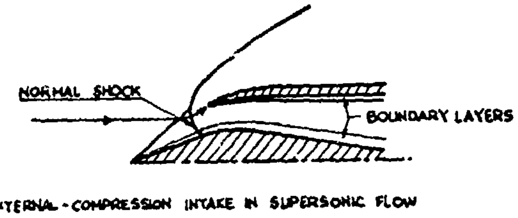


Fig. 13 - External pre-entry pressure rise in supersonic flow inlet

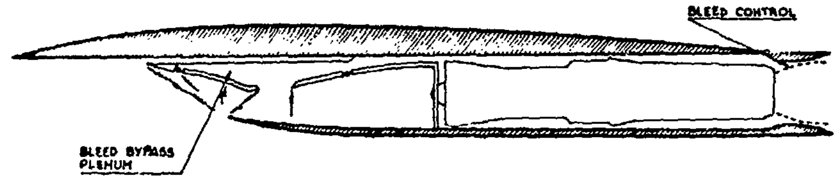


Fig. 14 - Bleed mass flow controlled by an orifice in two-dimensional shock system

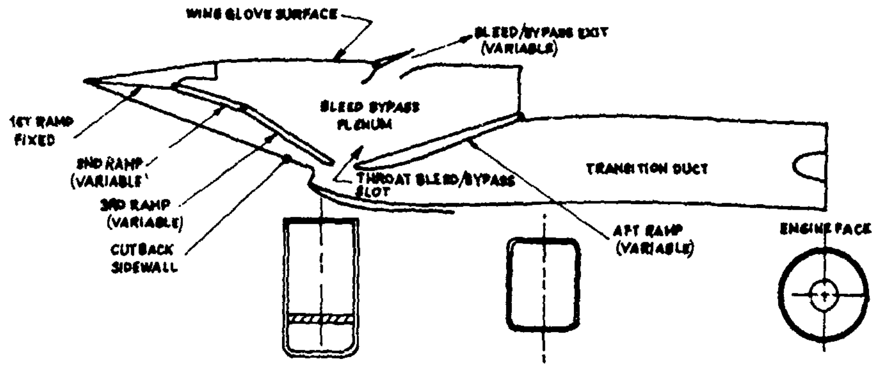


Fig. 15 - Two-dimensional all-external compression induction system design

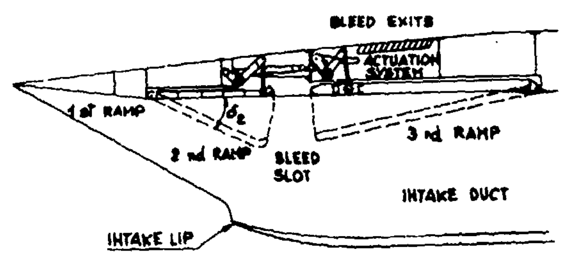


Fig. 16 - Variable geometry air intake configuration

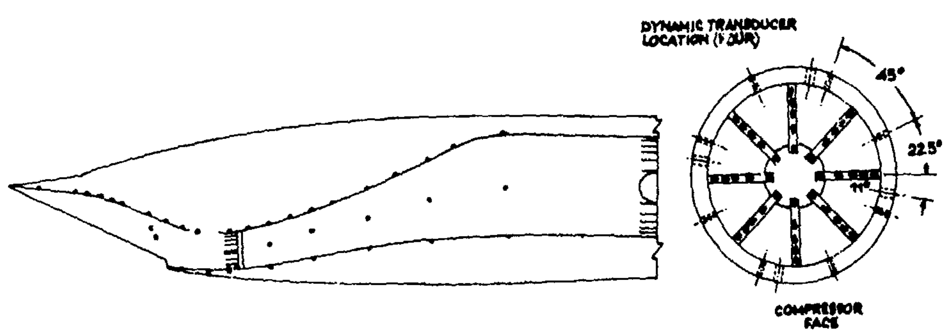


Fig. 17 - Fuselage side-mounted inlet and instrumentation for testing

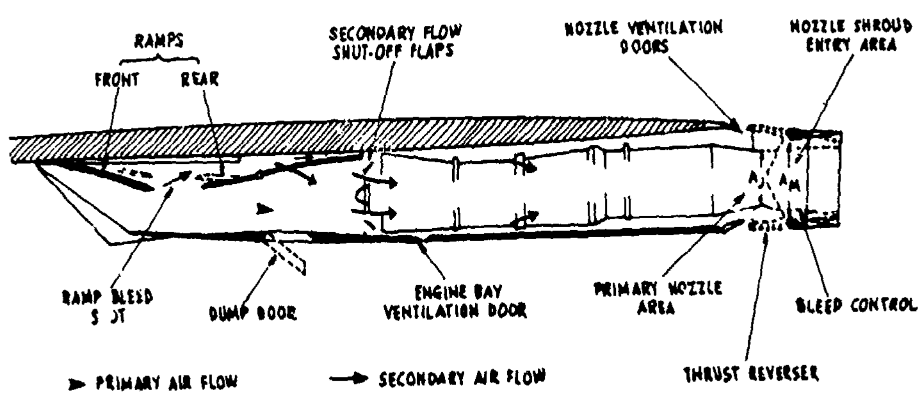
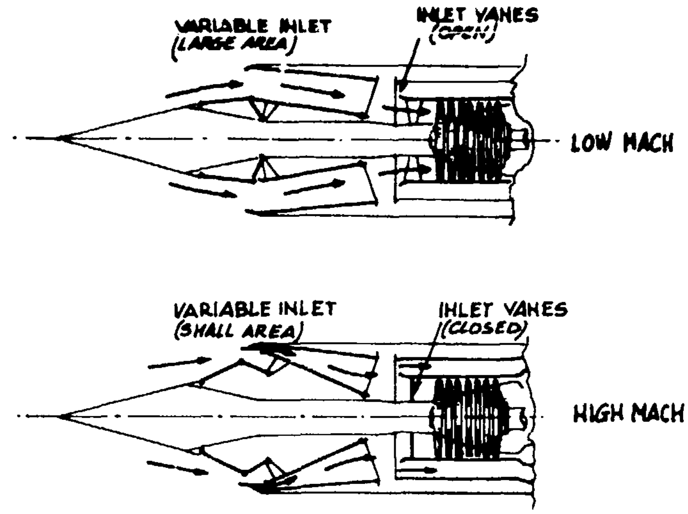


Fig. 18 - Concorde powerplant layout

Fig. 19 - Axisymmetric intake design for turbo-ram jet, as in figure 2



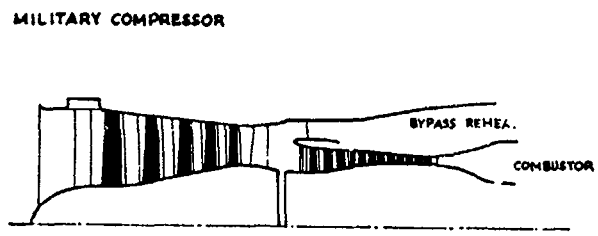


Fig. 20 - Multistage split flow fan, in conjunction with and indipendent multistage core compressor

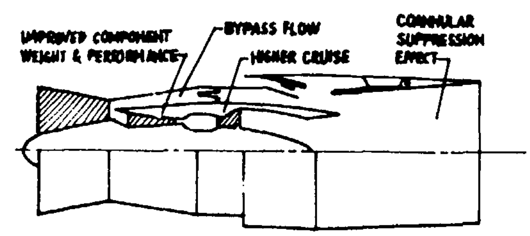


Fig. 21 - SCAR engine turbo-fan

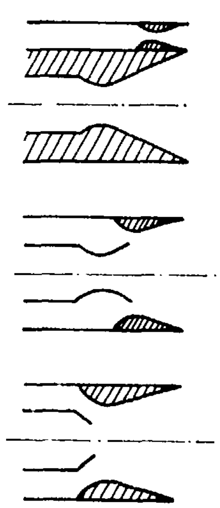


Fig. 22 - Secondary annulus flow nozzles

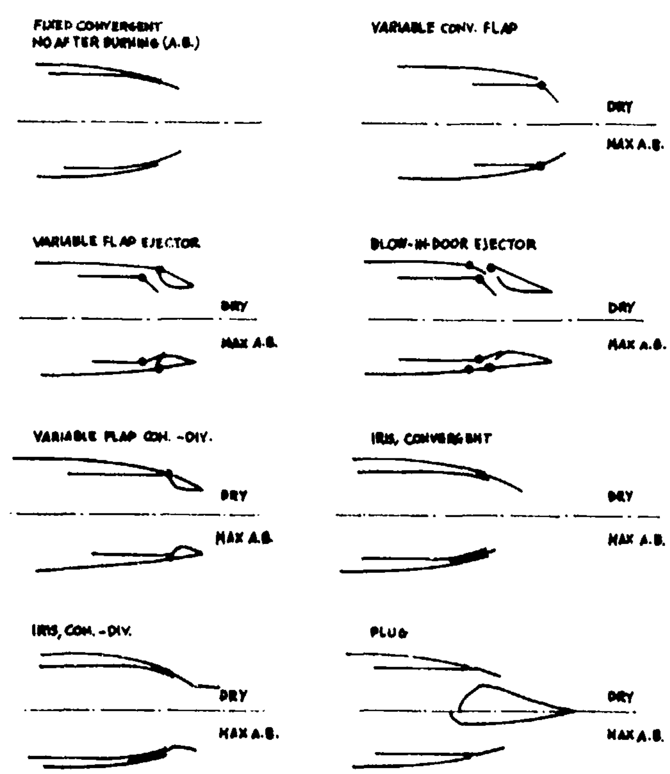


Fig. 23 - Features of existing nozzle systems

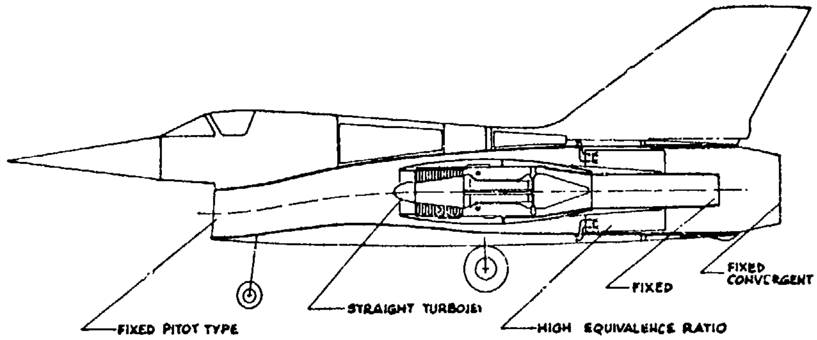


Fig. 24 - First airframe/turbo-ramjet integration, the Griffon (Ref. 7)

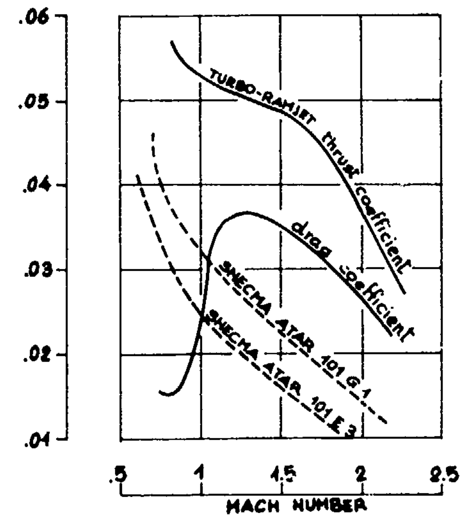


Fig. 25 - Drag and thrust coefficient along a typical Griffon climb acceleration profile (Ref. 7)

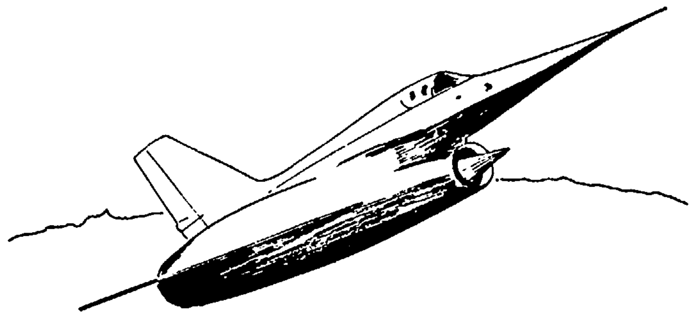


Fig. 26 - Inlet modified shape of the turbo-ramjet powered Griffon airplane

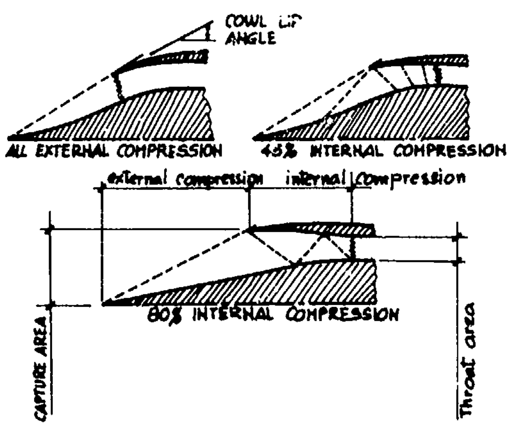


Fig. 27 - Mixed-compression inlet for long range cruise vehicle

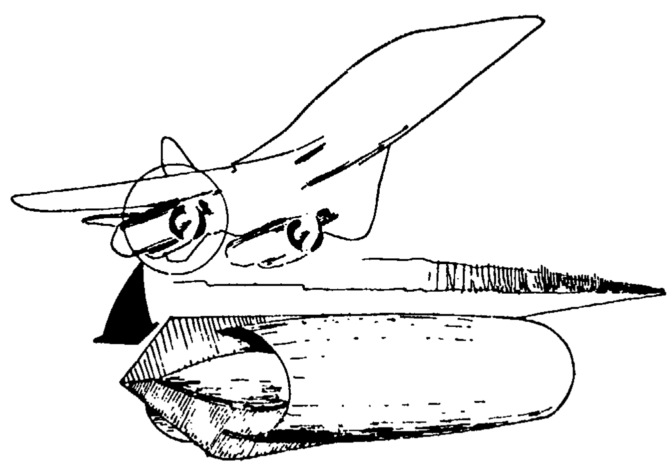


Fig. 28 - Possible twin-inlet propulsion design for turbo-ramjet engined transport

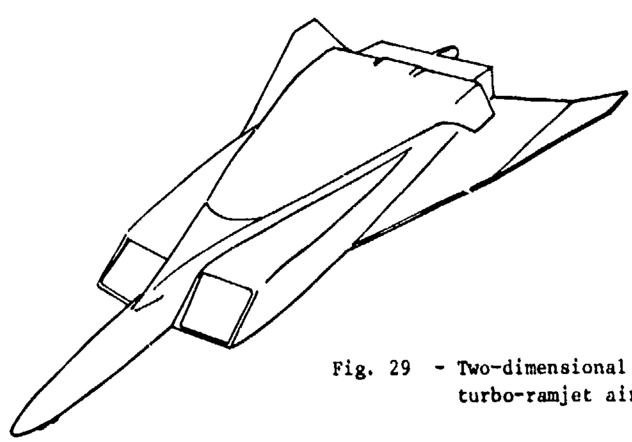


Fig. 29 - Two-dimensional inlet turbo-ramjet aircraft

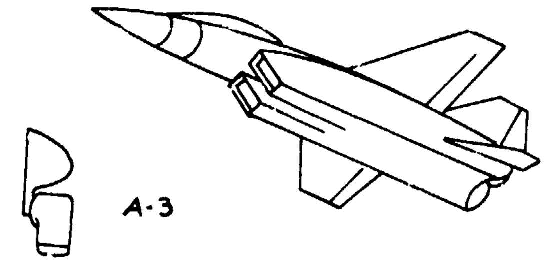


Fig. 30 - Two-dimensional inlet/airframe configuration

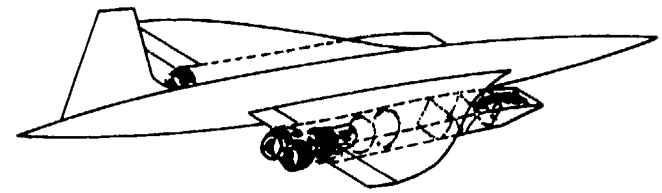
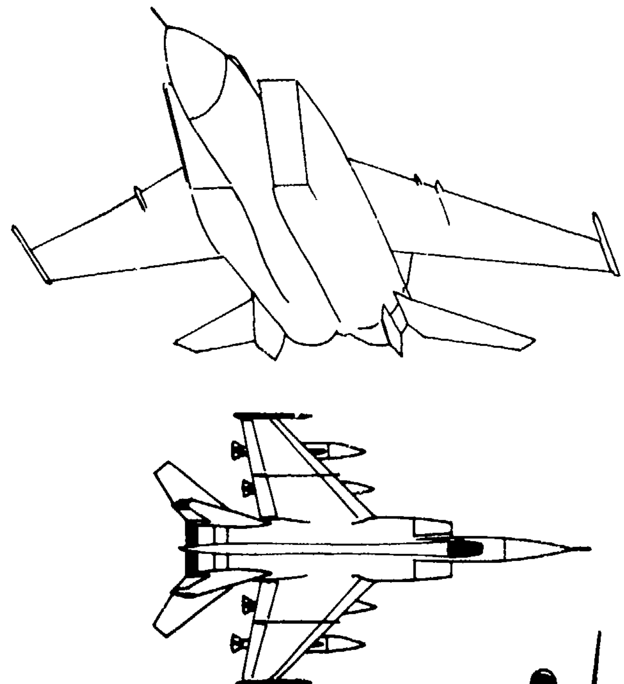


Fig. 31 - Two-dimensional underwing turbo-ramjet arrangement



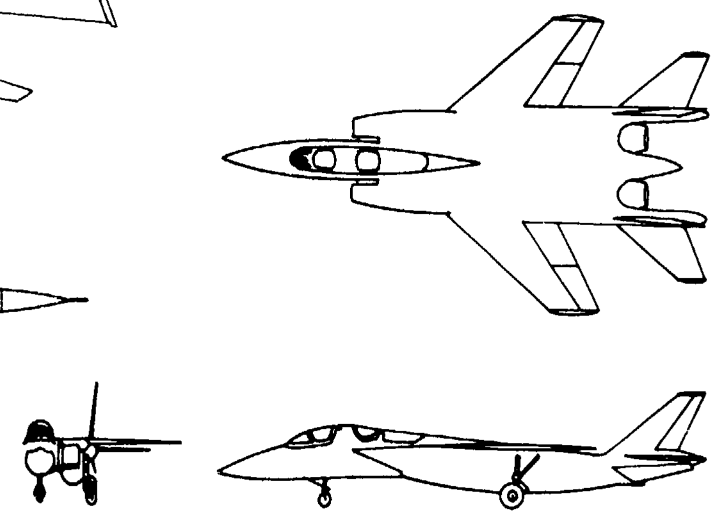


Fig. 32 - Two-dimensional inlet aircraft hydrogen fueled

Fig. 33 - Hydrogen fueled twin-turbo/ramjet engine configuration for multi-mission combat aircraft

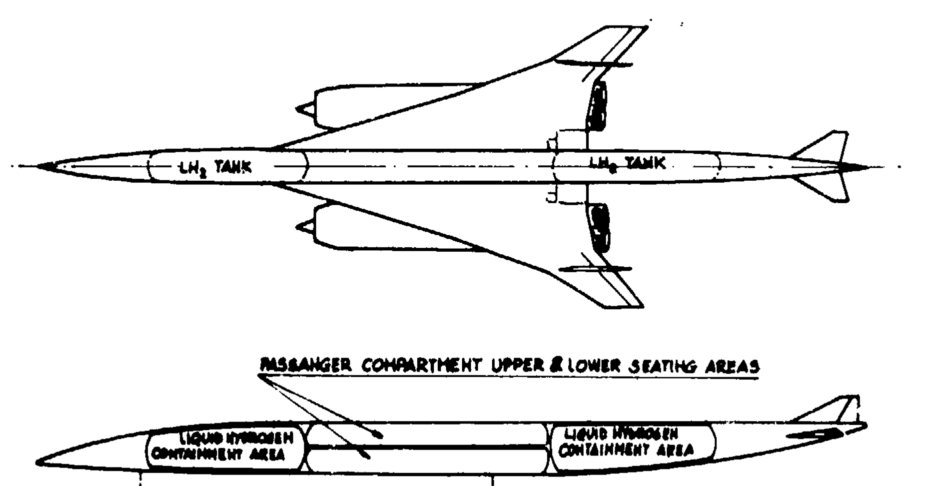
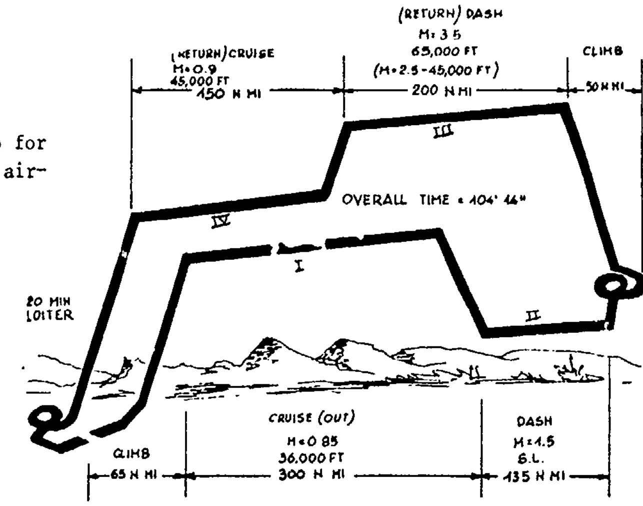


Fig. 34 - Hydrogen fueled turbo-ramjet configuration for supersonic transport

Fig. 35 - Mission scenario for advanced strike aircraft



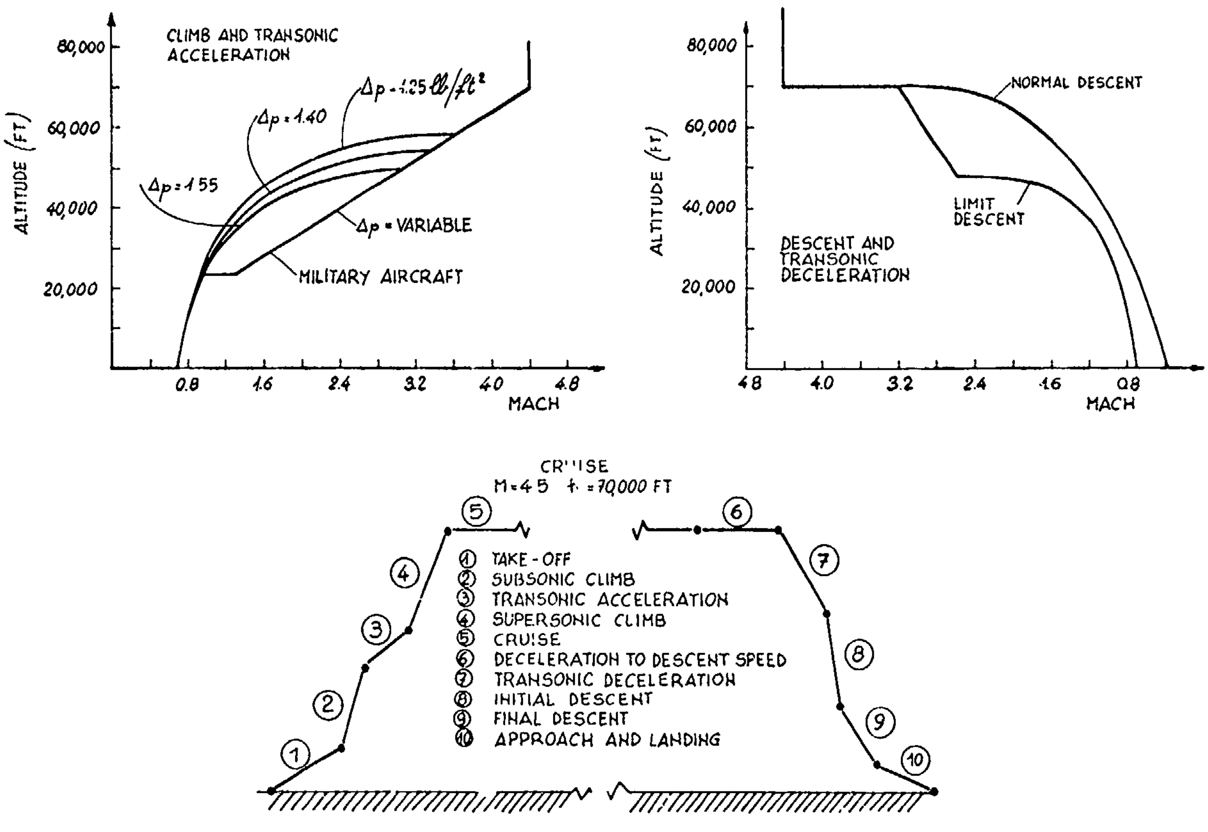


Fig. 36 - Mission scenario, climb and transonic acceleration and deceleration, for high supersonic transport

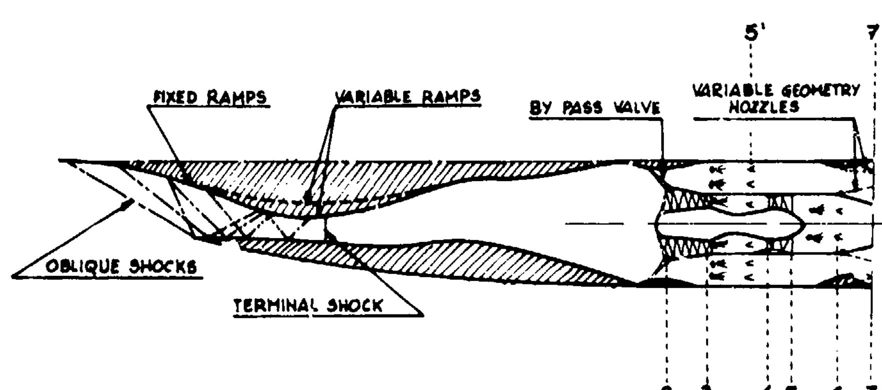


Fig. 37 - Turbo-ramjet for fighter aircraft

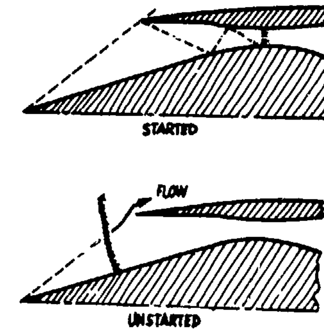


Fig. 38 - Started and unstarted conditions in two-dimensional supersonic inlets

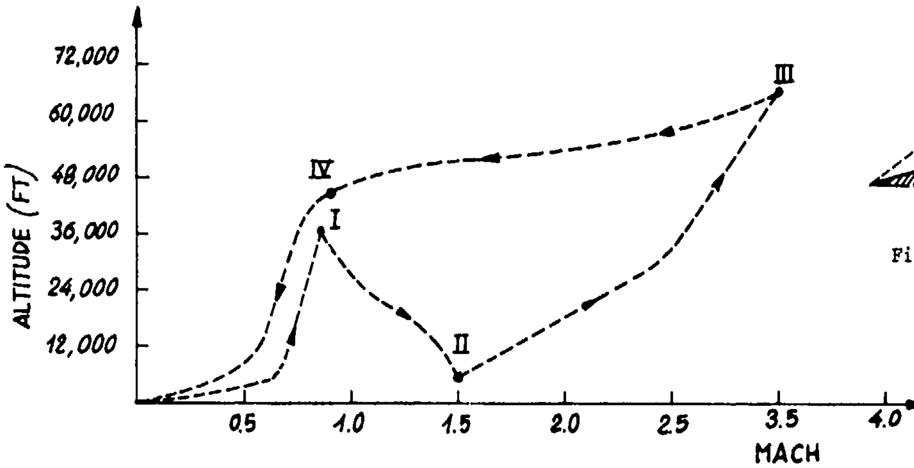


Fig. 39 - Flight Mach number/altitude relation according the mission scenario of Fig. 35

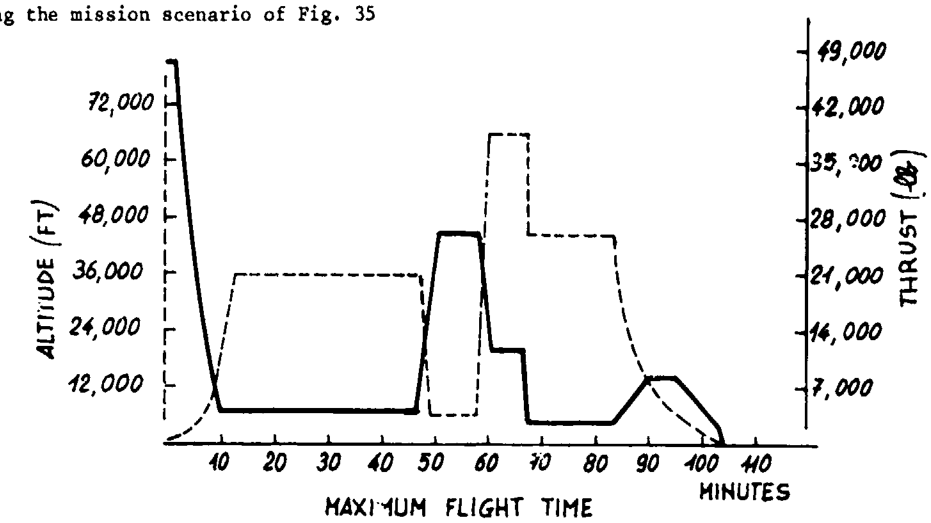


Fig. 40 - Engine/aircraft performances relative to mission scenario in figure 35

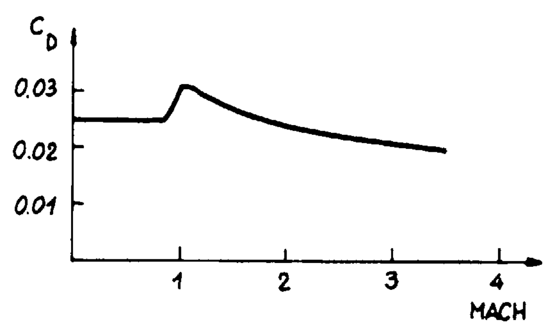


Fig. 41 - Drag coefficient for typical aircraft

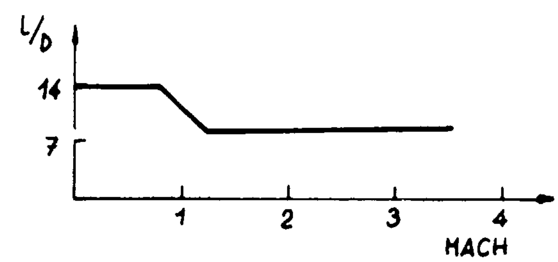


Fig. 42 - Aerodynamic efficiency for typical aircraft

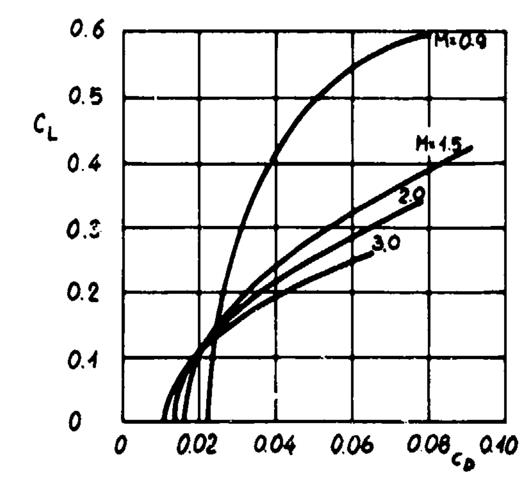


Fig. 43 - Airplane lift/drag coefficients as function of flight Mach number

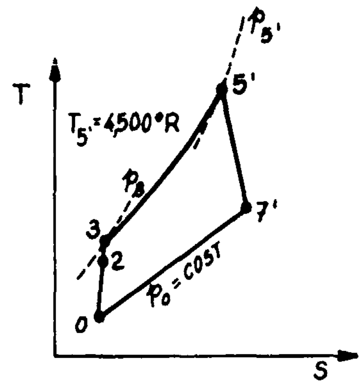


Fig. 44 - Thermal cycle for ramjet-engine

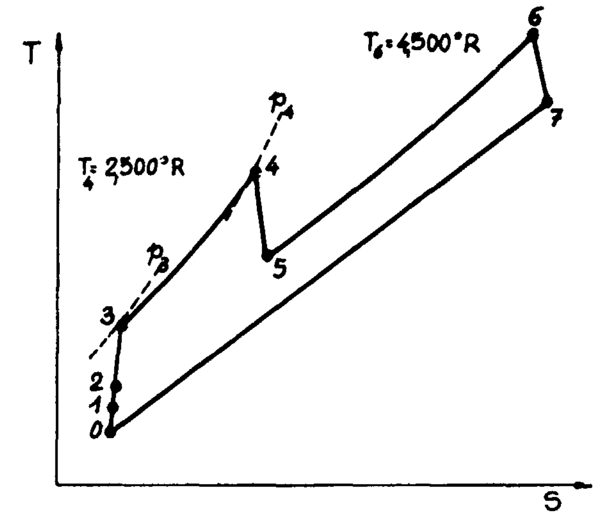


Fig. 45 - Thermal cycle for turbojet with afterburning

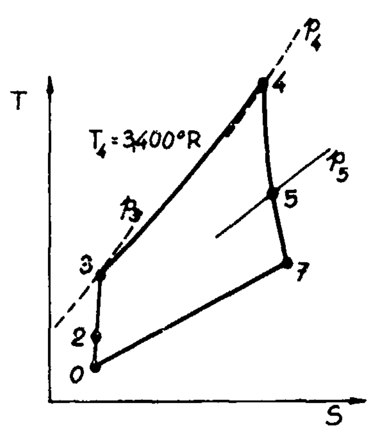


Fig. 46 - Thermal cycle for dry turbojet

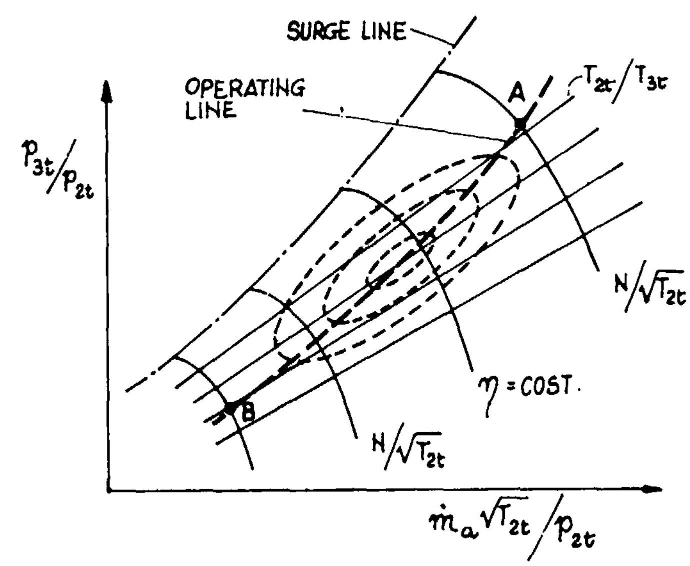


Fig. 47 - Compressor performance map of a typical turbojet engine, with operating line during increasing flight speed

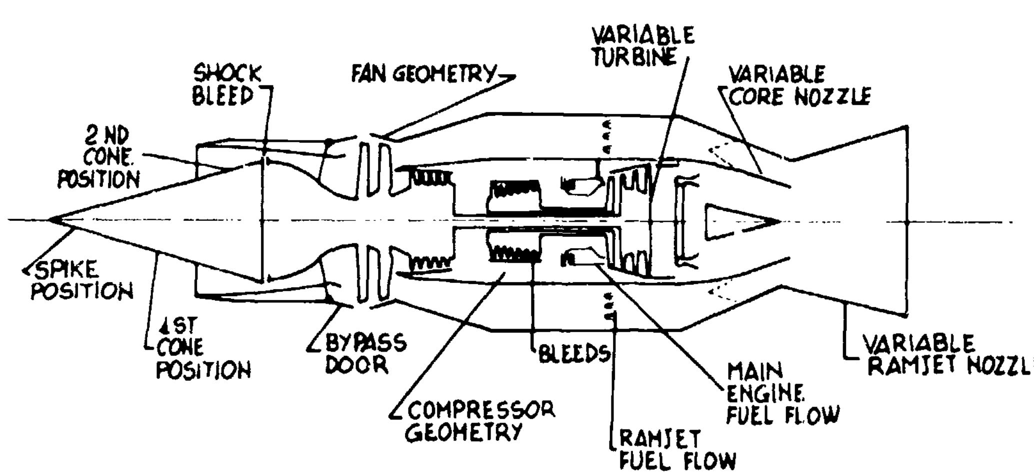


Fig. 48 - Turbo-ramjet for supersonic transport aircraft

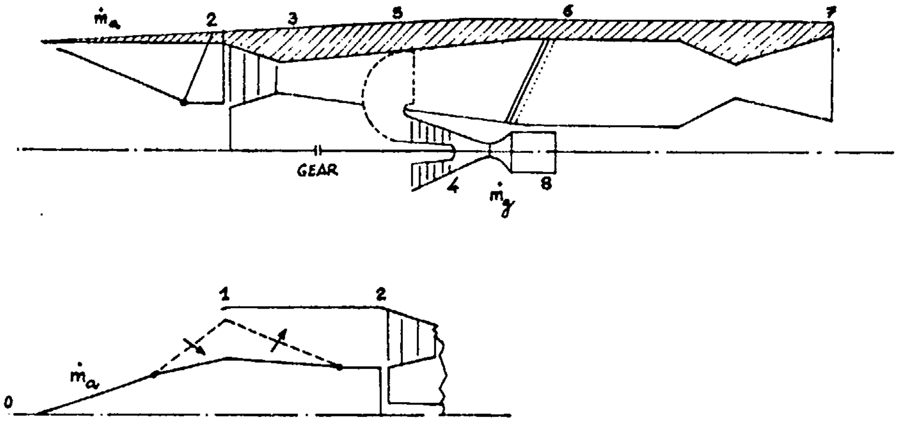


Fig. 49 - Turbo-rocket engine

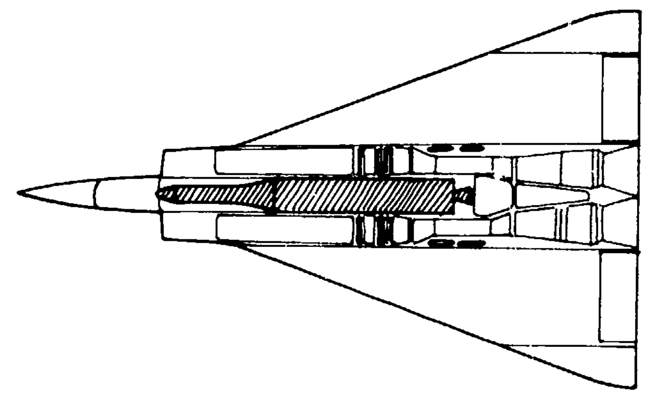


Fig. 50 - Turbo-rocket engined aircraft

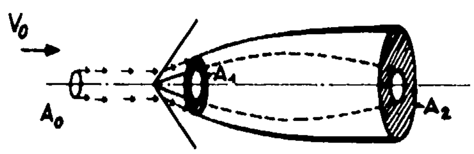


Fig. 51 - Inlet cross-sectional areas

RAM-TURBOJET ENGINE FOR LONG RANGE HIGH TERMINAL SPEED MISSIONS

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SUMMARY

Military missions which require long ranges and high speed terminal flight cannot be powered by a single conventional powerplant ; instead a combination of two engines or a multicycle engine is necessary.

Preliminary calculations are presented for a combined ramjet–turbofan engine where the long range, low speed, low altitude flight is powered by the turbofan, and the final supersonic approach at low altitude is powered by the ramjet after the turbofan has been jettisoned.

An evaluation of the system is made together with a comparison with possible competitive solutions, the design criteria are discussed, and an applicative example is presented.

LIST OF SYMBOLS

A	area
A_{ij}	A_i/A_j
A_∞	capture area
C_D	drag coefficient
C_T	thrust coefficient
D	drag
f_m	midbody fineness ratio
f_s	stoichiometric fuel/air
g	gravity acceleration
I_s	specific impulse
L	lift
M	Mach number
p	pressure
R	range
T	thrust
T_o	recovery temperature
V	speed
W_a	air mass flow rate
η_{ij}	recovery pressure between i and j
ϕ	equivalence ratio (fuel/stoichiometric fuel)

Subscripts

1, 2,..7	numbers defining characteristics area (see fig. 3)
R	ramjet or ram–phase
T	turbo or turbo–phase
∞	free stream.

1. INTRODUCTION

In a number of military applications of unmanned vehicles there may be several conflicting flight requirements such as long range, high speed, low altitude, which cannot be satisfied by a single cycle or by one type of engine.

In particular, a typical situation arises when a long range weapon must reach the target at high speed and at low flight altitude.

Specifications which are typical for a number of missions of interest are : overall range 500 - 700 Km.; terminal speed on the target : $M = 2 - 3$; high speed terminal range 50 - 150 Km. at sea level.

Some of these applications are dictated to ensure a low vulnerability and low cost/effectiveness (the system can be launched very far from the target at a safe distance).

The long range, subsonic low altitude flight calls for a low specific fuel consumption engine : the obvious solution would be a turbo engine.

The comparatively short range, low altitude, high speed terminal part of the mission calls for ramjet or rocket engines.

The second solution could be feasible obviously only when the second part of the trajectory is very short. In this case, however, the system appears to be too vulnerable.

It appears therefore that, in terms of performances, the best combination for relatively long range missions is a turbo engine (as the sustainer for the long range, low altitude high subsonic initial part of the mission) plus a ramjet (for the high speed, low altitude, medium range, terminal part of the mission).

In this paper an integration is sought between a turbo engine and a ramjet with the aim of defining a propulsion system which allows a small all-up weight (A.U.W.) of the vehicle.

In particular, the air intakes and the air ducts, and possibly some part of the fuel system, of the turbo engine and of the ramjet are the same so that a weight saving is obtained.

The adoption of a turbofan as sustainer appears to be suitable for the two following reasons : a) the SFC is better than the SFC of a pure turbojet (even at a high subsonic Mach number) and b) the by pass ratio of the turbofan is a convenient design parameter which can be used to match the geometrical requirements of the two engines and to minimize the launch weight of the system.

The proposed propulsion system is analyzed and preliminary calculations are presented which shows its feasibility and the advantages over other possible solutions.

2. VEHICLE ENGINE LAYOUT AND TRADE-OFF WITH OTHER POSSIBLE SOLUTIONS

A possible vehicle layout, which will be analyzed in the next sections, is shown in fig. 1.

The turbofan and the ramjet are in tandem configuration (fig. 2).

A sketch of a ramjet is represented in fig. 3.

The former is the " 1st stage " of the vehicle.

The fuel tanks necessary for the turbofan phase are "wrap around" in the aft end of the vehicle.

The ramjet is located between the main body and the turbo engine and air is supplied by four side inlets.

SP rocket boosters to provide acceleration from subsonic speed to supersonic cruise speed for the ramjet are also in the "wrap-around" configuration, e.g. right behind the inlets.

The fuel tank for the ramjet phase is in the midbody, while the payload is in the forebody.

The layout of the combination ramjet-turbofan is shown in fig. 2.

Under the assumption that the system is feasible (i.e. that a satisfactory matching between the turbo and the ramjet is possible), then one may expect a specific impulse for the turbofan operation (at $M \simeq 0.8$) of approximately 3600 (s) and a specific impulse for the ramjet of about 1600 (s) ; these values, together with the weight reduction obtained by integration of the two engines, lead to a propulsion system weight (engines + fuel + tanks), substantially smaller than the weight of other solutions.

If a supersonic ramjet is chosen for the entire mission, then one would have the advantage of reducing the time of flight; however, one would probably face complications in the guidance system and a very high all-up weight, due to the comparatively high SFC.

A single turbofan engine used for both flight phases (subsonic and supersonic) would be rather complex, heavy and expensive.

In this case probably a turbojet with afterburner should be utilized and variable geometry inlets and nozzles should be provided to modulate the airflow according to the engine requirements.

The possibility of using solid propellant rocket motors for the cruise phase has not been considered due to the very low specific impulse (250 sec or less), which rules out such motors for the missions considered.

An other turbo-ramjet combination considered (ref. 1, turbojet in front of the ramjet) does not appear feasible for the specified missions, because this tandem solution does not allow for turbojet jettisoning and would result in a higher A.U.W., because of the SFC penalization in both phases.

The cost/effectiveness of the proposed solution is very low also when compared to the solution of launching a ramjet powered payload from an aircraft. In this case the launch platform (which substitutes for the "first stage" of the system), has to fly close to the target, with a consequent penalization of cost and an increase of vulnerability.

3. PERFORMANCE CALCULATIONS AND DESIGN CRITERIA

In this section the performance of the turbofan and ramjet are presented separately. Turbofan performance is based on the specialized literature, see for instance ref. 2, and ramjet performance is based on the numerical program of ref. 3.

An integration of the two engines is made in terms of the by-pass ratio of the turbofan necessary to match the airflow requirements in the two phases.

3.1 Turbofan performances

The specific fuel consumption (SFC) of a typical turbofan, as a function of the by-pass ratio (BPR), is shown in fig. 4. The SFC, at M = 0.8 decreases while increasing the BPR, ranging from 1.25 to 1.0 Kg/h/dN, with BPR ranging from 0 (turbojet engine), to 3.

Fig. 5 shows the thrust/airflow ratio as a function of BPR, for the same Mach numbers. The curves were computed assuming a constant turbine inlet temperature.

High BPR lead to low values of SFC, but has a negative effect on engine size due to the higher airflow requirements for a given thrust.

Fig. 6 shows a typical thrust/weight trend as function of BPR. The plots are necessary in choosing the turbofan engine for the application in our system.

Once the thrust level has been chosen for a fixed outer diameter engine, the engine weight and SFC dictate the propulsion system weight (engine + tank + fuel) for the turbo-stage, and for the given mission.

At the design point, the air mass flow rate, W_a, must be supplied to the turbo engine by the common air intake ; this means that the air intake area (A_1) must be properly sized, and that choking must be prevented at all the stations downstream of the inlet area.

The final choice, however, cannot be done independently of the second stage, because the common parts of the engines require a proper design to match both the ram and the turbo operations.

3.2 Ramjet performances

Figgs. 7, 8 and 9 (taken from ref. 3) show the maximum specific impulse trend as a function of equivalence ratio ($\varnothing$) and as a function of the engine thrust coefficient C_T, based on the ramjet combustion chamber cross sectional area (A_3) (see fig. 3), and of flight Mach number (M_∞).

The computations have been performed numerically with the following assumptions :

a) One-dimensional flow
b) Subsonic combustion ($M_3 \simeq 0.2$)
c) Equilibrium chemistry
d) Combustion efficiency 0.95
e) Inlet recovery according to MIL Standard Specifications
f) No air spillage and no air bleed.

The possibility of obtaining the maximum specific impulse, for a given thrust level and flight Mach number, is dependent on a proper sizing of the ramjet i.e. geometric capture area (A_1), diffuser throat area (A_2), nozzle throat (A_6) and exit area (A_7), referred to the combustor area (A_3).

Geometric capture area A_1

The area ratio $A_{13} = A_1/A_3$, which depends on the flight Mach number, on the equivalence ratio, and on the thrust coefficient, is one of the most important design parameters because it prescribes the geometric air intake capture area, for a given combustor area (which coincides with the missile main-body cross sectional area).

The value of A_1 must be the same for both the turbo and the ramjet engine. In figgs. 7, 8, 9 the ratio A_{13} versus $\varnothing$ is presented ; the values of A_{13} which optimize the I_s, for $M_\infty = 2.5$ range between .25 - .5.

Diffuser throat area A_2

The diffuser throat area A_2, designed for the proper ramjet operations, is dictated by the starting of the diffuser (if a partially internal compression is required) and is given by (ref. 4).

$$A_{21} = \frac{F(M_{\infty R})}{F(1)} \frac{1}{\eta_{\infty 2}} \qquad (1)$$

where F(M) is the mass function (ref. 5) and $\eta_{\infty 2}$ is the total pressure recovery between stations 1 and 2.

Nozzle throat area A_6

The ratio A_{63} has been calculated to ensure the prescribed ramjet operations ($M_6 = 1$). A check should be made to advoid choking at A_6 during the turbofan operation. This will be shown in the following sections.

Exit area A_7

The exit area (A_7) is calculated to give full expansion unless $A_{73} = A_7/A_3 > 1$. In these cases $A_{73} = 1$ has been assumed allowing for an underxponded nozzle.

Supercritical operation of the inlet has been assumed.

3.3 Ramjet-turbofan matching

The combination of the engines, following the sketch of fig. 2, requires that the air duct be able to supply the necessary air to the engines during both flight phases.

Hence the matching of the two engines consists in : a) designing the inlet area (A_1) so that the turbo and the ramjet engines provide the necessary thrust at near optimum specific impulse conditions ; b) providing a satisfactory diffuser area distribution for both operations ; c) checking that there is no choking at the ramjet nozzle throat during turbo operation ; d) checking that Mach numbers in the combustion chamber and at the compression face during turbo operation are appropriate.

a) The thrust ratio (both flights at sea level) is given by :

$$T_T/T_R = D_T/D_R = C_{DT} M_{\infty T}^2 / C_{DR} M_{\infty R}^2 \qquad (2)$$

where the drag coefficients are related to the same cross sectional area (A_3).

The ratio of the required air mass flow rates are :

$$\frac{W_{aT}}{W_{aR}} = \frac{W_{aT}}{T_T} \frac{C_{DT} M_{\infty T}^2}{C_{DR} M_{\infty R}^2} g (I_s)_R \phi_R f_s \qquad (3)$$

On the other hand :

$$W_{aT}/W_{aR} = M_{\infty T} A_{\infty T} / M_{\infty R} A_{\infty R} \qquad (4)$$

where $A_{\infty T}$ and $A_{\infty R}$ are the capture areas for the two flight conditions.

The thrust/airflow ratio of the turbofan is a function of the by-pass ratio (BPR) and Mach number ($M_{\infty T}$) (fig. 5) :

$$W_{aT}/T_T = f(BPR, M_{\infty T}) \qquad (5)$$

The value of BPR can be found from eq. (5), by means of eqs. (3, 4) :

$$f(BPR, M_{\infty T}) = \frac{C_{DR}}{C_{DT}} \frac{M_{\infty R}}{M_{\infty T}} \frac{1}{g (I_s)_R \phi_R f_s} \frac{A_{\infty T}}{A_{\infty R}} \qquad (6)$$

Once the flight Mach numbers have been chosen and the drag coefficients evaluated, eq. (6) gives a first estimate of the BPR as a function of the specific impulse and of the equivalence ratio of the ramjet and of the ratio $A_{\infty T}/A_{\infty R} = A_{\infty T}/A_1$ (as we always assume $A_{\infty R} = A_1$ during ramjet operations).
Supposing that no spilling of air occors during both phases, a lower limit of BPR may exist. In fact, low values of BPR call for small intake capture areas, which could not ensure the required airflow to the ramjet, and thus the required thrust level.
This limit can be found from the following equation :

$$f(BPR) = 2(A_{13})_R / V_T\, C_{DT} \qquad (7)$$

and from the plots of ramjet A_{13} of figs. 7, 8, 9.

b) The diffuser throat area, designed for proper ramjet operation (see sub-sect. 3.2), may choke the flow during turbo operation resulting in a strong spilling of airflow.
In order to avoid the spilling, a variable geometry inlet has to be provided :
During turbo operation the throat area must be large enought ($A_{12} \cong 1$) to prevent spilling while during ramjet operation the throat area is dictated by the starting and swallowing ($A_{12} \cong 2$).
A rather simple diffuser has to be designed which allows a "switch" from one condition to another.
The other possibility (fixed geometry inlet) calls for spilling during the turbo phase, as the air mass flow is dictated by the inlet throat.

b.1) In the first case (variable geometry, no spilling), $A_{\infty T} = A_{\infty R} = A_1$, then the BPR is given by :

$$f(BPR) = \frac{C_{DR}}{C_{DT}}\,\frac{M_{\infty R}}{M_{\infty T}}\,\frac{1}{\mathcal{L}(I_s)_R\,\phi_R\, f_s} \qquad (8)$$

and the optimum ramjet equivalence ratio can be selected as a design point. Then the BPR is found, and the other turbofan parameters can be calculated.
For example, assuming (see also example shown in sec. 4) :

$M_{\infty T} = 0.8$
$M_{\infty R} = 2.5$
$C_{DT} = 0.6$
$C_{DR} = 0.4$

and (fig. 8) :

$(\phi_R)_{opt} = 0.4$
$(I_s)_R = 1650$ s

we obtain :

$T/W_a \cong 20$ dN/Kg/
BPR $\cong$ 2.5
SFC $\cong$ 1 Kg/h/dN

b.2) In the second case (a fixed geometry inlet), the diffuser throat size necessary for starting is given by eq. (1).
Hence equation (6) becomes :

$$f(BPR) = \frac{C_{DR}}{C_{DT}}\,\frac{M_{\infty R}}{M_{\infty T}}\,\frac{1}{\mathcal{L}(I_s)_R\,\phi_R\, f_s}\,\frac{F(M_{\infty R})}{(\eta_{\infty 3})_R\, F(M_{\infty T})} \qquad (9)$$

This equation sets a limit to the BPR. Matching between the two engines may be possible, provided that an adequately low BPR is chosen, according to eq. (9). For example, with the assumptions made, we obtain :

$A_{21} \cong 0.45$

and, then :

$A_{\infty T}/A_3 \cong 0.47\,(A_{13})_R$

From fig. 8 and eq. (9), the turbo parameters become :

$T/W_a \cong 42.5$
BPR $\cong 0.4$
SFC $\cong 1.2$

and therefore a pure turbojet may be chosen.

c) Depending mainly on the ramjet design Mach number, a condition may be reached where the nozzle throat area (A_6) necessary for ramjet operation, becomes too small for turbo operations thus leading to choking and spilling of air.
This may set a limit to the maximum BPR of the turbofan.
The condition of "no choking" in the nozzle throat may be represented by means of the following relation :

$$\frac{F(M_{\infty T})}{F(M_{\infty R})}\,\frac{F(1\,\gamma_{6R})}{F(1)}\,\frac{(\eta_{\infty 6})_R}{(\eta_{\infty 6})_T} \leq \sqrt{T_{o_4}\,\text{combustion}\,T_{o\infty R}\,(1+\phi_R f_s)} \qquad (10)$$

which implicately represents the condition :

(A_6) necessary turbo $\leq$ (A_6) necessary ram

Fig. 10 shows an example of nozzle throat area trend with ϕ, for $M_{\infty R} = 2.5$ and $C_T = 0.4$. It can be seen that by decreasing ϕ_R the ratio A_{61} decreases.

d) As far as the Mach number in the combustion chamber during the turbo phase is concerned, it ranges from 0.1 up to 0.3 depending on the design conditions, thus assuring small pressure losses.
A more appropriate criterium of designing the engines must be based on the minimization of the overall propulsion system

weight for a given mission, taking into account SFC, drag and engine weight variations with BPR still checking that no choking occurs during turbo operations.
Obviously the mission ranges influence the choice of turbofan BPR. In fact, in the limiting case of $R_R/R_T \rightarrow 0$ it would not pay to optimize the ramjet performances ; viceversa if the ramjet range is long (as compared to the turbo range) the overall propulsion system weight optimization calls for an optimum ramjet design.

4. SAMPLE MISSION AND ENGINE DESIGN

Let us show how to size a complete system for the following mission :

Ramjet operation :

Range : 100 Km
Mach : 2.5
Altitude : sea-skimming.

Turbofan operation :

Range : 500 Km
Mach : 0.8
Altitude : low altitude (sea-level)

A payload of 230 Kg. and an airframe weight of 90 kg. are assumed. The vehicle is to be propelled by the engine combination discussed in sect. 3 and whose configuration schematically appears in fig. 1. We will examine and conduct a preliminary design for each of the engines separately, i.e. the turbofan, the booster rockets and the ramjet.
Matching the turbofan and the ramjet will be performed by the criteria outlined in sect. 3.3.

4.1 Ramjet phase

The overall "second stage" weight and dimensions are determined according to figs. 11, 12, 13. In these figures the range, the weight, the length and the diameter are related in a parametric form. The curves were prepared by a computer program (ref. 6) with the following assumptions :

Launch Mach number 0.8
Cruise Mach number 2.5

The cruise drag coefficient has been assumed as a function of the midbody fineness ratio :

$C_D = 0.35 + 0.005 \ (fm)$

and the specific impulse of the ramjet as a function of C_D, according to figs. 7, 8, 9.
The boost phase accelerates the vehicle from M = 0.8 to M = 2.5 . A booster structural factor 0.3 has been assumed, with $I_s = 250$ s.
The curves of figs. 11, 12, 13 show the variation of range, dart weight and A.U.W. as a function of missile diameter and missile fineness ratio.
The mission defined above is satisfied by the missile having the following characteristics :

Diameter	0.4 m
Length	5.0 m
Payload	230 Kg
Propulsion system (ramjet)	280 Kg
Airframe	90 Kg
Dart	600 Kg
Boosters	280 Kg
A.U.W.	880 Kg.

Drag coefficient of this missile is about $C_D = 0.4$.

4.2 Booster phase

The booster phase provides acceleration from the turbo cruise Mach number (M = 0.8) up to the ramjet cruise Mach number (M = 2.5). It was determined (sec. 4.1) that the overall booster weight is

$W_b = 280$ Kg

and the relative propellant weight

$W_{bp} = 200$ Kg

We assume four wrap-around boosters, located in the aft end of the missile, (see fig. 1) having roughly the following dimensions :

diameter : 0.17 m
length : 1.5 m

4.3 Turbofan phase

In the turbofan phase the missile defined in sect. 4.1 is the payload. The main parameters of this phase are :

Cruise Mach	M_{crT}	= 0.8
Range	R_T	= 500 Km
Flight time	t_T	= 0.51 h

Furthermore the following ratios seem appropriate for the system :

Lift/Drag	LID = 3.5
Fuel/A.U.W.	F_R = 0.15

The SFC of the engine must not be greater than :

$$SFC = F_R \frac{L}{D} \frac{1}{t_T} \cong 1 \quad Kg/h/dN$$

Hence a double flux engine seems appropriate for this application (fig. 4).

It is supposed, in this and the following subsection that a variable geometry inlet is used, hence allowing $A_{ooT} = A_1$, unless spilling due to nozzle throat occurs.

Tab. 1 summarized the main parameters as a function of by-pass ratio, for the mission defined before. A drag coefficient slightly increasing with BPR has been assumed. In a detailed analysis a careful study of this point has to be made also taking into account the available engines.

The capture area (equal to the geometric capture area if there is no spilling), referred to $S_R = A_3$ is given by :

$$A_{\infty 3} = W_a / \rho_\infty V_\infty A_3$$

or by :

$$A_{\infty 3} = 1/2\, V_\infty C_D / g\, I_s \phi f_s$$

and is shown in fig. 14 as a function of BPR.

4.4 Matching of turbofan and ramjet. BPR choice

A variable geometry inlet, according to sec. 3.2, is assumed.

In the table 1, the ramjet equivalence ratio $\emptyset$, as a function of A_{13} calculated in the previous section, is presented. The equivalence ratio is found from fig. 8 (or fig. 10). In the next column of table 1, the maximum specific impulse of ramjet is presented, as found from fig. 8 (or fig. 10).

Drag during the ram phase, is about 22000 N and the fuel weight will be given by :

$$W_{fR} = D_R R_R / g\, I_{SR} V_R$$

Low values of BPR (less than about 0.6) call for small air intakes capture area, which does not ensure the required thrust level during the ram phase (see sect. 3.3, point a), as it is seen from tab. 1 ore fig. 8.

On the other hand, increasing values of BPR lead to values of the ramjet I_s, first increasing and then decreasing (tab. 1). In addition the condition is reached in which the nozzle throat dictated by ramjet chokes the flow during the turbo phase (sect. 3.3, point c). This condition corresponds to a BPR 2 ($\emptyset_R$ = 0.435, A_{13} = 0.32).

Fig. 15 shows the plots of the fuel weight required during the two phases and the weight of the turbofan. The two limits discussed above are also shown.

Fig. 16 shows the overall weight of propulsion system (engines + fuel + tank) as a function of BPR. For the sample mission considered the curve is quite flat, and the minimum is very near to the limit BPR = 2.

If a BPR = 2 is selected, the following weight breakdown is obtained for the entire vehicle :

Payload weight	230 Kg
Airframe weight	90
Fuel weight (ram)	160
Fuel tank (ram)	25
Ramjet engine	95
Dart second stage	600
Boosters	280
A.U.W. second stage	880
Fuel weight (turbo)	167
Fuel tanks (turbo)	25
Turbofan engine	65
Fittings-fins	30
A.U.W. first stage	1167

Overall length will be about 6 m. with a body diameter d = 0.4 m.

5. CONCLUSION

Preliminary calculations have been presented for a combined turbofan-ramjet engine where the turbofan is the sustainer motor for a long range, low altitude, subsonic flight and the ramjet powers a medium range, low altitude, supersonic flight, after the turbofan has been jettisoned.

The engine investigated may be of interest in the applications to the "cruise missiles" (ref. 7, 8), for which a final supersonic approach may be requested.

The vehicle configuration is also interesting as it increases penetration, and reduces the effectiveness of the defences.

The proposal combination appears, for these classes of missions, the most promising one. It has been shown that the turbofan BPR is a good parameter to match the engines and the design criteria have been illustrated. The relative values of ranges, as well as the Mach numbers have a direct influence on the engines design.

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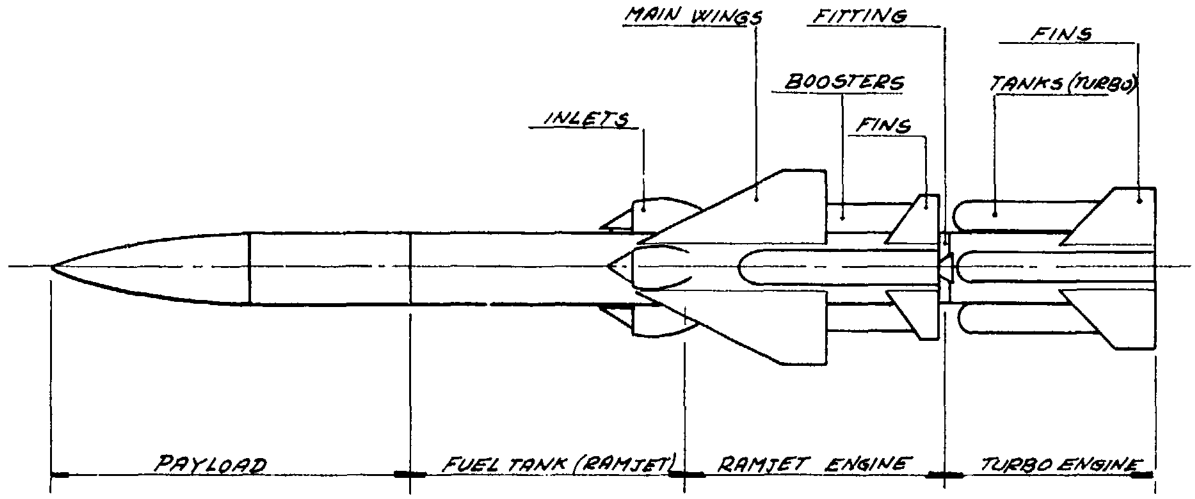

FIG. 1. VEHICLE LAYOUT

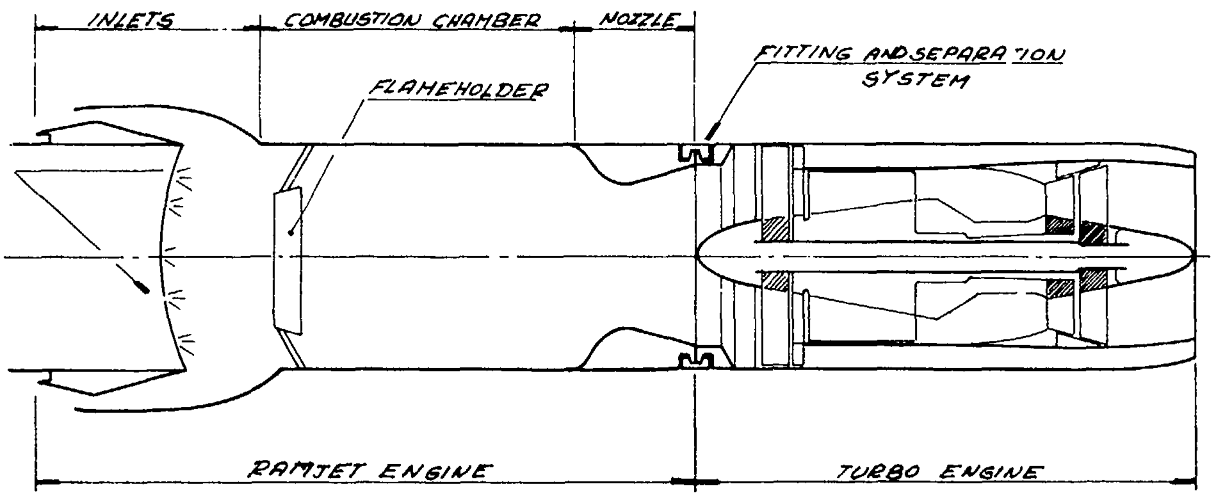

FIG. 2. RAM-TURBO ENGINE LAYOUT

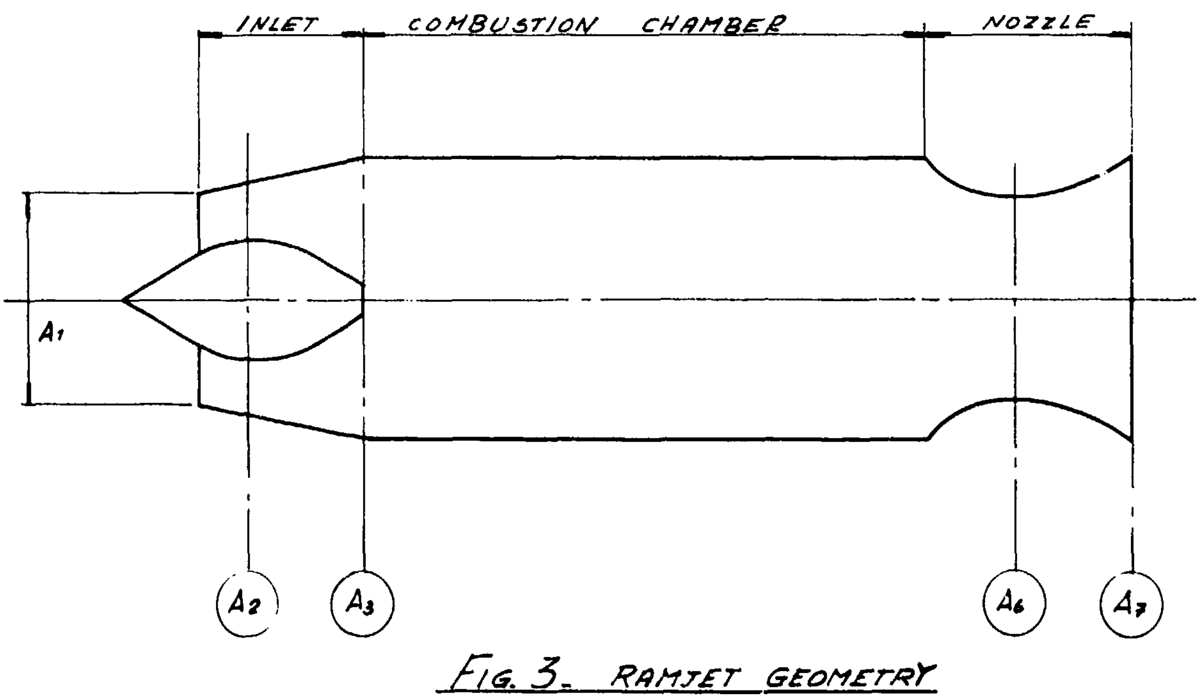

FIG. 3. RAMJET GEOMETRY

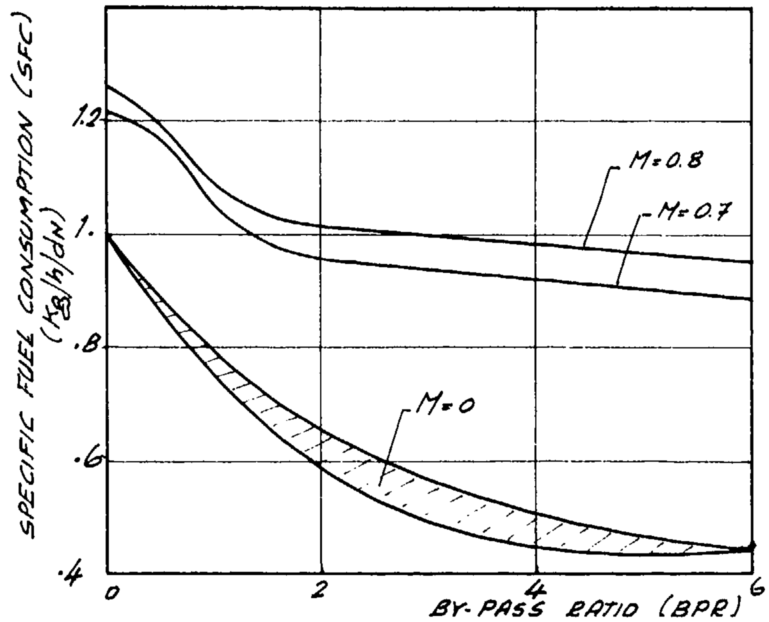

FIG. 4 – SPECIFIC FUEL CONSUMPTION AS A FUNCTION OF BY-PASS RATIO

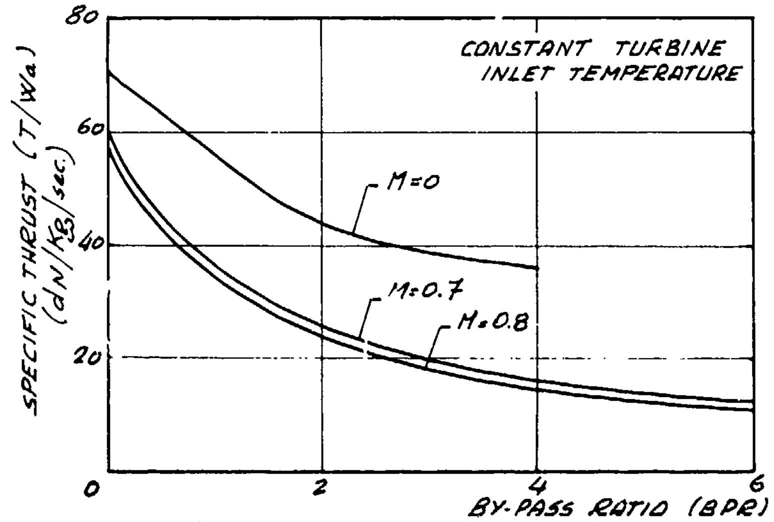

FIG. 5 THRUST-AIRFLOW RATIO AS A FUNCTION OF BY-PASS RATIO

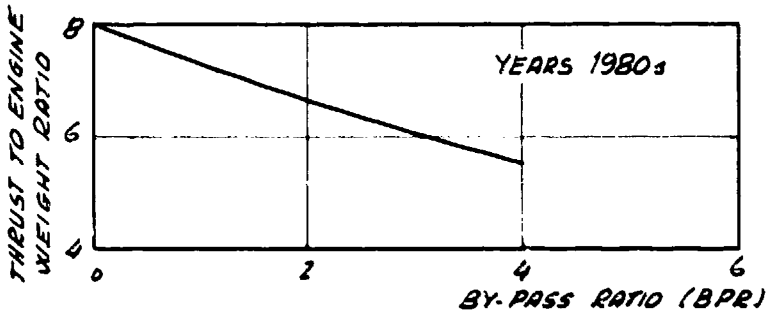

FIG. 6. THRUST/WEIGHT TREND

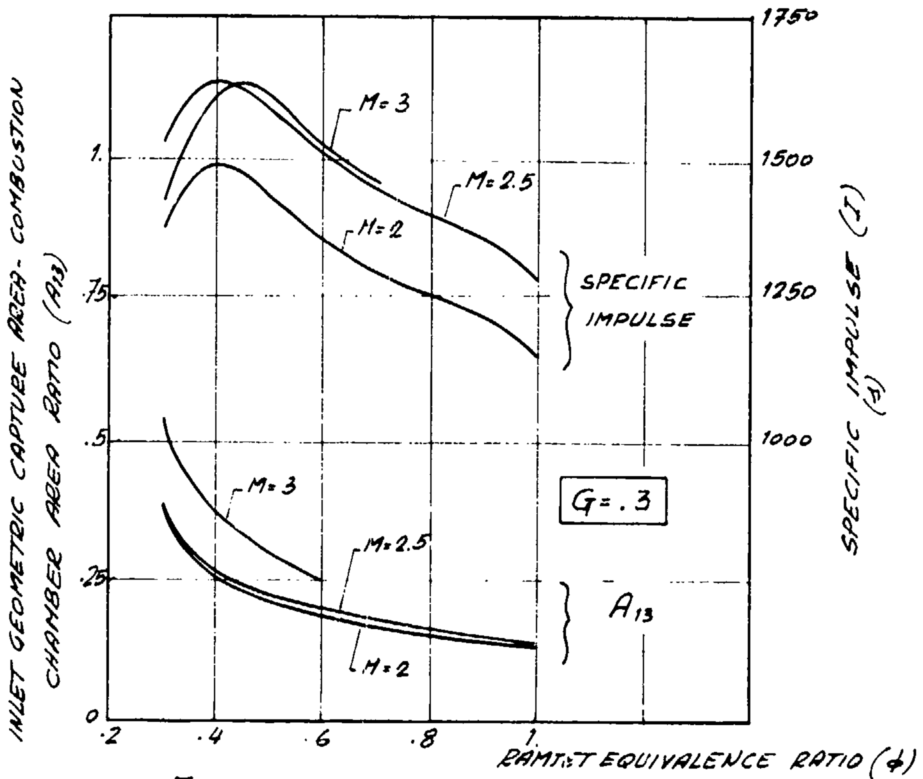

FIG. 7. RAMJET PERFORMANCES.
INLET GEOMETRIC CAPTURE AREA AND MAXIMUM SPECIFIC IMPULSE VERSUS THE EQUIVALENCE RATIO, AS FUNCTION OF DESIGN MACH NUMBER

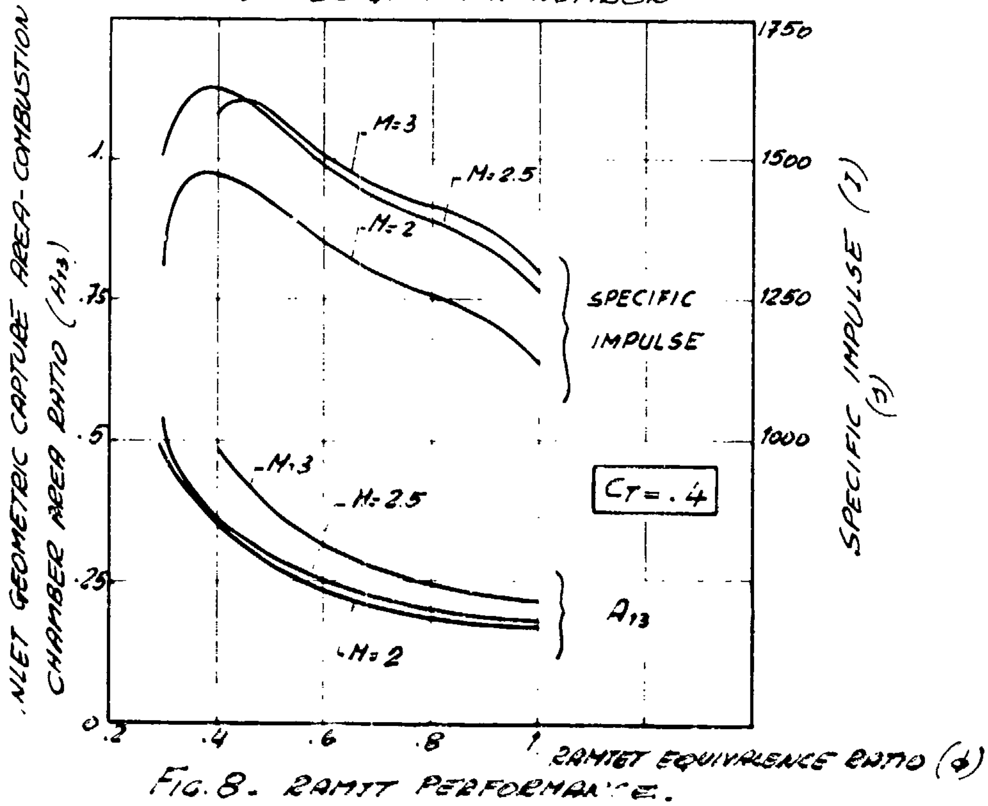

FIG. 8. RAMJET PERFORMANCE.
INLET GEOMETRIC CAPTURE AREA AND MAXIMUM SPECIFIC IMPULSE VERSUS THE EQUIVALENCE RATIO, AS FUNCTION OF DESIGN MACH NUMBER

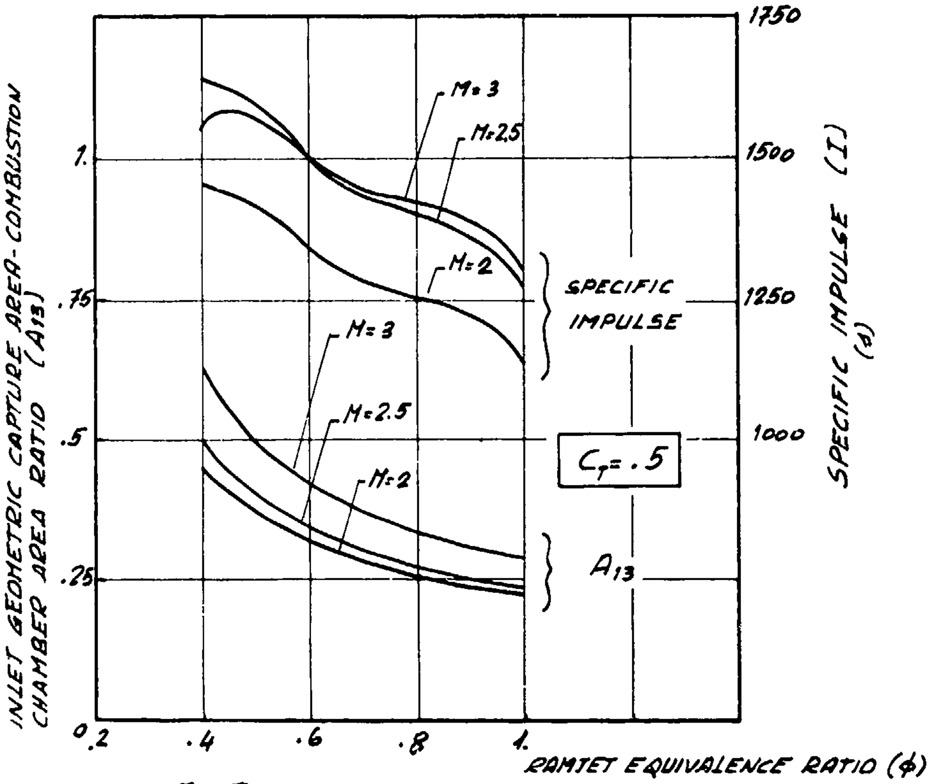

FIG. 9 RAMJET PERFORMANCES.
INLET GEOMETRIC CAPTURE AREA AND MAXIMUM SPECIFIC IMPULSE VERSUS THE EQUIVALENCE RATIO, AS FINCTION OF DESIGN MACH NUMBER

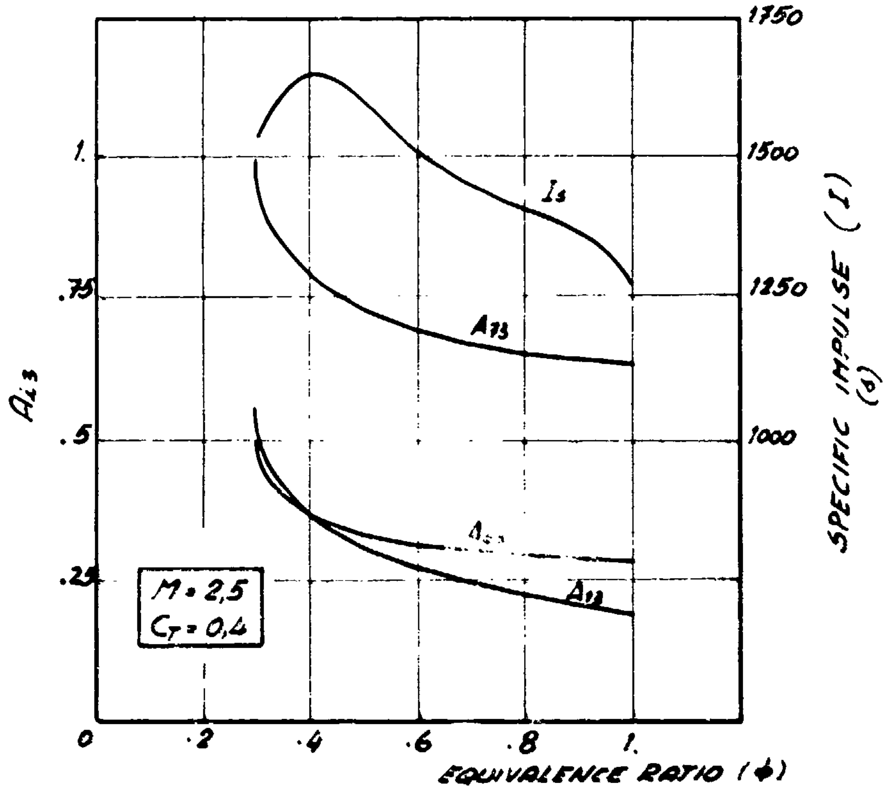

FIG. 10. RAMJET PERFORMANCES, INLET AREA, NOZZLE THROAT, NOZZLE EXIT AREA AND MAXIMUM SPECIFIC IMPULSE VERSUS ϕ.

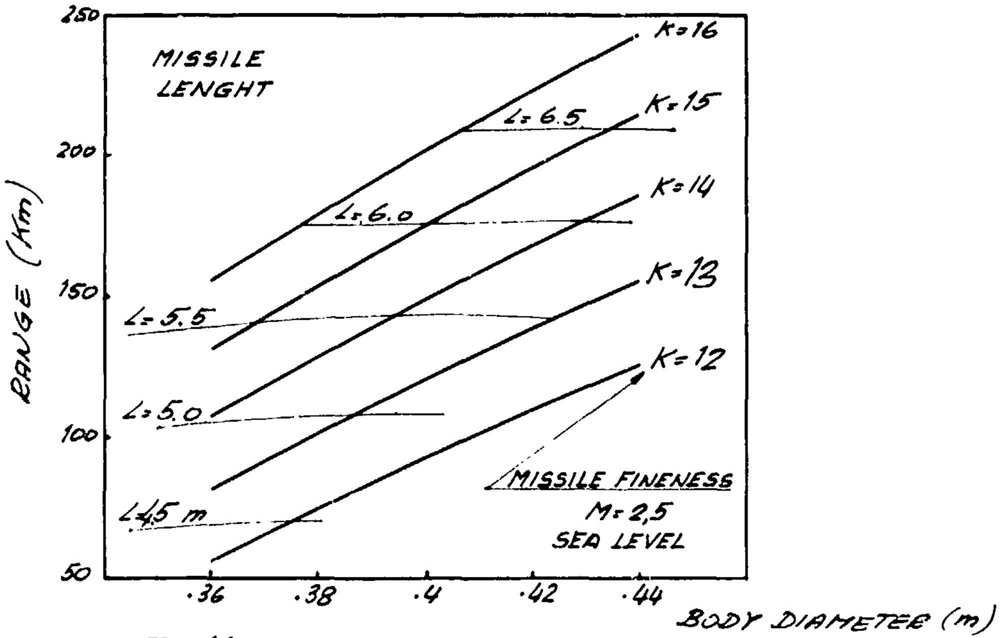

FIG. 11. RAMJET RANGE VERSUS MISSILE DIAMETER AS A FUNCTION OF FINENESS

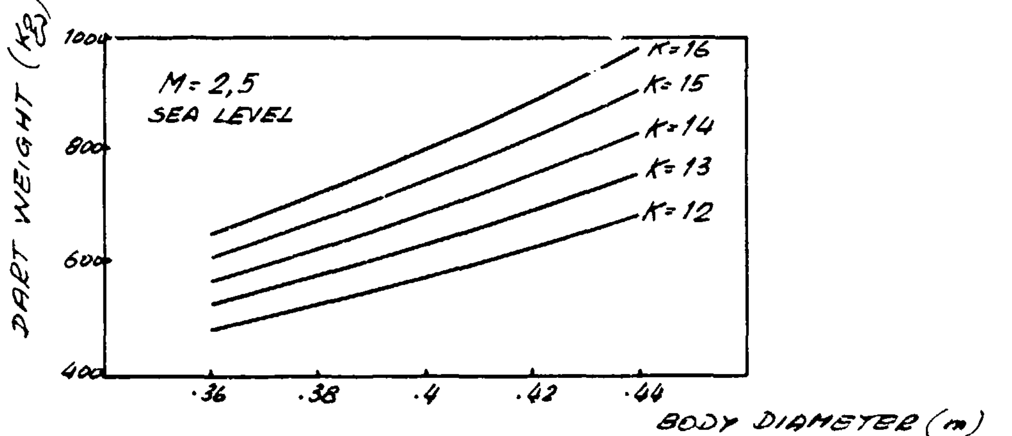

FIG. 12. RAMJET POWERED MISSILE DART WEIGHT VERSUS BODY DIAMETER

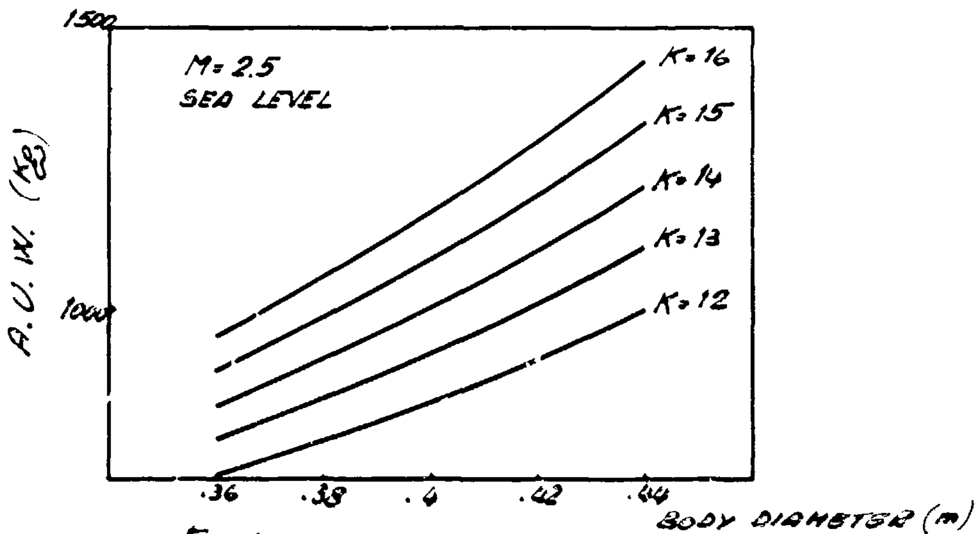

FIG. 13. RAMJET POWERED MISSILE A.U.W. VERSUS BODY DIAMETER

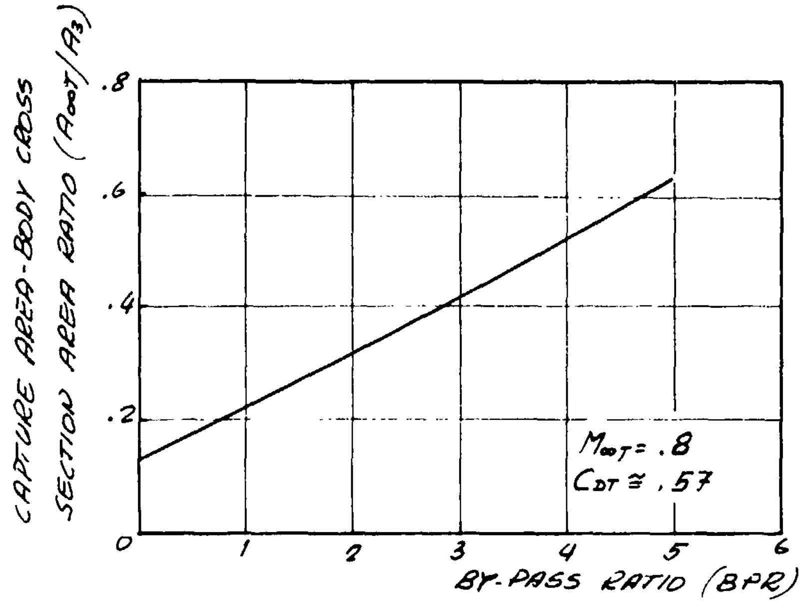

FIG. 14 - TURBO CAPTURE AREA-BODY CROSS SECTIONAL AREA RATIO VERSUS BY-PASS RATIO.

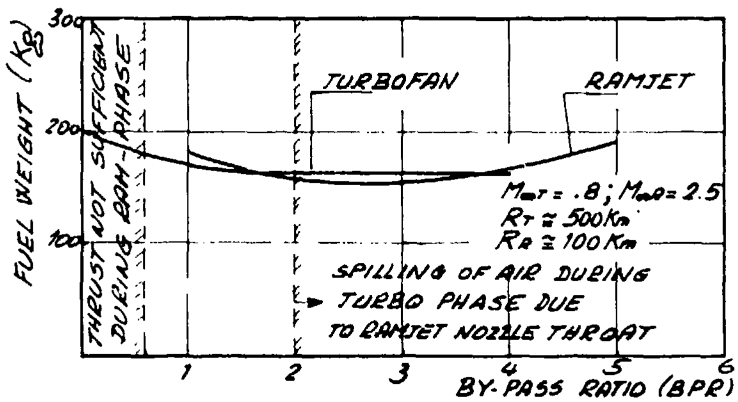

FIG. 15. TURBOFAN AND RAMJET FUEL WEIGHT AS A FUNCTION OF BPR. LIMITS OF MATCHING THE TWO ENGINES

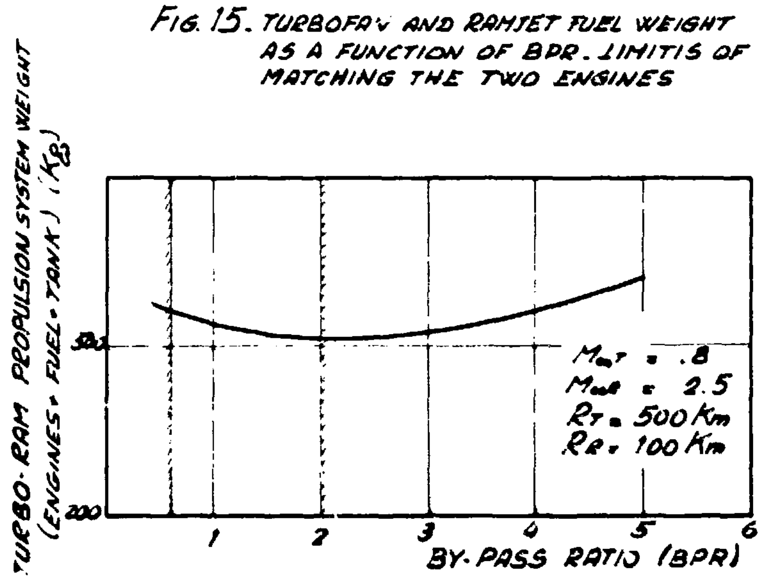

FIG. 16 - OVERAL TURBO-RAM PROPULSION SYSTEM WEIGHT VERSUS BY-PASS RATIO

BPR	$(SFC)_T$ Kg/h/dN	$(T/W_a)_T$ dN/kg/s	D_T (dN)	C_{H_T} Kg/h	W_{fuel_T} (Kg)	W_{turbo} (Kg)	W_{a_T} Kg/s	A_{13}	$\varnothing_R$	I_{s_R} (s)	W_{fuel_R} (Kg)
0	1.25	57	310	390	199	40	5.43	0.13	---	---	---
1	1.07	34	315	338	175	50	9.26	0.22	0.8	1410	183
2	1.02	23.5	319	328	167	65	13.570	0.323	0.5	1600	161
2.5	1.0	20	321	326	166	75	16.05	0.37	0.4	1650	160
3	0.99	18.2	324	324	165	85	17.8	0.423	0.37	1640	157
4	0.97	14.9	329	321	164	100	22.1	0.526	0.3	1520	170

TABLE 1

CONVERTIBLE FAN SHAFT ENGINE

(FOR ROTARY WING AIRCRAFT)

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INTRODUCTION

For years, the helicopter has been the only practical vertical takeoff and landing (VTOL) aircraft configuration. Even experimental high-disc-loading VTOL aircraft were not possible until development in propulsion technology permitted acceptable thrust-to-weight ratios. The increased horsepower to weight and the lower specific fuel consumption of the current generation of turboshaft engines have made significant improvements in helicopter performance and operating envelope.

The convertible fan shaft (CFS) engine, utilizing lightweight-fan and advanced turboshaft-engine technology, is proposed as a propulsion system for a compound helicopter that can potentially double the operating envelope of present rotary wing aircraft. Significant improvements in future VTOL aircraft depend upon propulsion technology and the proper tradeoffs between the hover and cruise requirements.

The compound helicopter's preformance capabilities and propulsion requirements are briefly reviewed here. The AH-56 Cheyenne aircraft has been used as a baseline to evaluate the CFS engine (Figure 1).

The views expressed are those of the author and not the US Army.

CONVERTIBLE FAN SHAFT ENGINE

The CFS engine is a combination of a high-bypass fan coupled with a turboshaft engine to provide auxiliary thrust capability for a rotary wing aircraft. As with all VTOL propulsion systems, the CFS engine must be designed to satisfy the hover horsepower requirement for the hover condition, while at the same time, providing adequate thrust and shaft power for the cruise or maximum speed requirement at the specified altitude and temperature. The sketch of the CFS engine shown in Figure 2 highlights the significant components: the high bypass fan, the gearbox with a shaft output to the main rotor gearbox, the turboshaft engine, and the integrated control system.

The Army conducted a number of CFS engine studies in the late 1960's. These studies were based on utilizing turbofan bypass ratios of approximately 3 to 12. These CFS engine investigations were based primarily on a stopped, stowed-rotor aircraft configuration and included the engine complexity to accomplish the transition to the stopped-rotor mode. Subsequently, prop-fan studies and hardware investigations have developed data for higher bypass ratios, which are more efficient for the 200-to-350 knot speed range. The CFS engine design goals for this investigation are based on the prop fan performance found in Reference 1 and the engine configuration defined in Reference 2.

The prop fan for the 220-knot maximum speed compound helicopter will have design pressure ratios in the 1.06 to 1.10 range with a total activity factor of 1000-1200 (solidity approximately .3). The 300-knot maximum speed aircraft will utilize pressure ratios on the order of 1.15. The fan configuration will be designed by cruise and high-speed requirements; the lower pressure ratio and solidity (less than 1.0) should permit reverse thrust, with blade pitch change in a conventional propeller mode (decreasing blade angles or through the low-pitch mode). The compound aircraft would require the capability of operating with varying inflight negative-net-thrust demands to control dive angle and rapid deceleration. The QFT-55 performance under reverse thrust conditions (Reference 3) has demonstrated acceptable inlet performance and fan-blade stresses with increasing blade angles through the feather mode, which is more severe. The root of the blade must be compromised for supercharging, particle separation and reverse operation. Erosion protection will be required for the nacelle lip, blades and stators, particularly in a rotary-wing hover environment.

During take-off or the hover mode, the fan will be at a low pitch setting to minimize the power absorption, which is estimated to not exceed 10-15% of take-off power. During transition flight, 60 to 80 knots, thrust will be provided by increasing the fan pitch, and the power to the main rotor will be decreased. For cruise and high-speed flight, the rotor horsepower will be held approximately constant; the fan shaft spool will operate at a constant speed. Higher-speed compound aircraft will require 10-20% reduced fan spool rpm to provide reduced rotor tip speeds for non-stowed rotor aircraft.

The CFS engine can provide inherent inlet particle separation, IPS, with the large swirl imparted at the low blade angles in the shaft mode of operation. A preliminary IPS assessment conducted during the CFS engine design of Reference 2 confirmed the potential. The accessory gearbox drive arrangement can be extended to the fan shroud/cowling by a jack shaft or engine mounted in a conventional turboshaft configuration.

The performance of the CFS engine used for the AH-56 comparison is based on the prop-fan performance given in Reference 1 and the engine cycle parameters of approximately 13:1 pressure ratio with a 2000°F turbine inlet temperature. The turboshaft engine configuration utilized in Reference 2 has been used for dimensional scaling: a five-stage axial and a one-stage centrifugal compressor, an annular combustor, a two-stage gas-producer turbine and a two-stage power turbine. The CFS engine characteristics for the three fan diameters considered for the comparison are shown in Table 1.

The CFS engine control system must provide constant-speed fan spool operation during hover and cruise flight. A cockpit control input device to change fan blade angle can be utilized to establish the desired cruise speed, similar to the AH-56 collective twist grip. The control system must include an automatic input into the pitch-change mechanism to minimize fan power absorption for autorotation or an inflight power loss condition. Power management, or the proper use of rotor and propeller capabilities, was cited as the biggest challenge for the pilot in transitioning into the AH-56 aircraft; independent lift and drag capabilities provide a wide range of flight possibilities. An electronic engine-control system will provide an effective interface with an automatic rotor-control system, if desired, to minimize the pilot's workload.

COMPOUND HELICOPTER CHARACTERISTICS

In order to understand the compound helicopter auxiliary thrust requirements, a brief review of experimental aircraft studies is in order. Rotary-wing aircraft may be limited by power, stress, or vibration; however, Reference 4 makes a strong point that the crew's concern for vibration levels is the usual practical limiting condition. The experimental compound helicopter flight test data demonstrates a significant improvement in the vibration levels as shown in Figure 3. The compound helicopter mode allows speed increases of 60 to 100 knots over the conventional rotary-wing mode for the same level of vibration. The government test pilot's evaluation of the NH-3A concluded that "the aircraft was unusually smooth throughout the flight envelope flown, including high speed and approach to hover" (Reference 5). In addition to the increased speed capability, there is a significant increase in the load-factor capability, which is essential for the use of higher speed capability in maneuvers and terrain avoidance. Figure 4, which is from Reference 6, shows the XH-51 compound- and pure-helicopter flight test results. The AH-56 envelope was expanded to 2.6g positive and .2g negative in the 160 to 180 knot range; there were no limiting characteristics.

The compound rotocraft experimental aircraft investigations reflected that auxiliary propulsion, with or without wings, can improve the vibration levels and relieve control-sytem vibratory-stress conditions. The NH-3A flight test results shown in Figure 5 reveal a more acceptable rate of control loads build-up with high speeds with auxiliary thrust or with auxiliary thrust plus wing than with a pure helicopter thrust alone. The rotor loads at advance ratios above .40 were strongly dependent on the rotor propulsive forces required for the UH-1 compound.

The wing and the increased power available for the higher-speed rotary wing aircraft increases the altitude-velocity envelope until the wing-induced drag becomes dominant. The compound helicopter has a significant increase in operating envelope, as shown in Figure 6 (from Reference 11), over the standard helicopter. The performance shown is for a compound helicopter that was designed for a 10,000-foot-altitude, 250-knot cruise condition.

The addition of an auxiliary thrust device on a helicopter also permits a significant improvement in its capability as a weapons platform; a stable hover can be maintained through a 22-degree range of pitch attitude. In maneuvering in close quarters in the NOE environment, the auxiliary thrust allows acceleration and deceleration without severe pitching attitudes and rates.

The AH-56 performance data will be reviewed to gain an insight into the power and thrust requirements for a 220-knot compound helicopter. The AH-56A rotor and propeller performances are shown in Figure 7 for a fixed collective position above approximately 80 knots. Figure 8 shows the horsepower required by the main rotor and propeller as a function of airspeed. Note that the horsepower required by the main rotor is approximately 750 horsepower and remains constant with speed. The propeller power continues to increase to offset the drag of the vehicle. The forward-flight performance for rotor thrust and the propeller thrust is shown in Figure 9. The aircraft is flown with a constant collective pitch as a fixed wing aircraft above 60 to 80 knots. The lift-to-drag ratios are shown for the NH-3A and AH-56A in Figure 10. The NH-3A data includes two aircraft configurations, with wings and without wings, and two different rotor collective pitch settings (i.e., rotor horsepower levels) are shown for the winged configuration. The rotor hub/fuselage interference drag is a significant drag area that can be improved; the main rotor head drag was the largest single item for the NH-3A estimate (9 ft^2 out of 35 ft^2). An Army research and development program is underway to improve the main rotor hub drag for future installations.

The flight speed for various rotary wing aircraft configurations is shown in Figure 11. The Bell UH-1 HPH compound helicopter speed of 274 knots established the speed record for the Army compound helicopter experimental work in the mid to late 1960's. The UH-1 HPH tail rotor loads were critical at high speeds; excitations from the main rotor vortices and the hub and rotating-control wakes were the probable causes. The tail rotor problems would strongly suggest the use of the ducted fan directional control approach. The NH-3A and AH-56A flight test maximum speeds were in the 200 to 220 knot range. The practical limit for an unloaded rigid rotor has been estimated at approximately 400 knots (μ of 1.6 combined with a tip speed of .98 Mach) (Reference 4). Although a maximum flight speed of approximately 400 knots could theoretically be achieved for a slowed rotor compound, the stowed rotor configuration would offer a more practical alternative at speeds approaching 400 knots (See Table 3).

The AH-56A flight capabilities used the propeller for a speed brake, permitting steep approaches and dives. Reference 7 describes an entry into a 35-degree dive at 1500 feet at 200 knots with the propeller being used as a "brake" to slow to 110 knots. A recovery was made at 100 feet, followed by a normal roll on landing. Thus, a compound helicopter propulsion system must permit the use of the thrust device in a negative thrust mode to provide the maximum versatility for the aircraft.

The most recent parametric design investigation of compound helicopter configurations considered the use of a variable-diameter rotor system (Reference 8). Helicopter, compound, slowed-rotor compound, variable-diameter rotor aircraft, and stowed-rotor configurations were assessed for the following mission: 5000-pound payload with a 4000-ft, 95°F hover and a 350-nautical mile range. A summary of the vehicle characteristics are shown in Table 2. The rotor systems were designed to minimize the rotor system weight fraction and utilized the horsepower installed for the design cruise condition for hover. The higher-speed aircraft generally had increased productivity, which offset their higher gross weights. The advancing blade concept (ABC) preliminary design data from Reference 9 is tabulated for a sea-level 300-knot maximum speed design requirement. The ABC configuration utilizes two convertible fan shaft engines that have an integral 1.15 pressure ratio 3.98-ft-diameter fan and are rated at 2790 horsepower at sea level static conditions. The breakdown of the total CFS engine weight of 813 lb was estimated to be: 430 lb for the engine, 195 lb for the fan, 86 lb for the fan shroud, and 102 lb for the fan gearbox.

The data establishes a general trend of power loading versus speed for compound helicopters. The AH-56A is the only aircraft for which data is available; its parasite drag characteristics are over 40% higher than the parametric aircraft. CFS engines are used for all the aircraft except the AH-56A and the compound, which both use an aft-mounted pusher propeller.

AH-56A PROPELLER CONVERTIBLE FAN SHAFT ENGINE COMPARISON

It is obvious that the efficiency of the high bypass ratio fan cannot compete directly with the propeller. The propeller is more efficient than the fan over the speed range that could normally be expected from a rotary-wing aircraft. The weight and/or installation advantages of the fan have to off-set the higher efficiency propeller. A twin CFS engine installation was compared with the AH-56A propeller installation and the results are shown in Figure 12. It is recognized that the speed range of this comparison presents a difficult challenge for the CFS engine.

The AH-56A parasite drag was reduced from 22.5 ft^2 to 16.5 ft^2. The auxiliary thrust required for a sea-level 220-knot maximum speed condition was used to establish three propeller and three cruise-fan configurations. The Hamilton Standard Generalized Method of Propeller Performance Estimation was used to calculate propeller performance. The computer performance program from Reference 1 was utilized for fan performance estimation.

The propulsive efficiencies of the propellers and fans are shown in Figure 12. Constant speed operation was assumed for the propellers and fans; the propeller tip speed was 899 feet per second, and the fan tip speed was 700 feet per second. The maximum engine horsepowers required are inversely proportional to the efficiencies shown at 220 knots. Note the significant improvement in cruise efficiency for the 10-ft-diameter propeller at lower speeds, although the engine size must be 6% larger. The fan efficiencies improve with reducing airspeeds, and the maximum values occur near the speed for best range.

The T64 engine data was scaled to the 220-knot airspeed requirement, and specific range performance data was calculated. The engine size was based on having two CFS engines to provide auxiliary thrust and additional 757 horsepower for the main rotor, 35 horsepower for the tail rotor, and 88 horsepower for accessories. The specific range performance is shown in Figure 13; the actual AH-56A aircraft performance is almost identical to the 3.5-ft-diameter fan performance. Thus, the propeller's specific range is estimated to be 13% higher than that for the fan installation. Supercharging, although practical, was not included in the CFS engine performance. Based on a review of past CFS engine design investigations, the maximum drag estimate for the CFS engine installation was estimated to be two percent up to a Mach Number of .3 (Reference 10). The fan shroud drag was included in the prop fan performance analysis; no additional drag was added for the engine cowling.

The weight data shown in Table 3 summarizes the AH-56A aircraft installation. The CFS engine installation is shown in Table 4. The CFS installation is believed to be conservative based on the review of a number of CFS engine preliminary design installations.

Thus the comparative engine and fuel weight summary for the AH-56A based on a 300-nautical-mile mission is as follows:

CFS ENGINE STUDY

	AH-56A	10-ft Propeller	3.5-ft Fan	4.25-ft Fan
Installation	1512 lb	1512 lb	1752 lb	1848 lb
Fuel	2307 lb	1967 lb	2316 lb	2222 lb
	3819 lb	3479 lb	4068 lb	4070 lb

The 300-nautical-mile range or approximately 2-hour mission represents an almost even trade between the 3.5- and 4.25-ft-diameter fans. The CFS engine installation with fuel is approximately 17% heavier than a pusher-propeller installation. The CFS engine estimates do not include a weight reduction for the potential integration into the fuselage wing root structure.

The sketch shown in Figure 14 depicts a 3.5-ft-diameter fan and a 7-ft-diameter propeller installed on an AH-56A. The pusher-propeller installation is ideal for the AH-56A aircraft; a larger aircraft, a cargo transport of higher-speed capability (fan-in-fin) may tend to offset the aft installation. A wing-mounted propeller would encounter wake interaction effects, and clearance problems, and would limit the launching of stores. Figure 14 also shows a fan-in-fin directional control force system that would eliminate the aforementioned tail rotor problems.

SUMMARY

The compound helicopter experimental aircraft have demonstrated a significant increase in the practical operating envelope for the rotary wing aircraft in airspeed, altitude, and load factor capability. The AH-56A aircraft configuration integrated a propeller auxiliary thrust system into an excellent installation for its gross weight and mission. A convertible fan shaft engine can offer significant potential for a larger aircraft of a higher-speed design.

The CFS engine technology base has been developed to a large degree by the prop fan work accomplished by Hamilton Standard and the NASA Lewis Research Center. Advanced turboshaft engine technology is available to minimize the power plant package.

The CFS engine configuration permits installation versatility in the potential combination of propulsion requirements: Inlet particle separator, auxiliary thrust (for hover-attitude control, low-speed maneuvering, flight-path control and high-cruise speed), shaft power for the rotor and directional control system, and integrated infrared suppression. The practical requirements for icing provisions, thrust reversing, horsepower-versus-thrust modulation, and CFS engine control will present a challenge, but there are rational engineering solutions available.

The CFS engine is an engine concept that provides an integrated thrust/shaft-horsepower propulsion system that has the potential of enhancing the operating capabilities of rotary-wing aircraft. One can predict a demand for efficient hover with increased flight envelope abilities: the CFS engine can meet the versatile compound helicopter requirements.

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TABLE 1

Convertible Fan Shaft Engine Characteristics

FAN DIAMETER FT	ACTIVITY FACTOR	TIP SPEED FT/SEC	220-KNOT SEA-LEVEL DESIGN POINT				SHP SEA LEVEL STATIC	WEIGHT LB
			THRUST LB	SHP	FAN PRESSURE RATIO	FAN BYPASS RATIO		
3.50	1200	700	1758	440	1.09	27.5	1980	701
4.25	1000	700	1758	440	1.06	39.4	1830	741
5.00	1000	700	1758	440	1.045	53.3	1770	826

TABLE 2

Parametric Mission Analysis Comparison

AIRCRAFT CONFIGURATION	DESIGN GROSS WEIGHT LB	CRUISE ALTITUDE SPEED FEET KNOTS	HOVER SHP SLS RATING	INSTALLED SHP SLS	DISK LOADING LBS/FT^2	$\frac{\text{GROSS WEIGHT}}{\text{SHP}}$ LB/SHP	THRUST DEVICE	
							DIA FT	$\frac{\text{HP SLS}}{(\text{DIA})^2}$
HELICOPTER	19223	5000' 175 KNOTS	3617	3617	6	5.31	0	0
COMPOUND	21666	5000' 200 KNOTS	5929	5929	10	3.65	11.0	49.0
SLOWED ROTOR COMPOUND	24654	10000' 250 KNOTS	6821	6821	9	3.61	4.24	189.7
VARIABLE DIA ROTOR COMPOUND	25469	20000' 300 KNOTS	9110	9110	13	2.80	4.22	255.8
VARIABLE DIA ROTOR SLOWED	28718	20000' 350 KNOTS	11398	12792	15	2.25	4.27	350.9
VARIABLE DIA ROTOR STOWED	26662	20000' 400 KNOTS	9387	10145	13	2.63	3.46	423.6
AH-56A	18300	SL 183 MAX	3925	3925	8.87	4.66	10.0	39.3
ABC	16000	SL 277 MAX	4460	5580	12.7	2.87	3.98	176.1

TABLE 3

AH-56A Engine and Propeller Weight Summary

ENGINE SECTION	181.0 lb
ENGINE	692.0
AIR INDUCTION SYSTEM	164.7
ENGINE OIL	44.1
ENGINE CONTROLS	12.1
STARTING	31.0
PROPELLER	333.7
DRIVE SHAFT WEIGHT REDUCTION	53.0
	1511.6 lb

TABLE 4

CFS Engine Weight Summary

DIAMETER	3.5 ft	4.25 ft	5.0 ft
ENGINE	440	407	394
FAN	192	256	348
GEARBOX	69	78	88
CFS ENGINE	701	741	826
INSTALLATION	140	148	165
STARTING	25	25	25
CONTROLS	10	10	10
	876 lb	924 lb	1026 lb
	x 2	x 2	x 2
	1752	1848	2052

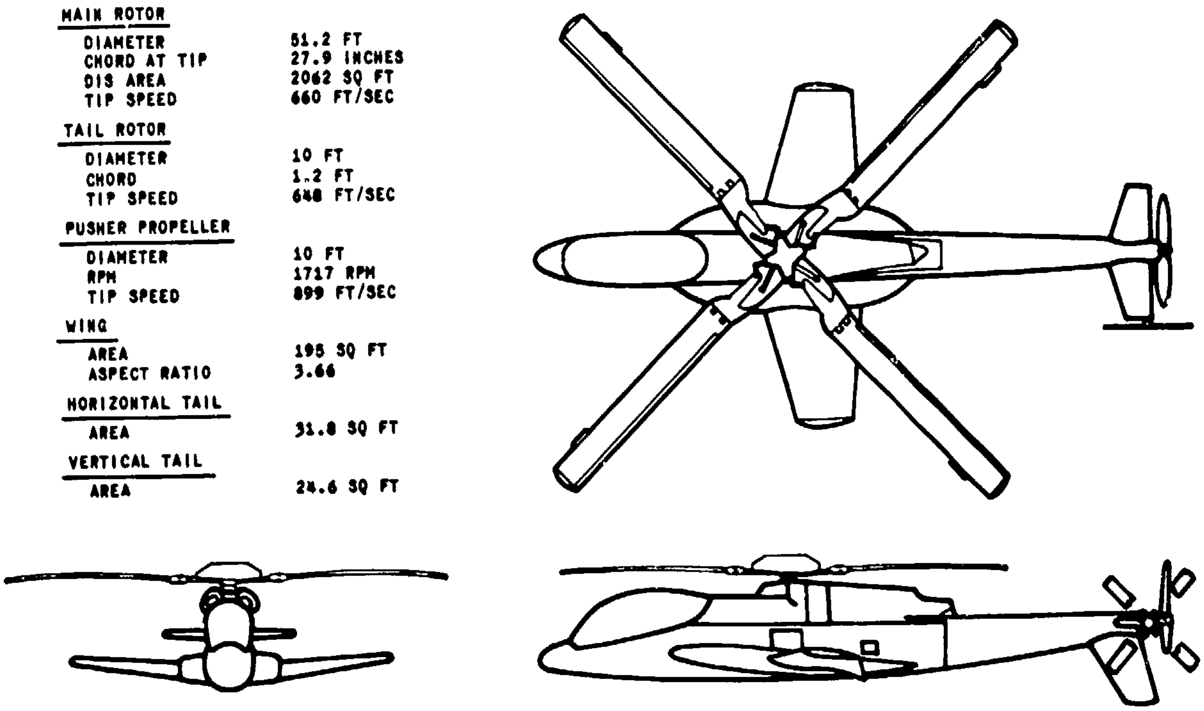

Fig.1 AH-56A Baseline compound helicopter

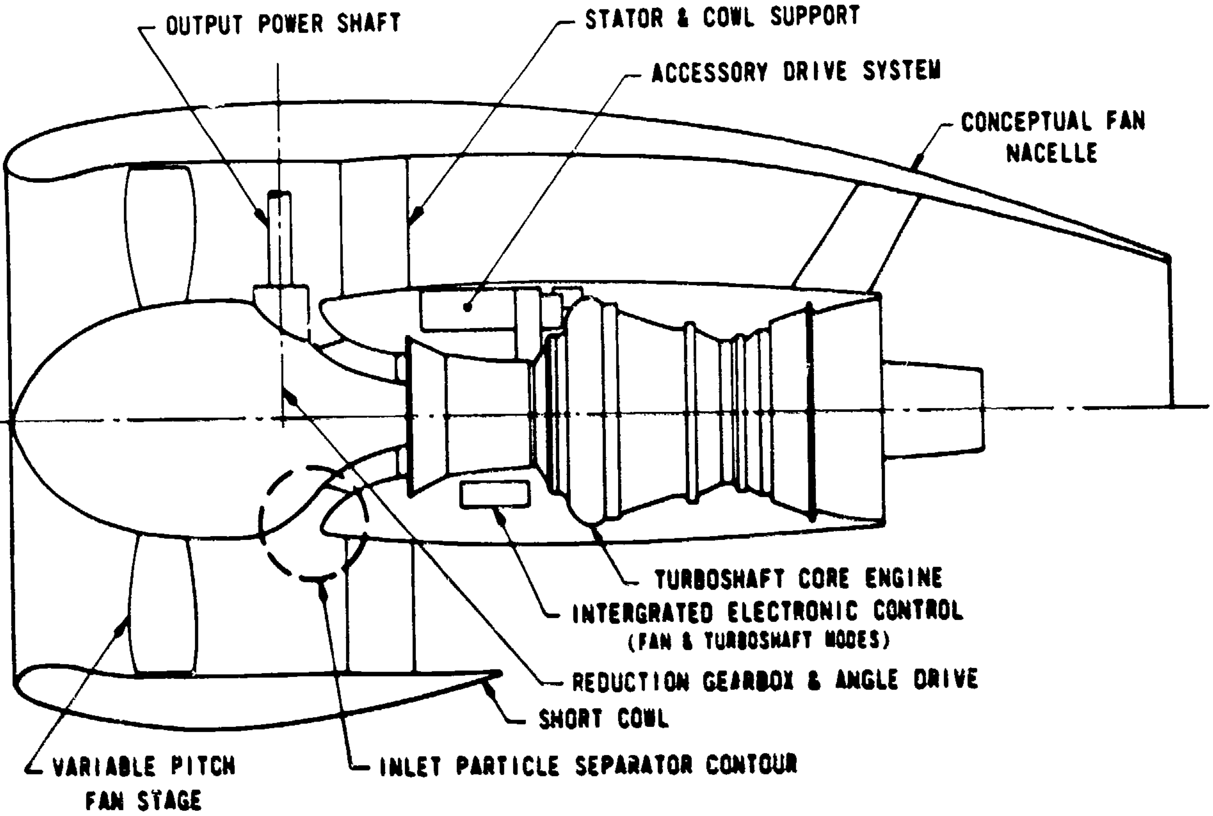

Fig.2 Convertible fan shaft engine

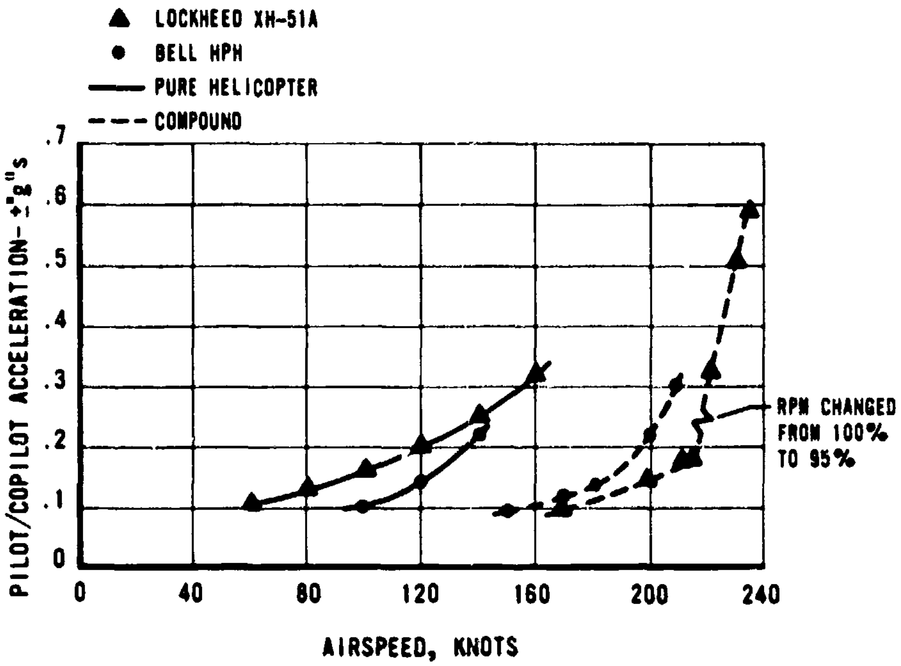

Fig.3 A correlation of crew-experienced vibration with speed

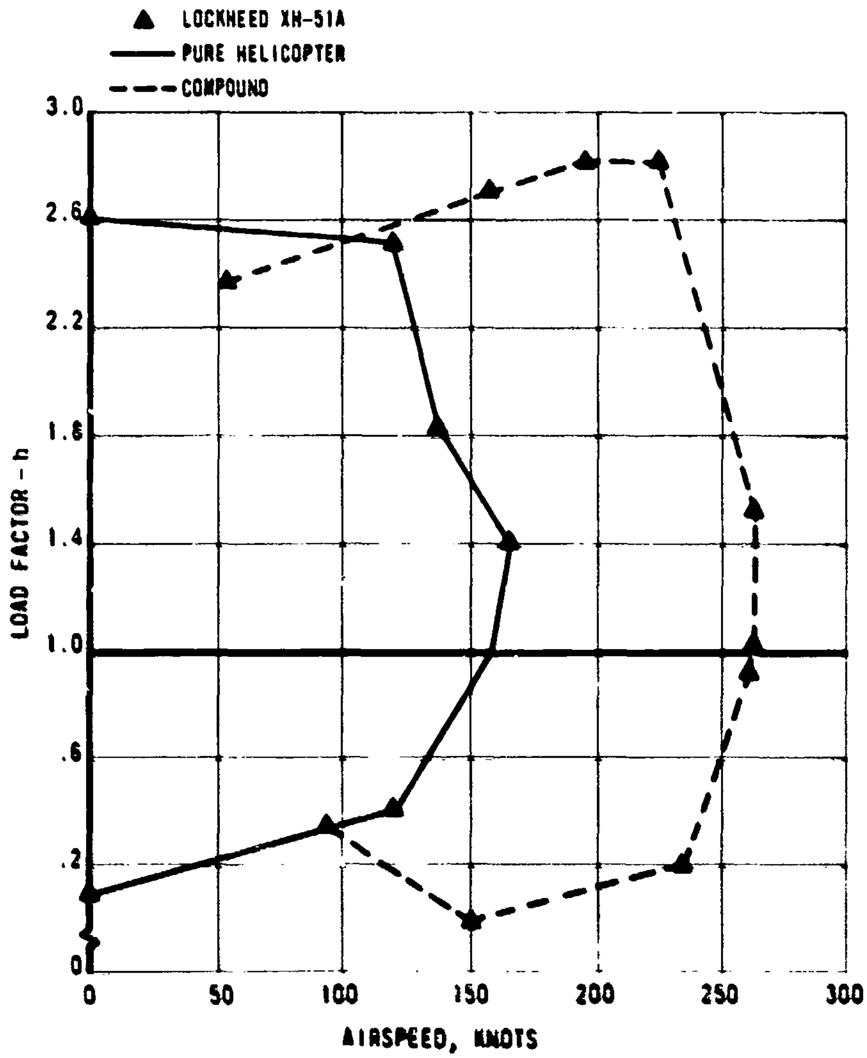

Fig.4 Load factor capabilities of the compound and pure XH-51A helicopters with respect to speed

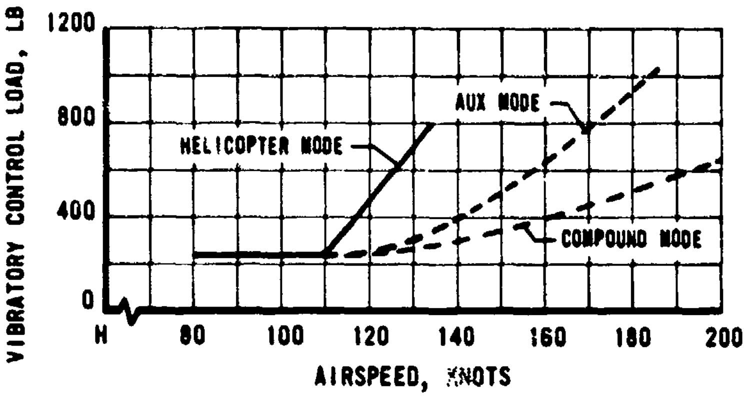

Fig.5 Auxiliary thrust and wing effect on control loads

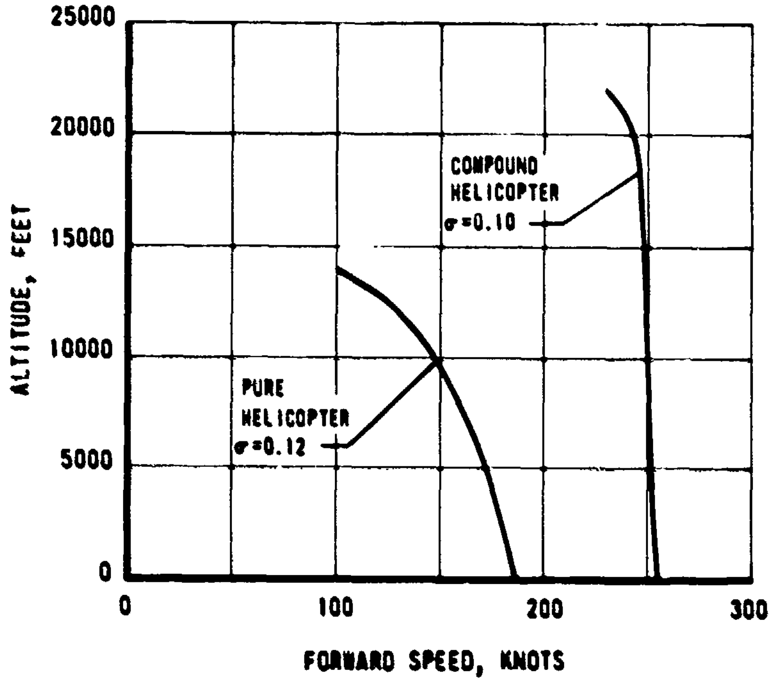

Fig.6 The altitude versus speed operating envelopes of the pure and compound helicopters

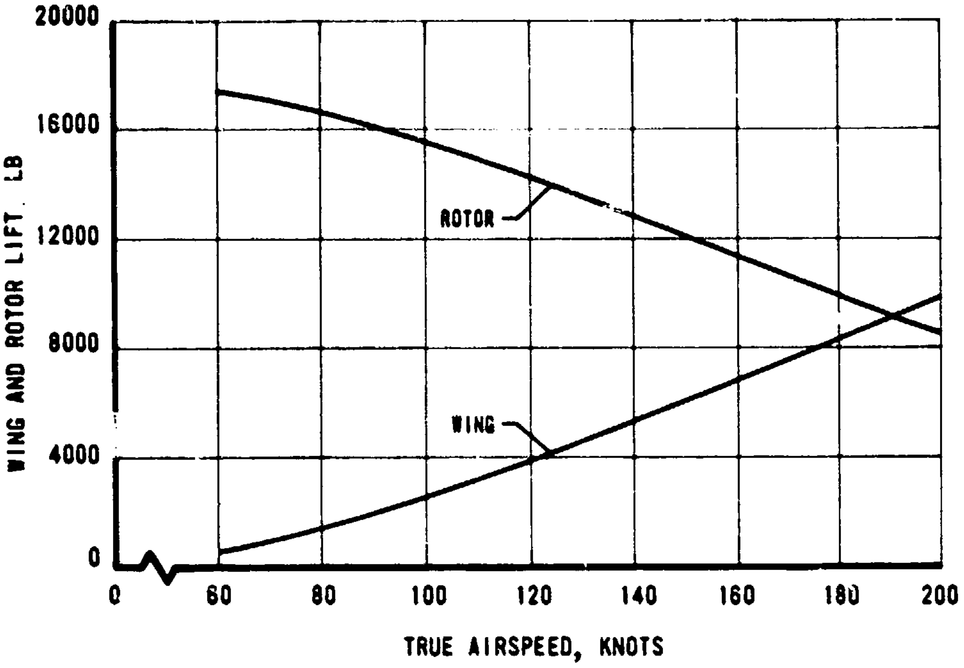

Fig.7 Wing and rotor lift components of the AH-56A

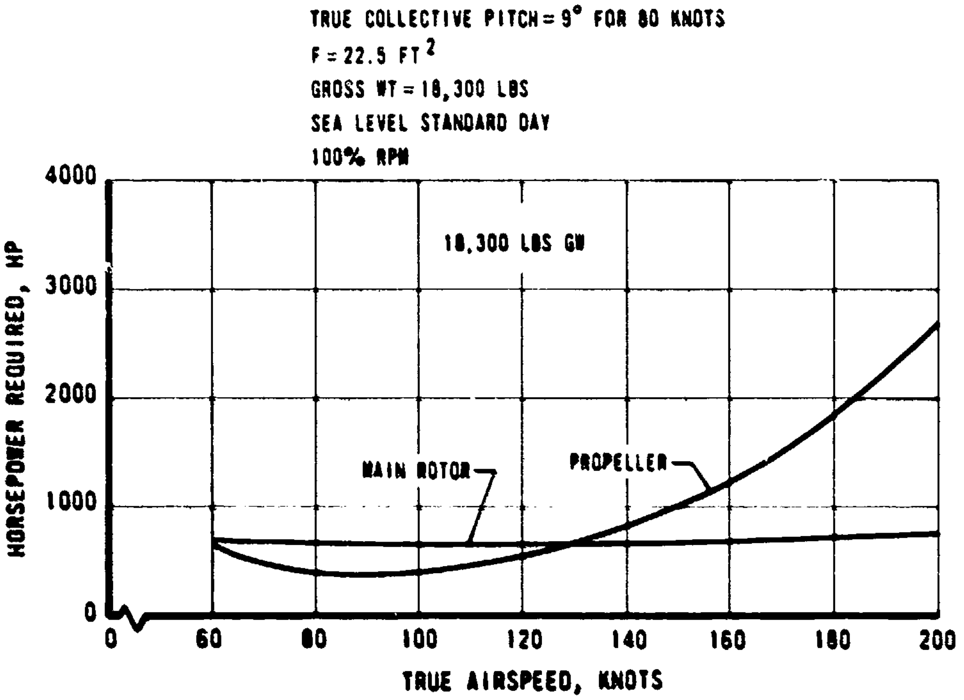

Fig.8 Propeller and main rotor power requirements of the AH-56A

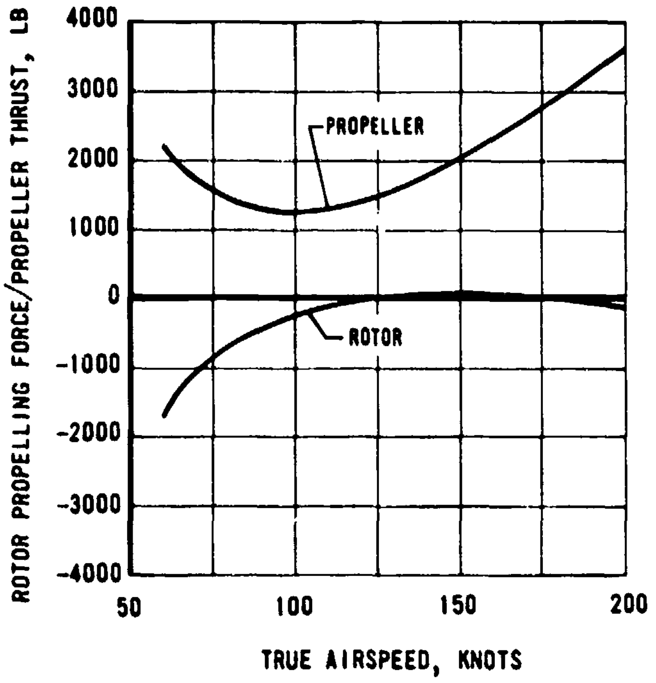

Fig.9 Propeller thrust and rotor propelling force requirements of the AH-56A

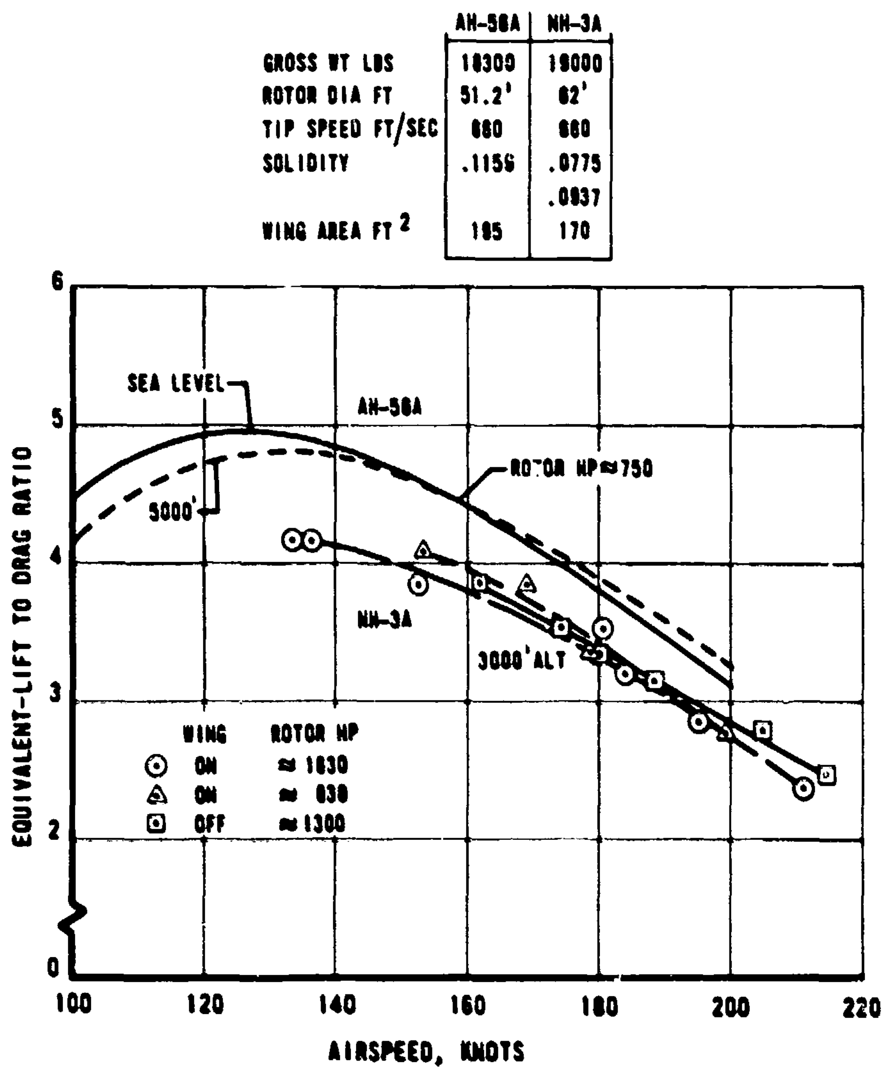

Fig.10 Equivalent lift to drag ratios for the AH-56A and NH-3A.

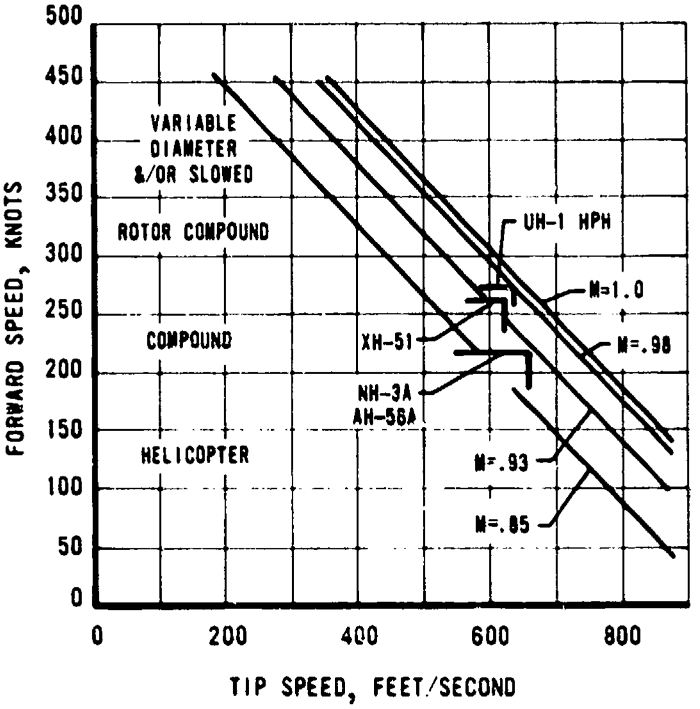

Fig.11 Limits to forward flight speed

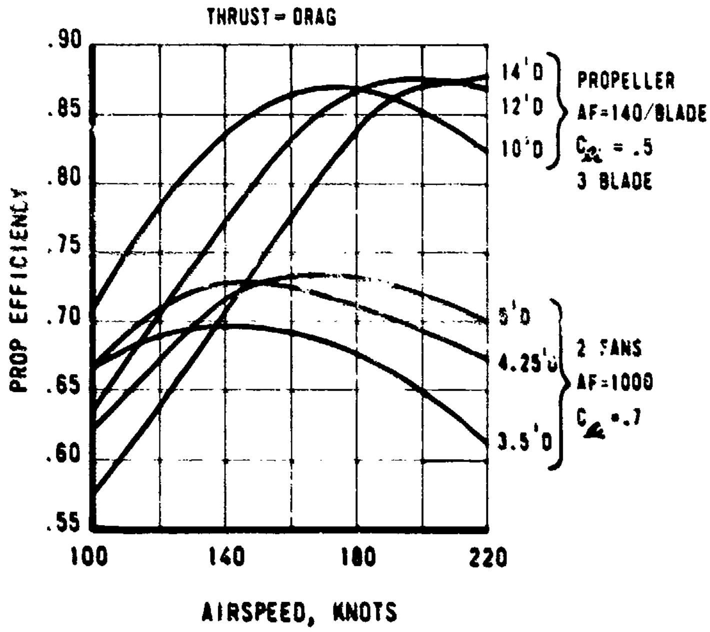

Fig.12 Comparison of the efficiencies of fan and propeller configurations

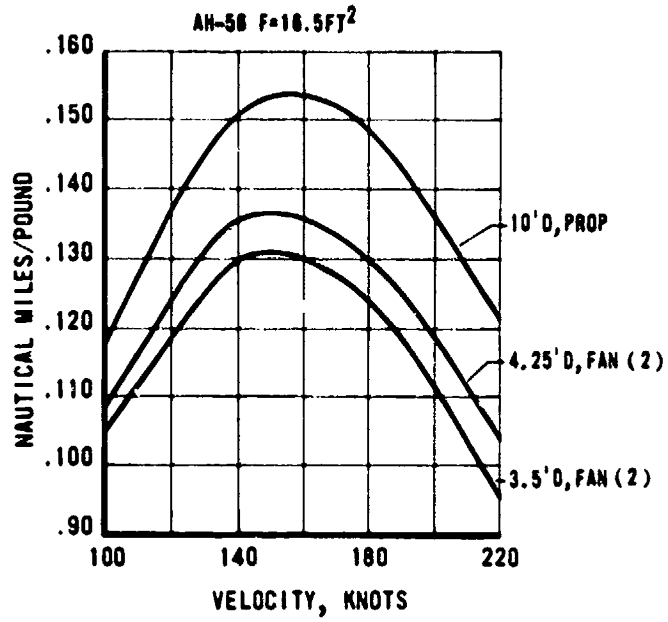

Fig.13 Comparison of the specific range of propeller and fan configurations

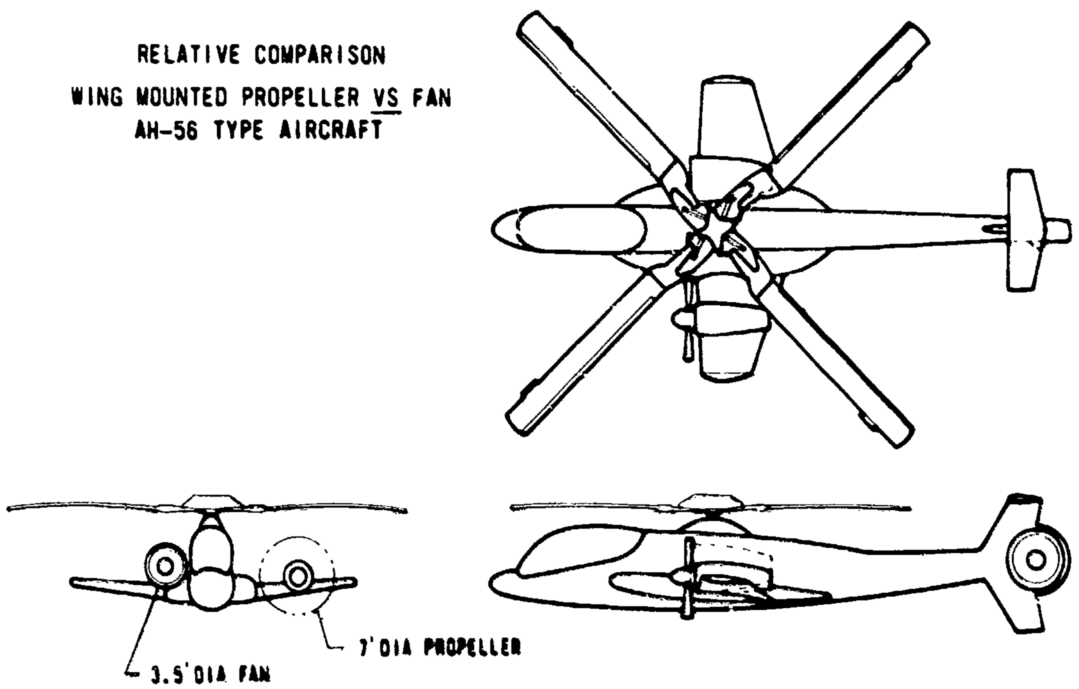

Fig.14 A comparison of a wing-mounted propeller with a fan on a AH-56

DISCUSSION

A.H.Jackson

This is more a comment than a question. The prop fan we talked about yesterday with very high solidity highly power loaded free prop would match very well with what you are describing here. The diameter would not be any significantly bigger than the variable pitch fan you are showing at that pressure ratio you have there. The prop fan does not do too well at static because it stalls but when you get up to the 200 knots range its efficiency ought to get well up and it still maintains a very compact installation because the blades would include very low camber. This is typical to the prop-fan.

P.Langdell

(1) Would the author explain the power plant configuration of the proposed CFS aircraft: in particular, is the main rotor driven by the forward propulsion engines or is it separately powered?

(2) Is it envisaged that the proposed CFS aircraft will be capable of hovering with one main engine inoperative.

Author's Reply

(1) The power plant configuration for the convertible fan shaft engined aircraft utilizes a conventional free power turbine gas generator with the power turbine output shaft directly coupled to a variable pitch prop fan and directly coupled to the main rotor gear box through an overrunning clutch. Thus, the same gas generator is used to drive the main rotor in hover and the prop fan for cruise and high speed. This shaft arrangement permits a variable power split between the main rotor and propeller such as is shown in Figure 8 of the paper.

(2) The proposed convertible fan shaft aircraft would not be capable of hovering with one engine inoperative at the design gross weight; however, after partial expenditure of the mission fuel and the payload, the aircraft could hover in ground effect on one engine and/or perform a rolling takeoff.

AN APPLICATION FOR VARIABLE INLET GUIDE VANES IN DISTORTION SUPPRESSION

by

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SUMMARY

The Paper considers the effects of a distortion upon compressor performance and discusses a design technique for inlet guide vanes to decrease the effect of any circumferential pressure distortion. In the instance of a predicted circumferential distortion, a method of selectively varying the stagger of a sector of the guide vane row to minimise the effect of change of rotor incidence created by this class of distortion is investigated analytically. Design charts for both cases are included in the text.

SYMBOLS

A_1 - Elemental aerodynamic inlet area
A_{th} - Cascade channel areodynamic throat area
c - Blade chord
C_{ax} - Axial velocity component
C_L - Lift coefficient
E - Generalised fluctuating lift function
G - Mass flow rate
g - Gravitational constant
i - Rotor incidence
L - Lift
ℓ - Width of a prescribed disturbance
M_u^* - Mach number based upon blade speed
m - Mean value
n - Integer = 1, 2, 3, ...
P_1^* - Inlet stagnation pressure
P_1 - Inlet static pressure
R - Universal gas constant
S(ω) - The Sears function
T_1^* - Inlet stagnation temperature
T(ω) - Function obtained by Horlock
t - Time
U - Blade speed
u - Rotor inlet relative air velocity
y - Tangential linear co-ordinate
v - Transverse component of incremental velocity

α - Guide vane exit air angle
β - Rotor inlet air angle
γ - Ratio of specific heats
ζ - Vorticity
ν - Frequency of distortion
λ - Stagger angle
ρ - Density of air
τ - Inlet pre-rotation
ϕ - Axial velocity ratio
$\Delta\phi$ - Distortion level
$\Delta\phi_{on}$ - Fourier coefficient
ω - Reduced frequency

SUFFICES

d - in the distorted flow region
t - transverse component
c - chordwise component
n - integer = 1, 2, 3, ...
q.s - Quasi-steady
n.s - non-steady

1.0 INTRODUCTION

The overall effects on the performance of gas turbines of distorting the inlet flow to the compressor is well known. In general, a reduction in performance is accompanied by reduced surge margin which can, in certain circumstances, lead to a surge condition. The distortion also creates unsteady cyclic loading on rotor blades which can accelerate fatigue, leading to mechanical failure. Further, the cyclic loading may act as a noise generator.

Of the various classes of distortion that contribute to the complex distortion patterns experienced in compressors that which is normally reckoned to make the major contribution to reduced performance is the circumferential distortion. The circumferential pressure distortion in compressors has been the subject of much modelling and experimental observation. Typically, it may be encountered within intakes downstream of struts, or may be the major component in a distortion created by a separated flow due to a high angle of yaw or incidence to the intake. For the installation of ducted fans or propulsors in light aircraft, pusher configurations are frequently used and a circumferential inlet pressure distortion is inherent in the design.

Based upon two-dimensional modelling of the rotor incident flow by velocity triangles, the purpose of this Paper is to examine, firstly, the effects of both a distortion and the associated time rate of change of rotor incidence upon a compressor stage with a prescribed inlet guide vane geometry. Secondly there is an examination of a method by which inlet guide vanes may be selectively re-staggered around the compressor circumference to reduce or eliminate the change of rotor lift due to a circumferential distortion which is steady in nature.

2.0 EFFECTS OF CIRCUMFERENTIAL DISTORTION

2.1 Alteration of Incidence

Assuming that, in passing through a row of inlet vanes a circumferential distortion is sensed as a deficit in axial velocity and there is no change in deviation of the inlet guide vane flow, there is from the velocity triangle, (fig 1).

$$\tan\beta = \frac{\phi}{1-\tau} \quad \text{at the design condition} \tag{1}$$

$$\tan\beta_d = \frac{\phi_d}{1-\tau_d} = \frac{\phi - \Delta\phi}{1-(\tau-\Delta\tau)} \quad \text{which accounts for distortion} \tag{2}$$

and $$\frac{\Delta\phi}{\Delta\tau} = \frac{\phi}{\tau} = \tan\alpha \quad \text{for no change of deviation at the guide vanes} \tag{3}$$

Substituting equation (3) into (2)

$$\tan\beta_d = \frac{\phi - \Delta\phi}{1 - \tau\left(1-\frac{\Delta\phi}{\phi}\right)} \tag{4}$$

The change of rotor incidence due to the distortion yields

$$\tan \Delta i_d = \tan(\beta - \beta_d) = \frac{\tan\beta - \tan\beta_d}{1+\tan\beta\,\tan\beta_d}$$

By use of equations (1) and (4) the change of incidence may now be resolved in terms of the axial velocity ratio, guide vane exit whirl velocity and distortion change of axial velocity.

$$\Delta i_d = \tan^{-1}\left(\frac{\phi\left(\frac{\Delta\phi}{\tau}\right)}{\left(1-\frac{\Delta\phi}{\phi}\right)\left(\phi^2 + (1-\tau)^2\right) + \frac{\Delta\phi}{\phi}(1-\tau)}\right) \tag{5}$$

For a range of values of axial velocity ratio, the change of incidence due to a distortion is plotted in fig. (5) as a function of the distortion intensity parameter $\frac{\Delta\phi}{\phi}$, and the pre-rotation τ.

Reduced distortion sensitivity may, in general, be associated with low rotor incidence change and it is seen that, at reduced levels of axial velocity ratio, the incidence change is at its lowest. Unfortunately, this does not, in general, help the compressor designer because a reduced axial velocity ratio implies a low mass flow per unit frontal area for the compressor. It may however be concluded that, because of the higher blade speed, tip sections of rotors are likely to be less sensitive to distortion than hub sections.

It is further observed that, at all levels of axial velocity ratio, reduced pre-rotation leads to reduced incidence change due to distortion and preferred geometries would have negative pre-rotation at the rotor. Negative pre-rotation types of design tend however to have a high degree of reaction. This may not be a design disadvantage since experiments (Ref. 1) have indicated that, due to a combination of rotational and viscous effects upon rotor rows, reactions significantly greater than 50% could be advantageous. A further result of negative pre-rotation is that rotor inlet velocities are high and in a highly loaded compressor, transonic flows, with their associated problems, would be inevitable. Such a type of design would however demand a close investigation of rotor choking behaviour.

2.2 Mass Flow Criterion

It is often convenient to determine the design mass flow of a compressor by choking the first stage rotor. The choking mass flow through a rotor element is given (Ref; 2) by:

$$\left[\frac{G\sqrt{T_1^*}}{P_1^*}\right]_{MAX} = A_{th}\left[\frac{2}{\gamma+1}\right]^{\frac{1}{2}\frac{\gamma+1}{\gamma-1}}\sqrt{\frac{g\gamma}{R}}\left(1+\frac{\gamma-1}{2}.M_{u_1}^{*2}\;(1-2\tau_1)\right)^{\frac{1}{2}\frac{\gamma+1}{\gamma-1}} \qquad (6)$$

where $M_{u_1}^* = \frac{U}{\sqrt{g\,\gamma\,R\,T_1^*}}$ and is a Mach number based upon blade speed and stagnation inlet temperature.

Equation (6) holds for any geometry of rotor blade. Considering for example a rotor cascade of infinitely thin, flat plate aerofoils, the throat area A_{th} is related to the elemental inlet area A_1 through the stagger angle λ by the equation:

$$A_{th} = A_1 \operatorname{Cos} \lambda \qquad (7)$$

For this cascade geometry the choking mass flow per unit frontal area of the flow is given (from equation (6)) by:

$$\left[\frac{K\sqrt{T_1^*}}{P_1^*}\right]_{MAX} = \left[\frac{2}{\gamma+1}\right]^{\frac{1}{2}\frac{\gamma+1}{\gamma-1}}\sqrt{\frac{g\gamma}{R}}\left(1+\frac{\gamma-1}{2}.M_{u_1}^{*2}\;(1-2\tau_1)\right)^{\frac{1}{2}\frac{\gamma+1}{\gamma-1}} . \operatorname{Cos}\lambda$$

The relationship is plotted in fig. 6 for a range of blade speed mach numbers and stagger angles.

As the blade speed is increased, it is seen that there is a general increase in the level of the choking mass flow, the effect being most pronounced at low or negative levels of pre-rotation with which it also increases. As the stagger angle increases however the choking mass flow is progressively suppressed. The normal rules of compressor design lead for optimum incidence to increased stagger with reduced or negative pre-rotation and, since the choking mass flow parameter $\frac{K\sqrt{T_1^*}}{P_1^*}$ is normally stipulated in a design, this fixes the lower limit to which pre-rotation may be decreased.

2.3 Time Rate of Incidence Change

It has been observed experimentally (Ref; 3) that the reaction of rotor blades to circumferential distortion is dependent upon the time rate of change of incidence as the rotor blade enters a distorted region. Higher time rates of incidence change tend to have a more powerfully de-stabilizing effect upon the rotor areodynamics (Ref; 3) and have led (Ref; 4) to a distortion criterion $\frac{c}{u} . \frac{di}{dt}$ similar to the reduced frequency parameter.

The effect of time rate of change of incidence may be examined mathematically by differentiating equation (5) with respect to t

$$\frac{d}{dt}(\Delta i) = \frac{d}{dt}\left[\tan^{-1}\left(\frac{\phi\ \Delta\phi}{(\phi - \Delta\phi)\ \{\phi^2+(1-\tau)^2\} + \Delta\phi\ (1-\tau)}\right)\right] \tag{8}$$

Now $U = \frac{dy}{dt}$ is the blade speed

and $\frac{d(\Delta i)}{dt} = \frac{di_d}{dt}$, the time dependent effect of a rotor blade through the shear flow at the entry to a circumferential distortion. We may, from equation (8) obtain

$$\frac{di_d}{dt} = \frac{Ud(\Delta\phi)}{dy} \cdot \frac{\phi^2 + (1-\tau)^2}{\Delta\phi^2 + \{\phi^2 + (1-\tau)^2 - \frac{\Delta\phi}{\phi}(\phi^2 - \tau(1-\tau))\}^2} \tag{9}$$

Where $\frac{Ud(\Delta\phi)}{dy} = -\frac{\partial Caxd}{\partial y} = \zeta$, the vorticity in the inlet flow.

The parameter $\frac{1}{\zeta}\cdot\frac{di_d}{dt}$ is plotted as a function of the distortion intensity factor $\frac{\Delta\phi}{\phi}$ and the pre-rotation level τ over a range of axial velocity ratios ϕ in fig. 7.

It is observed that at higher levels of axial velocity ratio this parameter is reduced, suggesting a better stability at high mass flow levels. In general, this term is also reduced both with reduction of distortion intensity and pre-rotation. For any distortion intensity encountered then an improved stability might be found with designs of low or negative pre-rotation. The inversion noted at low values of axial velocity ratio and positive pre-rotation is a consequence of the form that the parameter $\frac{1}{\zeta}\cdot\frac{di_d}{dt}$ takes. In physical terms it may be understood that increased vorticity ζ in a flow is likely to add to increased time rate of change of incidence $\frac{di_d}{dt}$

2.4 Application

Should a compressor installation have the potential for provoking a circumferential distortion, figs: 5,6, and 7 may be used to establish design criteria whereby the effects of the distortion on performance may be reduced.

Fig 5 may be used as a feature in a compressor design method and fig. 6 would, in conjunction, indicate the minimum level of pre-rotation permissible. With appropriate modification of equation (7) a modified form of fig. 6 would yield limiting data for any geometry of blade. From fig. 5 low values of axial velocity ratio and low or negative values of pre-rotation are found to be desirable characteristics in designing for low distortion sensitivity.

The conclusions gained from the data of fig. 7 are, to some extent, at variance with those of fig. 5. In order to reduce $\frac{1}{\zeta}\frac{di_d}{dt}$, a high axial velocity ratio is desirable whereas reduced Δi calls for a low value of axial velocity ratio. It is seen however that negative pre-rotation reduces the level of $\frac{1}{\zeta}\frac{di_d}{dt}$, re-inforcing the criterion established in section 2.1 where negative pre-rotation reduces the change of incidence Δi.

The comparative importance of the Δi_d term and the $\frac{1}{\zeta}\frac{di_d}{dt}$ term is not yet quantifiable since little is known of the effects upon cascade performance of time rate of change of incidence. Certainly, although the effects of circumferential distortion may be reduced by including this feature in a design method, they cannot be eliminated entirely.

For a known compressor geometry, assuming quasi-steady response of the rotor, fig.5 may be used to assess the effect of distortion on compressor performance. With an eventual knowledge of unsteady response characteristics of rotors, the method may be used with more confidence. Fig. 5 may also be used to assess the relative sensitivities of different compressors to distortion.

3.0 VARIABLE GEOMETRY CONSIDERATIONS

3.1 Transverse Gust Elimination

One component of the change of rotor incident velocity due to a circumferential distortion is the transverse gust analysed by Sears (Ref: 5) and this may be related to a change of lift. It is an elimination of the transverse gust, and hence the associated change of lift, that is proposed here. This involves selective re-staggering of the inlet guide-vanes in the region of the known circumferential distortion in order that the rotor inlet flow angle β remains constant (fig. 2)

Considering the velocity triangle in fig 2.

$$\tan\alpha = \frac{\phi}{\tau} \qquad \text{the design condition, holds.} \tag{10}$$

When subjected to a disturbance of magnitude $\Delta\phi$ however, maintenance of constant β is obtained by a new guide vane exit angle given by:

$$\tan\alpha_d = \frac{\phi - \Delta\phi}{\tau + \Delta\tau} \tag{11}$$

and a change of pre-rotation given by:

$$\Delta\tau = (1-\tau)\frac{\Delta\phi}{\phi} \tag{12}$$

The change of guide vane exit angle $\Delta\alpha$ is obtained from

$$\tan\Delta\alpha = \tan(\alpha - \alpha_d) = \frac{\tan\alpha - \tan\alpha_d}{1 + \tan\alpha\,\tan\alpha_d} \tag{13}$$

Substituting from equations (10), (11) and (12) into (13) the change of guide vane exit angle is resolved in terms of the mass flow coefficient, the pre-rotation and the distortion intensity factor $\frac{\Delta\phi}{\phi}$

$$\Delta\alpha = \tan^{-1}\left(\frac{\phi\left(\frac{\Delta\phi}{\phi}\right)}{\left(1-\frac{\Delta\phi}{\phi}\right)\left(\phi^2+\tau^2\right)+\frac{\tau\Delta\phi}{\phi}}\right)$$

This relationship is plotted in fig. 8.

As may be anticipated, small disturbances need only small corrective angles to the inlet guide vanes though this increases with reducing level of mass flow coefficient. The maximum correction is needed for designs in the range of pre-rotation from $0 > \tau > -0.2$ depending upon the distortion intensity encountered.

It may be noted that the inlet guide vane reset is always to increase the pre-rotation, so that the reset of individual guide vanes by this method is akin to the operation of a row of variable inlet guide vanes under part speed performance. It may also be noted that this correction calls for guide vane rotation to reduce distortion effect opposite in sense to that proposed in the design method (section 2.1).

In particularly adverse conditions of low axial velocity ratio and high distortion intensity, correction angles are prohibitively high and are not likely to be obtained in practice.

3.2 Longitudinal Gust Elimination

A second effect of the change of incident flow that results from a distortion is a change in the level of the chordwise component of incident velocity. Horlock (Ref:6) has investigated this theoretically and shown that under certain circumstances, the change of lift on an aerofoil due to a chordwise gust may be of the same order as that due to a transverse gust.

The elimination of the chordwise gust component in a distortion may be achieved by a selective restaggering of the inlet guide vanes in the region of a distortion to maintain the chordwise component of velocity constant. From fig. 3 it is seen that this involves a change of guide vane exit angle (α_d-α) to produce a change of blade inlet angle (β -β_d). The resulting change of pre-rotation becomes:

$$\Delta\tau = \frac{\phi\ \Delta\phi}{1 - \tau}$$

yielding a guide vane re-staggering of

$$\Delta\alpha_d = \tan^{-1}\left(\frac{\Delta\phi\ \{\phi^2 - \tau(1 - \tau)\}}{(1 - \tau)(\phi^2 + \tau^2) - \phi\ \Delta\phi}\right)$$

The required change of angle $\Delta\alpha$ is plotted for various axial velocity ratios ϕ in terms of the pre-rotation τ and distortion intensity factor $\frac{\Delta\phi}{\phi}$ in fig. 9.

At low values of axial velocity ratio, fig. 9 indicates a design geometry for which no correction of stagger angle is needed virtually irrespective of the distortion intensity factor $\frac{\Delta\phi}{\phi}$. This occurs when the guide vane exit flow and the rotor relative velocity are perpendicular so that the Δu effect is minimal.

3.3 Suppresion of Both Compoents of Distortion

When a rotor is subjected to a circumferential pressure distortion there are two effects upon the lift; that (ΔL_t) due to increase of incidence, the transverse gust, and that (ΔL_c) due to the reduction in the chordwise component of velocity, the chordwise gust.

Sections 3.1 and 3.2 present criteria by which the transverse gust effect ΔL_t may be eliminated through preferential re-staggering of the guide vanes in the region of a known, steady and quantifiable distortion. In each case however, the elimination of one component of incremental lift enchances the other. In this section, based on the unsteady thin aerofoil theories of Sears (Ref; 5) and Horlock (Ref; 6) a design criterionis presented by which the resultant component of the fluctuating lift may be minimised. When quasi-steady flow is assumed, a simple practical design criterionis obtained.

The disturbance in the inlet flow ahead of the guide vanes may be expressed by the Fourier series:-

$$\Delta\phi = \sum_n \Delta\phi_{on}\ \mathrm{Sin}\ (2\pi\ ny/\ell) \tag{14}$$

Where ℓ is the width of the disturbance measured in the circumferential direction y.

Assuming no change in the deviation from the inlet guide vanes, the inlet disturbance $\Delta\phi$ results in a chordwise and transverse components of disturbance given from fig. (4) by:-

$$\Delta u_c = \frac{\Delta\phi}{\sin\alpha}\ \cos(\alpha + \beta_m)$$

$$\Delta v_t = \frac{\Delta\phi}{\sin\alpha}\ \sin(\alpha + \beta_m)$$

Considering the n^{th} term of the Fourier series, the disturbance components are given by:-

$$\Delta u_{cn} = \Delta\phi_{on} \cos\beta_m (\cot\alpha - \tan\beta_m) e^{j\nu t} \tag{15}$$

$$\Delta v_{tn} = \Delta\phi_{on} \cos\beta_m (1 + \cot\alpha \cot\beta_m) e^{j\nu t} \tag{16}$$

The non-dimensional inlet relative velocity is obtained from fig (4) as:-

$$u = \frac{1}{\sin\beta_m \ (\cot\alpha + \cot\beta_m)} \tag{17}$$

Taking equations (16) and (17) into consideration, the transverse component of the lift fluctuation is given by:-

$$\Delta l_{tn} = 2\pi\rho U^2 \Delta\phi_{on} \cot\beta_m \frac{1 + \cot\alpha \tan\beta_m}{\cot\alpha + \cot\beta_m} e^{j\nu t} \ S(\omega)$$

which reduces to

$$\Delta L_{tn} = 2\pi\rho U^2 \Delta\phi_{on} \ e^{j\nu t} \ S(\omega) \tag{18}$$

where $S(\omega)$ is Sears function (Ref. 5)

$\nu = 2\pi Un/\ell$ is the frequency

and $\omega = \frac{2\pi n}{\ell} \sin\beta_m (\cot\alpha + \cot\beta_m)$ is the reduced frequency.

The chordwise component of lift fluctuation is obtained from Ref. (6) and equations (15) and (17) giving:-

$$\Delta L_{cn} = 2\pi\rho U^2 \Delta\phi_{on} \cot\beta_m \frac{\cot\alpha - \cot\beta_m}{\cot\alpha + \cot\beta_m} e^{j\nu t} \ i \ T(\omega)$$

Which reduces to:-

$$\Delta L_{cn} = 2\pi\rho U^2 \Delta\phi_{on} \ e^{j\nu t} \ i \left\{ \frac{\tau(1-\tau)}{\phi} - \phi \right\} T(\omega) \tag{19}$$

Where $T(\omega)$ is a function defined by Horlock (Ref. 6)

Since Sears and Horlock's solutions are linearised, the resultant fluctuating lift component is obtained by superimposing equations (18) and (19) yielding

$$\frac{\Delta L_n}{2\pi\rho U^2 \Delta\phi_{on} \ e^{j\nu t}} = E(\omega, i, \phi, \tau) \tag{20}$$

Where

$$E(\omega, i, \phi, \tau) = S(\omega) - i \ A \ T(\omega) \tag{21}$$

and $$A(\phi, \tau) = \phi - \frac{\tau(1-\tau)}{\phi} \tag{22}$$

The function $E(\omega, i, \phi, \tau)$ reduces to Horlock function $H(\omega, i, \phi)$ of Ref. (6) when $\tau = 0$. This function is plotted in fig. (10) for various values of iA and a range of reduced frequencies from 0 to 5.0. The function A is plotted in fig. (11) versus τ for different levels of mass flow ratio ϕ.

If quasi-steady flow is assumed, $\omega = 0$, then $S(\omega) = 1$ and $T(\omega) = 2$. The fluctuating lift can be eliminated by setting $E = 0$, yielding

$$iA = 0.5 \tag{23}$$

In this instance the incidence (angle of attack) is given by:-

$$i_{q.s} = \frac{0.5\,\phi}{\phi^2 - \tau(1-\tau)} \tag{24}$$

where q.s. stands for quasi-steady case.

For all $\phi \leqslant 0.5$, when $\phi^2 = \tau(1-\tau)$, infinite incidence is required to eliminate lift fluctuations. Since there is practical upper limit to the design incidence and the mathematical derivations are only rigorous for small values of i, high values of A are required to satisfy equation (23) in the practical range. Fig. (12) shows a plot of equation (24) for various values of ϕ. It can be seen that for all ϕ, values of negative pre-rotation τ yield practical values of i. Only for high values of $\phi>1$ that values of $i<40°$ can be obtained over the whole range of τ. It is, therefore, recommended that designs with negative pre-rotation shall be less sensitive to lift fluctuation and noise. This is again in agreement with the results of section 2 which were obtained by a different approach.

Equation (23) was derived on a quasi-steady basis. However from fig (10) it can be seen that the function E, and hence ΔL_n, is minimised when $iA = 0.3 \sim 0.5$, which gives

$$0.6 i_{qs} \leqslant i_{ns} \leqslant i_{qs} \tag{25}$$

where ns stands for the non-steady case.

Equation (25) validates the above analysis for the un-steady case as well, as far as the design trend is concerned.

For accurate quantitative estimates, further theoretical and experimental investigations are required.

3.4 Application

Fig. 13 indicates a steady circumferential distortion measured in the intake annulus of the fan for the R.F.B. Fanliner. This aircraft uses a pusher fan situated immediately downstream of the wing and fuselage intersection.

In such circumstances it becomes possible to calculate the circumferential variation in stagger of inlet guide vanes in order to reduce or eliminate the fluctuating incremental lift of the passing rotor blades. Where a fan is not normally fitted with inlet guide vanes, as with the R.F.B. Fanliner, they could be fitted locally in the region of the measured wakes (fig. 13).

The technique involves the use of the fan or compressor design parameters, pre-rotation τ, and axial velocity ratio ϕ. The solution, a prediction of the inlet guide vane reset $\Delta\alpha$, depends upon the distortion intensity factor $\frac{\Delta\phi}{\phi}$. Fig. 8 gives directly the value of $\Delta\alpha$ to eliminate the chordwise gust effect and fig. 9 gives the value to eliminate the transverse gust effect. Fig. 10, 11 and 12 may be used to obtain the guide vane setting to give a nett minimum incremental lift.

3.5 Limitations of the Method

It is assumed that the flow deviation at the inlet guide vane exit remains constant under all conditions. In some cases large changes in stagger are needed, with associated change of incidence so in these cases this assumption may not be valid.

In calculating lift variation ΔL with incidence it is assumed that the rotor response is linear. The model is therefore limited to unstalled flow.

Although the method allows for the transverse and chordwise gust effects in the flow, no account is taken of the effect of vorticity that will also be encountered. In section 2.3 an indication is given of how a design may accomodate to some extent this effect, but the relative importance of the shear flow term is not yet understood. It is known that a shear flow term $\frac{\partial(Cax)}{\partial y}$ has a marked effect upon an isolated stationary areofoil (Ref; 7).

For a rotor this term is closely related to the term $\frac{di_d}{dt}$ (equation 9) and is likely to affect the lift finally developed. This is the subject of theoretical and experimental research programmes in progress at Cranfield.

The method as proposed takes no account of unsteady flow effects and it has been observed that, particularly in the presence of large scale disturbances, these can be significant (Ref. 3).

4.0 CONCLUSIONS

Two methods are presented in which an account may be taken of a quantifiable pressure distortion in a known location. In both methods, the design parameters, pre-rotation τ, axial velocity ϕ and the off-design parameter, the distortion intensity factor $\frac{\Delta\phi}{\phi}$, are needed.

Using the first method, a reduction in distortion sensitivity can be accomplished by an appropriate choice of pre-rotation (or inlet guide vane stagger) and axial velocity ratio. These data may also be used to assess performance change of a rotor row of known geomtry and response.

The second method involves selective re-staggering of inlet guide vanes in the annulus sector affected by a distortion. The transverse gust effect, or the longitudinal gust effect, or the combined effect of both can be reduced depending upon the degree of re-staggering employed. The method does not however account for the vorticity effects present in the flow.

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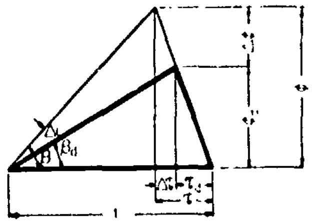


FIG. 1 PERTURBATION OF A ROTOR INLET VELOCITY TRIANGLE WITH A DISTORTED FLOW

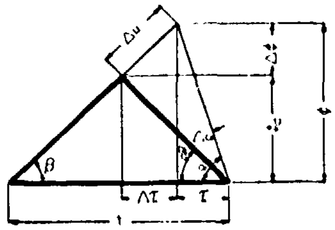


FIG. 2 EFFECT OF RESTAGGERING INLET GUIDE VANE TO ELIMINATE TRANSVERSE GUST EFFECT

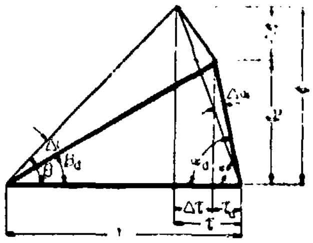


FIG. 3 EFFECT OF RESTAGGERING INLET GUIDE VANE TO ELIMINATE CHORDWISE GUST EFFECT

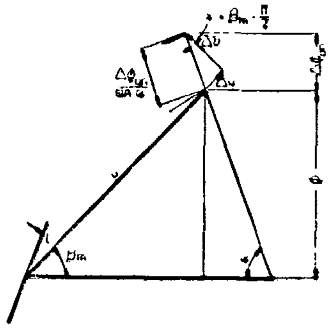


FIG. 4 ROTOR OF FLAT PLATES MOVING THROUGH DISTURBANCE IN INLET FLOW AHEAD OF INLET GUIDE VANE

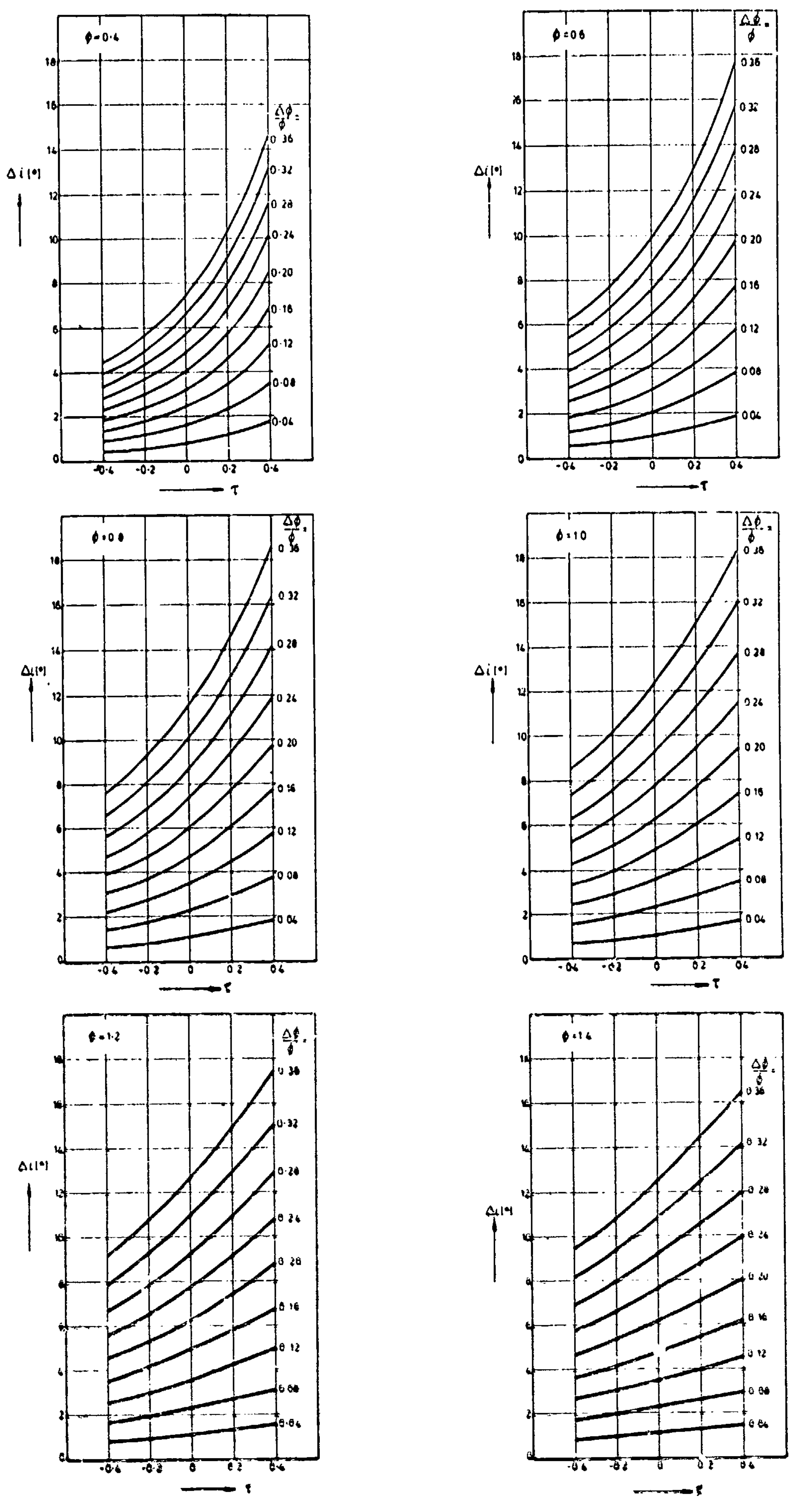


FIG. 5 INCIDENCE VARIATION FOR VARIOUS LEVELS OF DISTORTION

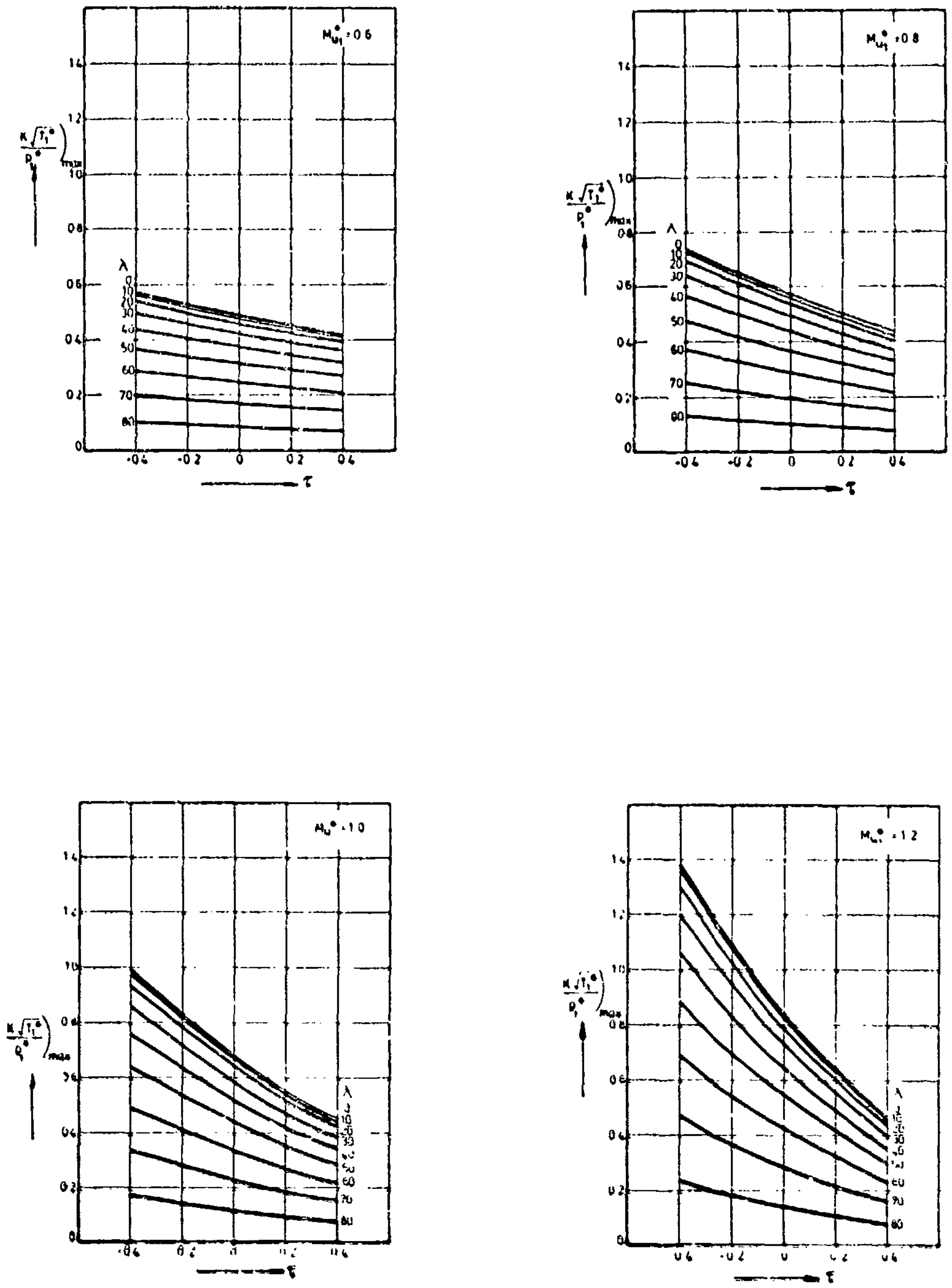


FIG. 6 CHOKING LIMITATIONS FOR VARIOUS ROTOR GEOMETRIES

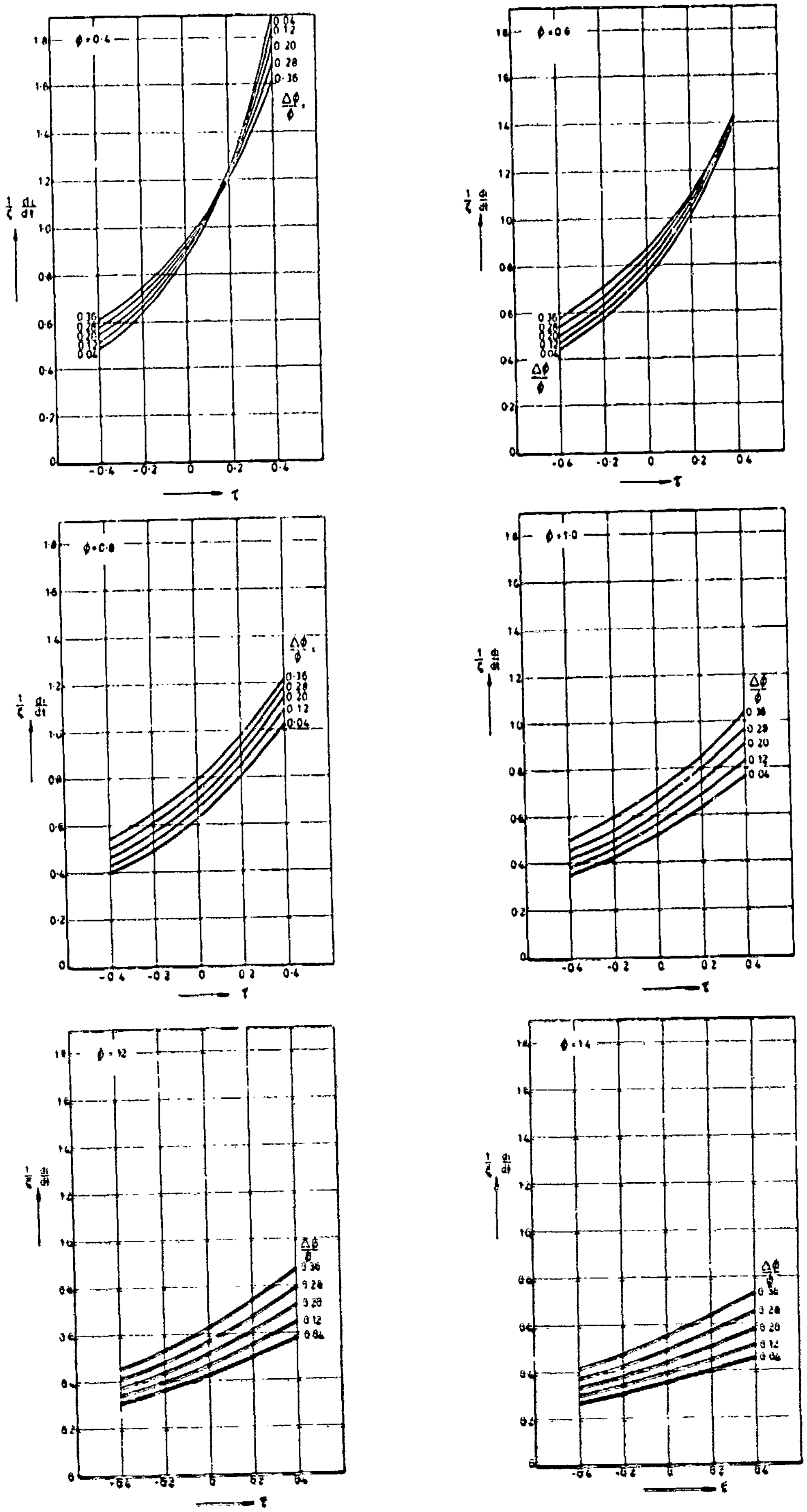


FIG. 7 VARIATION OF TIME RATE OF CHANGE OF INCIDENCE FOR VARIOUS DISTORTION LEVELS

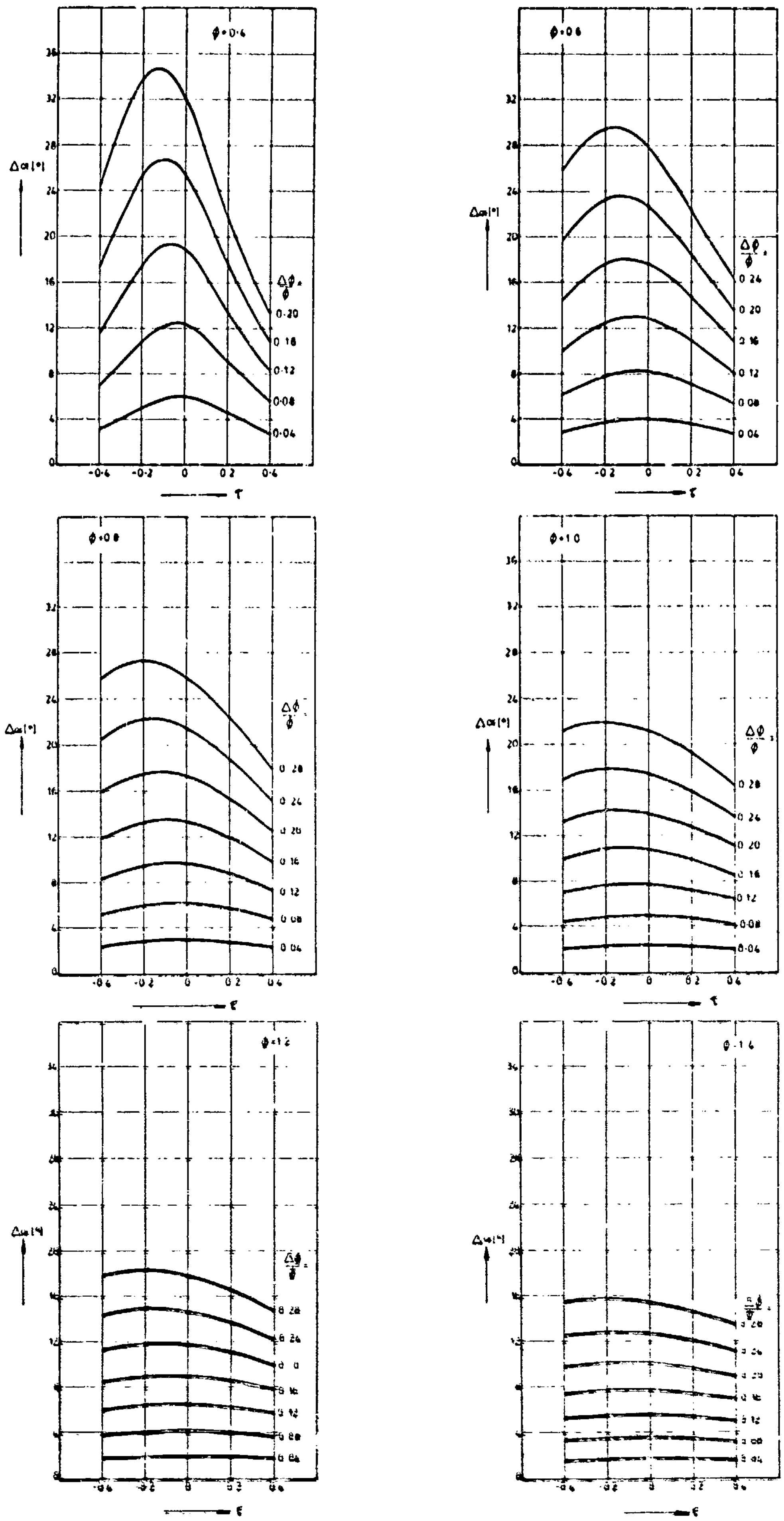


FIG. 8 INLET GUIDE VANE RESET TO ELIMINATE TRANSVERSE GUST EFFECT

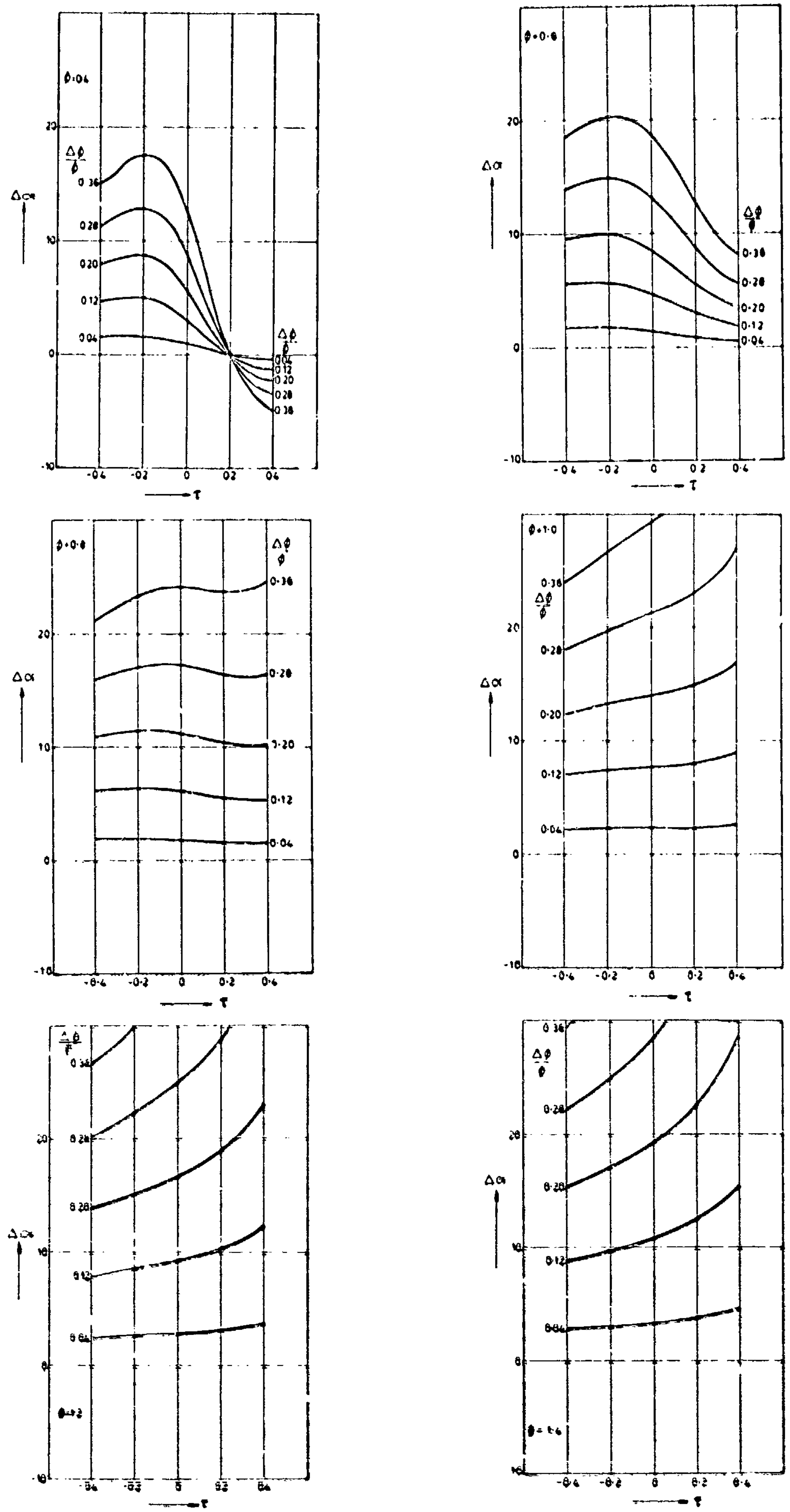
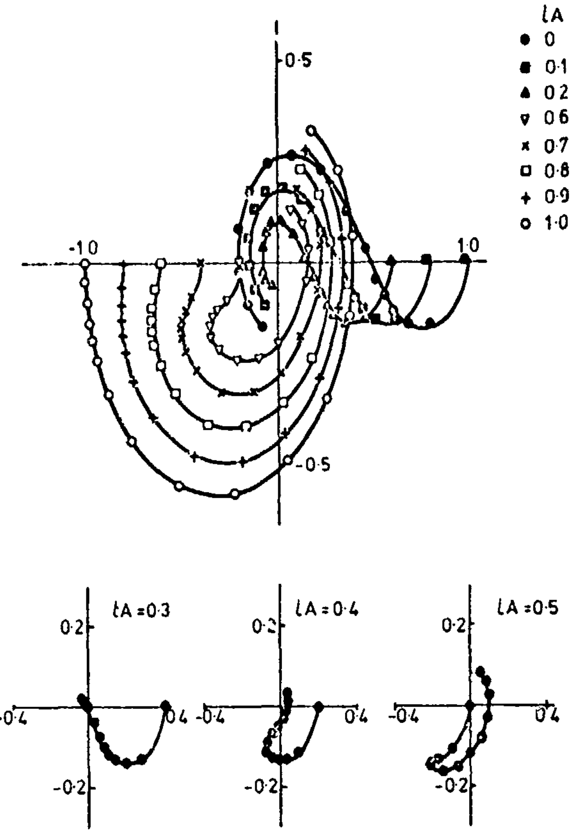


FIG. 9 INLET GUIDE VANE RESET TO ELIMINATE CHORDWISE GUST EFFECT



Values of ω shown are ω = 0, 0.1, 0.2, 0.3, 0.4, 0.5, 0.7
1.0, 1.5, 2, 2.5, 3, 3.5, 4, 4.5,5

FIG. 10 ROTOR LIFT FLUCTUATION E(ω,i,d,τ) FOR VARIOUS iA

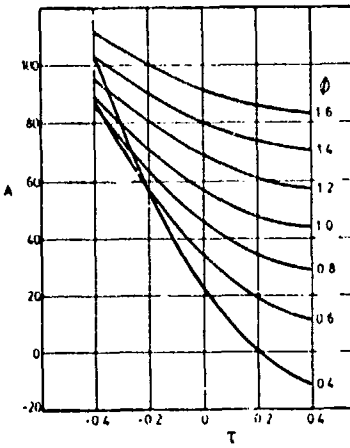


FIG. 11 THE DESIGN PARAMETER A FOR VARIOUS VALUES OF τ AND ϕ

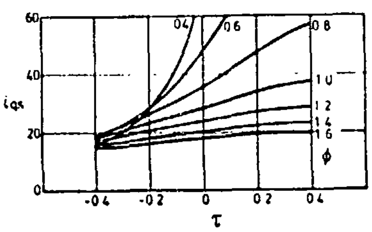


FIG. 12 QUASI-STEADY ANGLE OF ATTACK TO ELIMINATE LIFT FLUCTUATION

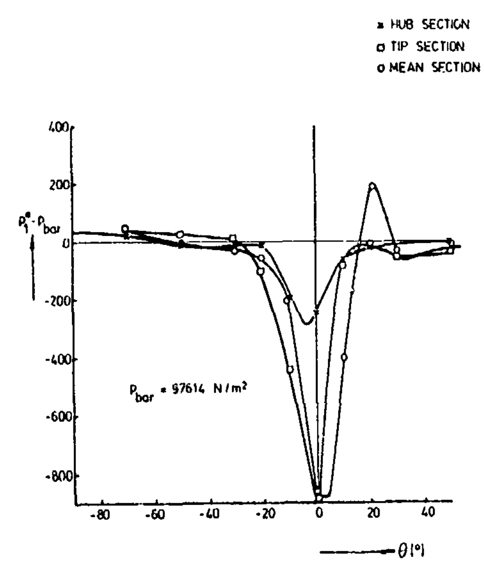


FIG. 13 MEASURED CIRCUMFERENTIAL DISTORTION IN R.F.B. FANJET INLET ANNULUS

DISCUSSION

H.Grieb

In you paper the local flow distortion is defined as the local change of the flow coefficient. There are two types of flow distortions

- distortions due to local stagnation pressure losses ahead of the compressor
- distortion due to curved duct walls ahead of the compressor.

Does the theory presented hold for both types of distortions?

Author's Reply

It will hold for both types of distortion providing they are sensed purely in the circumferential direction. If therefore one had a non-axisymmetric intake for a gas turbine engine which gave a distortion around the annulus one could design for that feature. If one had a stagnation pressure deficit which could have been introduced by an upstream strut or perhaps by a high turning manoeuvre creating distortion in the inlet, one can design for that as well. The main restriction however is that we are speaking of distortions which will vary only slowly with time, and perhaps I should have mentioned this because I may have inadvertently mislead you. When I speak of the unsteady incidence as quasi steady, I am not suggesting that the blades are actually moving in an angular manner. The blades are set at that incidence to take account of unsteady flow at entry.

J.Kurzak

Do you need a variable guide vane for all stages of the compressor if you get a multistage compressor? If we do have, say, a two-compressor engine, and you say that you keep the lift coefficient constant over the compressor you probably do not attenuate the distortion. You are carrying it through the first compressor and give it to the next one and have the problem there.

Author's Reply

The type of solution I have been speaking of is probably best applying to a single stage fan. In the case of the multistage compressor one could in fact seek for two types of solution. The one that we proposed here which will then maintain the distortion through the machine and would need to have variable stages down through the machine to accommodate this. Alternatively, one could carry out a slightly different type of analysis, where instead of minimizing on the fluctuating lift, one in fact would maximize on the fluctuating lift to attenuate the distortion by the second stage of the machine. If one goes for that type of solution one has the problem both of the fatigue life of the blade which is involved in the large perturbation and of the noise which is provoked. I am afraid that in the multistage application we are in a cleft stick and we can either go to the type of solution which we propose and which as I said is mainly applicable to the single stage or the latter solution which calls for very robust blades.

THE PREDICTION AND OPTIMISATION OF VARIABLE GEOMETRY STATORS FROM COMPRESSOR BASIC DATA

by
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SUMMARY

With high performance axial flow compressors of the type being developed today, there is a positive requirement for variable geometry. The amount of variable geometry required in terms of the number of variable stators rows, the stator travel and rate of travel are all important parameters. The effects of these parameters on compressor performance can be predicted and optimised using the "stage stacking" method of analysis.

LIST OF SYMBOLS

$\Delta\gamma$	change of blade stagger angle
R	gas constant
J	Joules equivalent
C_p	Specific heat at constant pressure
γ	Ratio of specific heats
ρ	air density
M	Mach number
T	total temperature
t	static temperature
P	total pressure
p	static pressure
Va	axial velocity
W	air mass flow
α	gas angles
K_b	annulus boundary layer blockage factor
β	blade angles
ε	blade deflection
δ	blade deviation
U	blade rotational speed
F_D	stage static pressure rise function
η	adiabatic efficiency

Suffixes

0	stage entry
1	rotor entry
2	rotor exit
3	stator entry
4	stator exit
al	alignment point
r	rotor
s	stator
st	stage

INTRODUCTION

The type of compressor being developed today for the high bypass ratio low specific fuel consumption engine calls for a high degree of technology.

The implications of achieving a low weight, low fuel consumption and high thrust on the high pressure compressor is to increase the aerodynamic loading, i.e. maximum pressure ratio with the minimum number of stages. This cannot be achieved without a penalty. If we examine Fig. 1, which is a typical overall characteristic obtained from a high pressure ratio, high loading axial flow compressor, we can see that off design performance is severely restricted.

The part speed performance of the compressor is degraded by the effect of rotating stall and thus, the restaggering of stator blades to reduce the stall region will improve the performance. Having defined a requirement for variable geometry stators, we now need a method to predict their effect on the compressor performance map. The problem arises now of what variable geometry to incorporate into the compressor since there are several parameters which have to be decided. These are:-

1) The number of variable stator rows.

2) The angular movement of each variable stator row and the relation to the movement of other variable stators.

3) The rate of blade movement, i.e. the compressor speed range over which movement of the stator blades is required.

Initially, we need a method to accurately simulate the part speed performance of the compressor. Then we need to determine the effects of variable geometry stators and to optimise the variable geometry to give the best possible starting characteristics and an adequate surge margin to cater for engine acceleration conditions.

In recent studies by Rolls-Royce (1971) Ltd., it has been found possible to satisfy these requirements by adopting a method of compressor analysis based on stage stacking, i.e. a method whereby individual stage performance is derived and then all the stages "stacked" together to give an overall performance.

Details of the work carried out at Rolls-Royce and the main points of the stage stacking program used are included in the following text. Basic equations are included in Appendix 1.

BACKGROUND

These problems became evident at Rolls-Royce several years ago. A programme had been set up to develop a high pressure ratio, high loading axial flow compressor. A rig had been built with a design target of achieving a 16:1 pressure ratio in 9 stages. To reduce costs, the rig utilised adjustable vanes for each stator row as opposed to variable vanes. With the vanes set at their nominal design angles, a characteristic as shown in Fig. 1 was achieved.

The need for variable geometry was evident, and in an attempt to determine the best variable geometry system, a 'stage stacking' computer simulation of the compressor was attempted.

A successful simulation of the compressor was obtained in as much as a reproduction of the test characteristic was achieved.

The stage stacking program was then used to determine the schedule for the variable geometry stators which solved the part speed performance problems and eliminated the rotating stall. It was found that increasing the number of variable stages improved the part speed performance. The final number of variable stages was decided by offsetting the effectiveness of additional stages against the increased manufacturing costs.

The scheduling finally selected is shown in Fig. 2 and the characteristic determined by this schedule is shown in Fig. 3. The rig compressor was later modified to include variable geometry to this standard and the test results achieved are shown compared to the prediction in Fig. 3.

Since this work was started, it was decided to use this compressor with the first stage removed as the core compressor for the RB.401 engine. A characteristic for this compressor was predicted using the stage stacking analysis and the variable geometry optimised.

Modifications to the compressor to achieve the required surge margins were recommended based on the stage stacking analysis. These modifications have recently been incorporated into the rig compressor. Rig results and engine running to date indicates that the compressor behaviour is as prediction.

Recently, stage stacking predictions have been made which show a potential use for this compressor in other engine cycles. Modifications of the variable geometry schedule can be used to alter the speed/flow relationship at high compressor speeds.

The stage stacking program utilises the compressor geometric parameters from the design data, and blade row loss coefficients determined from test data.

The use of the program is divided into two phases:-

1) Achieving a satisfactory simulation of the compressor test characteristic.

2) The use of this simulation to predict the effect of mechanical changes.

The program calculates overall compressor performance for various air mass flow and compressor speed relationships. Individual stage performance is also calculated. The modifications which can be made to the compressor are:-

1) The effect of variable geometry stators on any stage can be determined.

2) The effect of recambering or restaggering of blades can be determined.

3) The centrifugal twist on rotor blades can be simulated.

4) The effect of air bleeds on compressor performance can be simulated. Bleed off-takes at rotor or stator exits can be accommodated.

UTILISATION OF PROGRAM

The program input consists of the compressor geometry, i.e. annulus dimensions, mean diameter blade data, design air angles and efficiencies and blade row loss/incidence curves obtained from compressor test.

The calculation of the loss incidence values is detailed in Appendix 1. As can be seen from this, the calculation of these parameters from test data is dependant on the values of α_0 and α_2. Standard practice is to use design values for the reduction of the loss incidence relationship.

Thus, unless the compressor operates in the exact manner in which it was designed, these values are incorrect. Certainly in off design cases, where there will be changes in blade deviation thus causing changes in α_0 and α_2, these values are incorrect.

As a result, it has been found impossible to utilise the loss incidence curves directly since the measured loss curves are misaligned with regard to both loss and incidence. Hence, before an accurate simulation of the compressor characteristic can be obtained, the loss curves have to be aligned correctly.

Initially, a point is chosen from the test characteristic on which the loss curves are to be aligned. This point is best chosen as close to the design point as is possible, preferably a point on a constant speed line where the efficiency is at a peak. This is designated the 'alignment point'. The alignment mass flow, pressure ratio and speed are used as inputs to the program.

The point corresponding to this alignment point on each of the loss/incidence curves is noted and an alignment incidence is determined for rotors and stators as shown in Fig. 4. This alignment incidence is determined for all stages through the compressor and is used as an input to the program.

All program calculations are performed at passage mid height.

A continuity calculation into the first rotor yields an incidence at the alignment point (using the flow and speed from the alignment). This incidence should equal the incidence specified on the first rotor loss curve, but is not generally the case due to the misalignment of these curves.

The program now modifies the loss/incidence curve for the first rotor. The loss curve is moved with respect to incidence such that the loss specified by the alignment point incidence lies over the incidence calculated from the continuity equation.

The continuity calculation at the inlet to the first stator yields stator incidence. The first rotor loss curve is now moved vertically to achieve the rotor efficiency and the stator loss curve moved horizontally to align with the incidence calculated.

A continuity calculation at the second rotor inlet plane yields the second rotor incidence. Using the stage efficiency, the stator loss curve is aligned vertically. The second rotor loss curve can now be aligned horizontally.

This process is repeated until all the loss curves have been aligned. The gas angles, α_0 and α_2, used in the alignment process are those input which are usually design values. As we have already noted, these angles may not be those at which the compressor operates. Thus a further operation is necessary. The pressure ratio calculated by this process is now compared to that specified in the input data and if there is a discrepancy, the α_0 and α_2 values are modified by a fixed increment until the correct alignment pressure ratio is achieved.

PROGRAM FEATURES

The program contains the following features:-

1) Variable geometry may be specified. This is achieved by specifying a constant for each variable blade row and a graph of variable geometry.

   A constant is input for each blade row. If this constant is zero there is no variable geometry for that blade row. The alternative to zero is a number corresponding to the angular movement in degrees required for that blade row. A graph is specified of $\Delta(-\gamma)'/\Delta(-\gamma)$ against compressor speed where

   $\Delta(-\gamma)$ is the constant input for each blade row

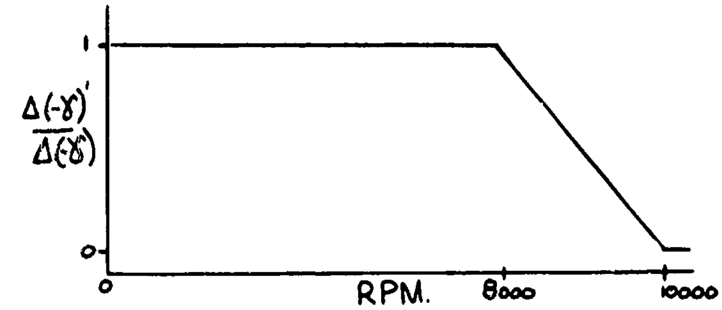
   $\Delta(-\gamma)$ is the required restagger.

   The value of $\Delta(-\gamma)'/\Delta(-\gamma)$ interpolated at the compressor speed specified is multiplied by the constant to achieve the required restagger.

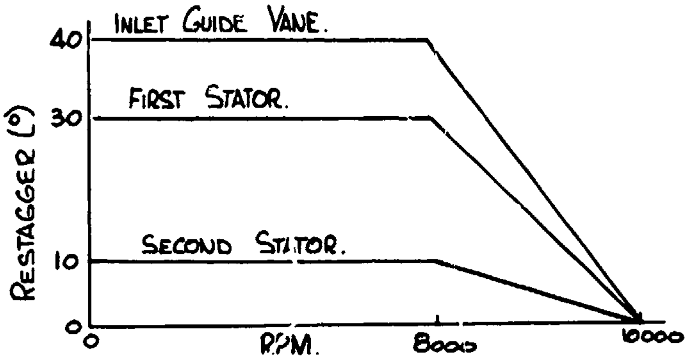
   For example, if the following constants are input to the program:-

$\Delta(-\gamma)$	for inlet guide vane	=	40
$\Delta(-\gamma)$	for the first stator	=	30
$\Delta(-\gamma)$	for the second stator	=	10

and the following graph used,



the implied schedule will be:-



2) Bleeds taken from the compressor are allowed for by including a flow factor in the input data for each stage, where

$$F_F = \frac{\text{Mass flow entering rotor}}{\text{Mass flow entering compressor}}$$

for F_F positive. If F_F negative then

$$-F_F = \frac{\text{Mass flow entering stator}}{\text{Mass flow entering compressor}}$$

3) Splitting of the loss coefficient curves can be simulated by means of a graph of ω_{cor}/ω versus blade inlet Mach number for each loss curve. Input to the program for each blade row is a Mach number graph indicator, this being 1 or 0. 1 indicated that a Mach number/loss correction graph is required for that blade row.

ω_{cor} is the corrected loss coefficient.

ω is the loss coefficient interpolated from the aligned loss curves.

Thus to achieve a split characteristic, the loss coefficient interpolated from the aligned loss curve is multiplied by ω_{cor}/ω interpolated from the specified curve using inlet Mach number.

4) At points other than the alignment point the blade outlet air angles are modified by using a graph which modifies the blade deviation. This graph is of the form

$$\frac{\delta - \delta_{al}}{\varepsilon_{al}} \quad \text{versus} \quad \frac{i - i_{al}}{\varepsilon_{al}}$$

This correlation is derived from data presented by Howell, ref. 1. Modification of this curve is possible. It has been found from analysis on several compressors that different curves for off design deviation are required to achieve an accurate alignment. Typical curves are shown in fig. 5.

5) For all points calculated in the program, a function F_D is determined for each stage, where F_D is a stage pressure rise coefficient parameter. See Appendix 1.

For a given compressor stage at a given compressor rpm the peak value of the pressure rise coefficient is designated the point at which the stage stalls. By overlaying stage stall lines on the compressor characteristic, a prediction of the surge line can be made. An example of this is shown in Fig. 6.

6) A plotting routine is available to plot the aligned loss coefficients, pressure rise coefficients and the overall characteristics.

REFERENCE

1. A.R. Howell   The Present Basis of Axial Compressor Design
Parts I and II 1942 ARC RM 2095.

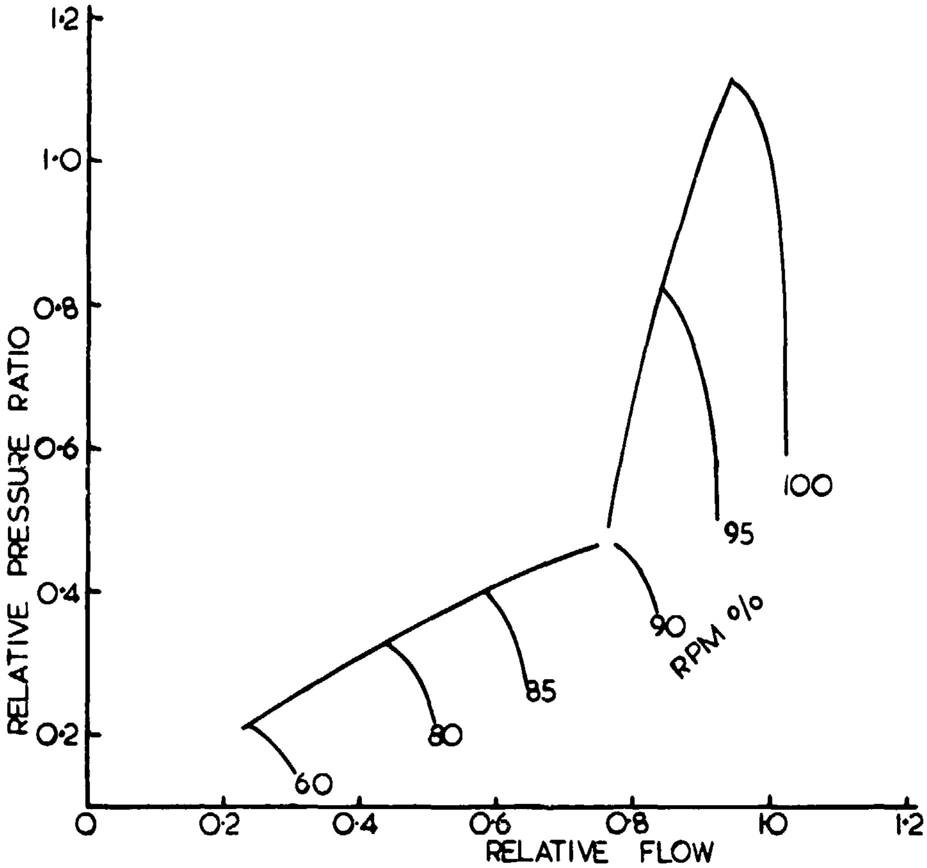


FIG 1. FIXED GEOMETRY COMRESSOR CHARACTERISTIC

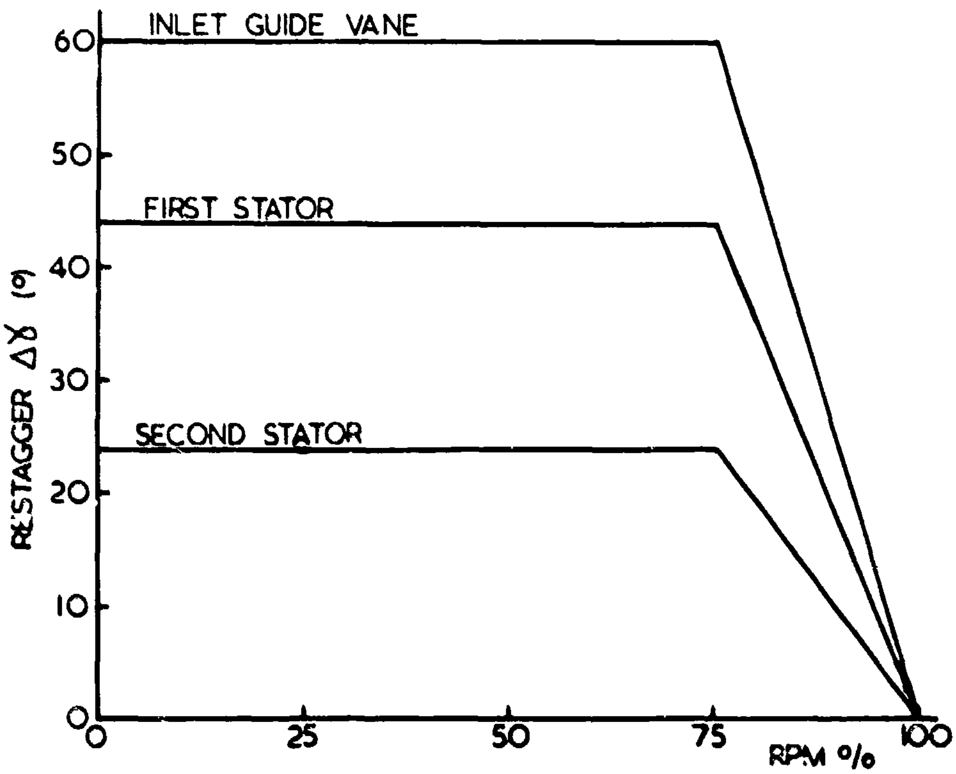


FIG 2 VARIABLE STATOR SCHEDULE

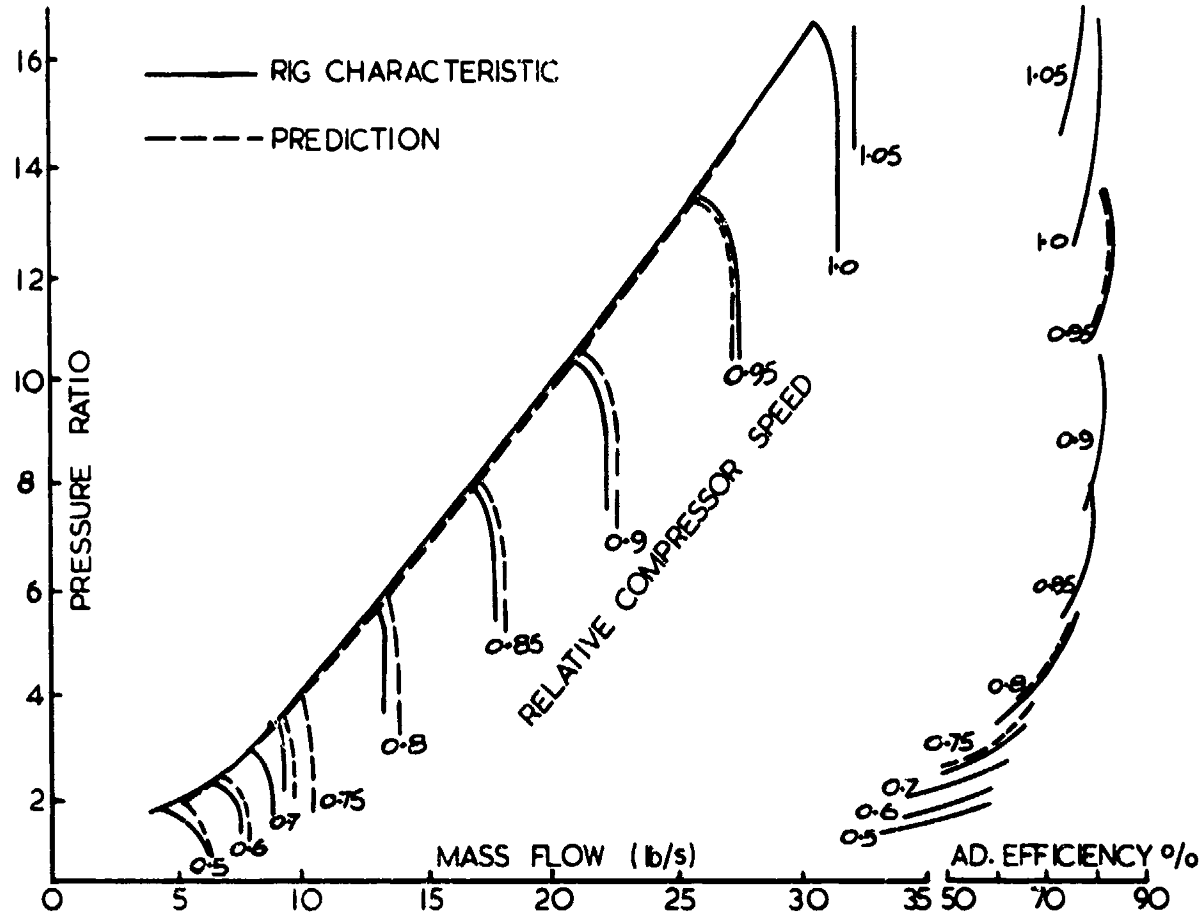


FIG 3 COMPARISON OF RIG RESULTS AND PREDICTION

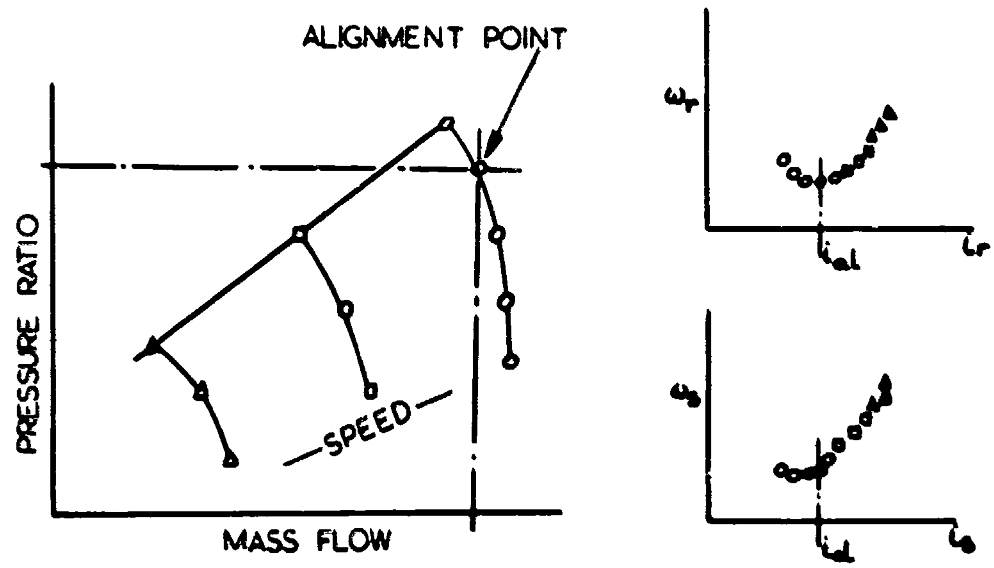


FIG 4 DERIVATION OF ALIGNMENT POINT INCIDENCES

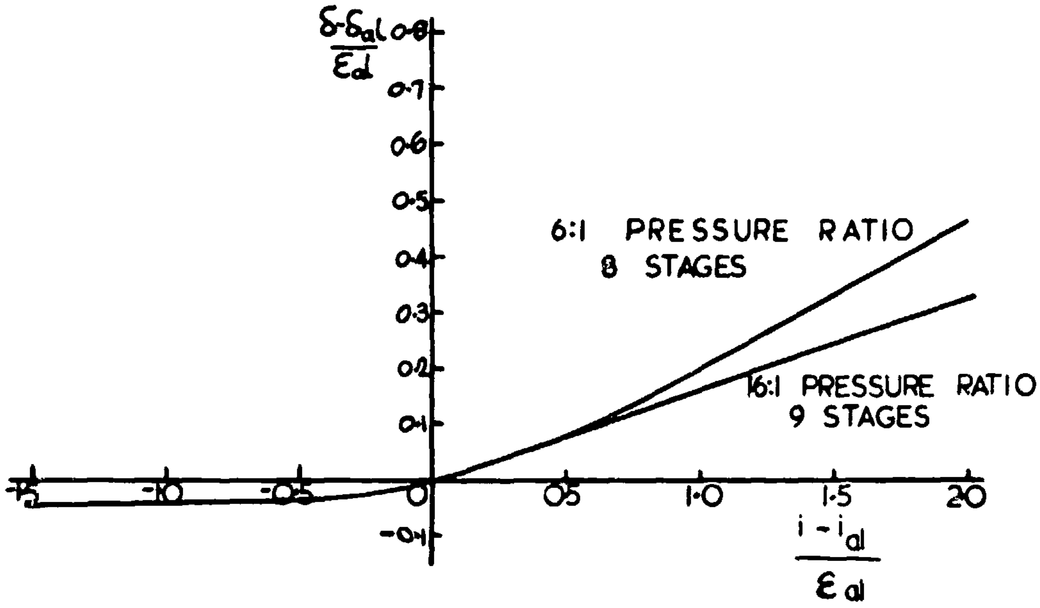


FIG.5. OFF DESIGN DEVIATION CURVE

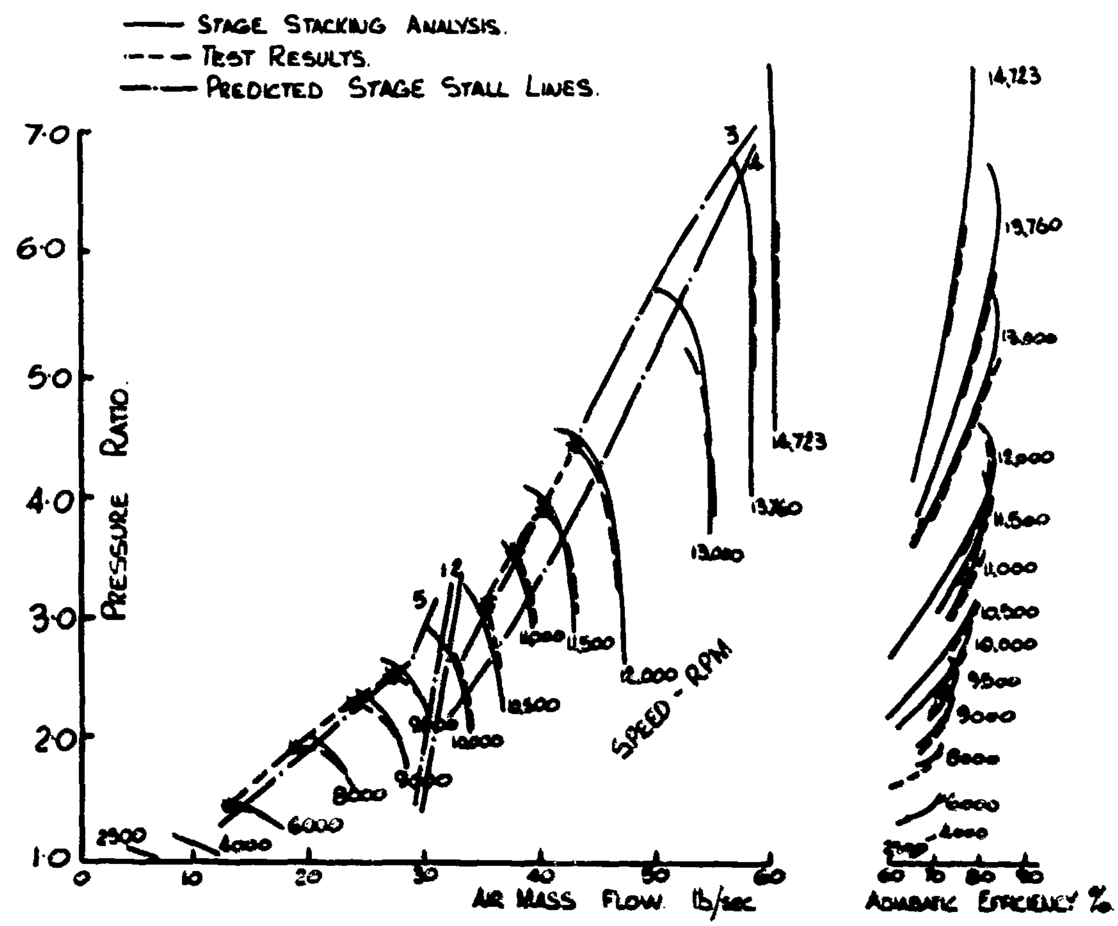


FIG.6. COMPRESSOR CHARACTERISTIC WITH STAGE STALL LINES

BASIC EQUATIONS:-

$$\gamma = \frac{1}{1 - R/JC_p}$$

Guess M_0

$$t_0 = T_0 / (1 + \tfrac{\gamma - 1}{2} M_0^2)$$

$$p_0 = (t_0/T_0)^{\gamma/\gamma-1} \times P_0$$

$$V_a = W.R.t_0 / p_0.A_0.K_{b0}$$

$$M_0 = \frac{V_a}{\cos\alpha_0} \times \frac{1}{\sqrt{g.\gamma.R.t_0}}$$

iterate on M_0

Guess V_{a3}

$$\alpha_3 = \tan^{-1}(U_3/V_{a3} - \tan\alpha_2)$$

$$\Delta T = \frac{(U_3 V_{a3} \tan\alpha_3 - U_0 V_{a0} \tan\alpha_0)}{g.J.C_p}$$

$$P_3 = P_0 (1 + \eta_{ral}. \frac{\Delta T}{T_0})^{\gamma/\gamma-1}$$

$$T_3 = T_0 + \Delta T$$

Guess M_3

$$t_3 = T_3 / (1 + \tfrac{\gamma-1}{2} M_3^2)$$

$$p_3 = P_3 (t_3/T_3)^{\gamma/\gamma-1}$$

$$V_{a3} = W.R.t_3 / p_3.A_3.K_{b3}$$

$$M_3 = \frac{V_{a3}}{\cos\alpha_3} \times \frac{1}{\sqrt{g.R.C_p}}$$

iterate on M_3; iterate on V_{a3}.

$$P_4 = P_0 (1 + \eta_{ad} \frac{\Delta T}{T_0})^{\gamma/\gamma-1}$$

Repeat for each stage

Pressure Ratio = $\frac{P_4 \text{ (last stage)}}{P_0 \text{ (first stage)}}$

Check: Pressure Ratio compared to Alignment Pressure Ratio
α_0 and α_2 modified to give Al. Pressure Ratio.

The following calculations are then performed for each stage:-

$$\alpha_1 = \tan^{-1}(U_0/V_{a0} - \tan\alpha_0)$$

$$M_1 = M_0 \frac{\cos\alpha_0}{\cos\alpha_1}$$

$$P_1 = p_0 (1 + \frac{\gamma-1}{2} M_1^2)^{\gamma/\gamma-1}$$

$$M_2 = M_3 \frac{\cos\alpha_3}{\cos\alpha_2}$$

$$P_2 = p_3\left(1 + \frac{\gamma-1}{2} \cdot M_2^2\right)^{\gamma/\gamma-1}$$

$$\omega_r = \frac{P_1 - P_2}{P_1 - p_0}$$

$$i_r = \alpha_1 - \beta_1$$

Thus rotor loss curve aligned

$$\omega_s = \frac{P_3 - P_4}{P_3 - p_3}$$

$$i_s = \alpha_3 - \beta_3$$

Allowing stator loss curve to be aligned

$$\left.\begin{aligned} \varepsilon &= \alpha_1 - \alpha_2 \\ \delta &= \alpha_2 - \beta_2 \end{aligned}\right\}$$ used to determine off design deviation correction

$$F_D = \frac{\Delta p_{ST.}}{\tfrac{1}{2}\rho U^2\left(1 + (\gamma-1) M_1^2\right)^{1/\gamma-1}}$$

where $\rho = \dfrac{p_0}{R g t_0}$

and $U = \dfrac{U_1 + U_2}{2}$

After alignment, for other points, the calculations are repeated with ω_r and ω_s derived from loss curves input.

DISCUSSION

H.Ahrendt

Referring to figure 3 your plotted predicted surge line is identical with the rig characteristic. Did you actually measure the compressor surge line? For me, there is a surprising good correspondence between predicted and measured data.

Author's Reply

Yes, we did. At each speed we measured the flow and the pressure ratio and took that out to the surge point. We measured that on test.

H.Grieb

In the performance map on figure 3 the line of optimum efficiencies lies rather close to the surge line. My questions are:

- Would you fear engine stability problems in the case of a turbofan due to the moderate level of surge margin, if you set the working line close to optimum efficiencies?
- Did you make an attempt to increase the distance between surge line and optimum efficiency line, for example, by the introduction of a further row of variable stators?

Author's Reply

I don't really think it is necessary for engine stability to have a large separation between surge line and optimum efficiences. But having said that we did in fact do some work on the compressor and attempted to move the surge line further away from the peak efficiency. In doing this work we utilised this programme to try to determine what blades changes were needed in the compressor to achieve this end. Later test results have shown that this was successful.

H.Grieb

Could you imagine that you would have a good chance for a progress in achieving more distance between optimum efficiencies and surge line when applying more variable stators?

Author's Reply

No, I don't think you would have needed more variable stators. If you look at the way we put the variable stators in (see Fig.2) you will see that at the 100% point we have no variable geometry. They are all at their design conditions. The variable geometry we have put is not going to affect the distance of peak efficiencies and surge line.

M.Giraud

Dans votre exposé il est fait mention d'un compresseur à trois grilles d'aubes variables. La première grille (I.G.V.) a un calage de 60° (voir Fig.2). Ne pensez-vous pas que cette valeur est excessive et peut entrainer des décollements sur l'aubage I G V qui peuvent annuler l'effet de pré-rotation de l'écoulement. Quelle est, à votre avis, la limite pratique de cet angle de giration?

Author's Reply

Well, this is true. By adapting very high changes of swirl, that is very high movements of the inlet guide vane, we did in fact create problems for ourselves because of incidence range and such like. We, in fact, designed this blade row to have its optimum incidence at a point below the 100% design conditions for the compressor. By this, we preserved reasonable working down to a fairly low rpm and did not significantly compromise the blade load at the 100%.

J.Kurzak

Since your programme is able to predict the surge line very exactly I have the question. Is it possible to predict surge for distorted flow condition and also the stalling blade row when using this programme?

Author's Reply

I don't really know. We have never attempted to look at distorted flow conditions. We have only used steady state not deformed distributions at the inlet. It may be somewhat difficult to deal with that as it is a very simplified programme and so far used to a mid-radius calculation only. Our basic data we use as the input to the programme are mid radius values. Therefore we might only use a circumferential distortion or even a radial distortion but it will probably be very difficult to do so. Although I admit we have looked in fact to tip twist due to centrifugal effects. But I think inlet distortion would be out of the scope of what is basically a very simple programme.

PRÉDICTION DE PERFORMANCES D'UN COMPRESSEUR A GÉOMÉTRIE VARIABLE PAR UN CALCUL DE SIMULATION

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RESUME

L'effet de la géométrie variable est étudié numériquement. Une approximation dans l'écriture des dérivées dans le sens axial permet de calculer l'écoulement moyen (uniforme t ngentiellement) en régime subsonique, transsonique et supersonique réversible, en tenant compte des pertes dues aux frottements, dissipation des sillages, non adaptation des bords d'attaque, variations de sections. Les prévisions du calcul sont comparées aux résultats expérimentaux obtenus sur un compresseur à aubes à calage réglable à l'arrêt en régime subsonique.

◇

PREDICTION OF VARIABLE GEOMETRY COMPRESSOR PERFORMANCES (OFF DESIGN)

SUMMARY

Effect of variable geometry is investigated with a numerical method. This approach, based on a simplified way of writing derivatives with respect to the axial direction, allows to calculate the trough flow (uniform tangentially) in subsonic, transsonic and supersonic regimes, provided no shock waves occur in the calculation planes. The overall flow analysis takes into account losses caused by fluid friction, wake mixing, angle of attack, cross section variation. Results of computation are compared to experimental data obtained on a low speed compressor with variable blade angle setting.

◇

NOTATIONS

C_{f_p} Coefficient de pertes secondaires, fonction du rayon.

C_p Chaleur massique à pression constante.

k Pente de la ligne de courant méridienne.

k_o Pente du carter interne.

k_1 Pente du carter externe.

m Pente de la tangente au squelette d'une aube dans le plan tangent à une surface de courant.

$\dot{m}$ Débit masse.

p Pression statique.

P Pression d'arrêt.

P_o Pour l'exploitation des résultats expérimentaux, pression d'arrêt à l'entrée de la roue mobile.

r Rayon.

r_o Rayon du moyeu.

r_1 Rayon du carter externe.

$\mathcal{R}$ Constante massique du gaz.

t Fraction de circonférence offerte au fluide.

T_i Température d'arrêt en axes absolus.

u Composante radiale de la vitesse.

v Composante azimutale de la vitesse en axes absolus.

w Composante axiale de la vitesse.

$\bar{w}$ Vitesse axiale moyenne à la traversée d'une grille sur une surface de courant.

γ Constante isentropique.

β_o Angle d'attaque nominal.

β_e Angle réel de sortie.

δP Perte de charge, $\delta P = \rho \omega r v - P + P_0$

$\delta \Psi_i$ Coefficient de perte de charge, $\delta \Psi_i = 2\delta P / \rho \omega^2 r^2$

Δs Accroissement d'entropie.

φ Coefficient de débit, $\varphi = \bar{w} / \omega r$

Ψ_i Coefficient d'accroissement de pression en fluide incompressible, $\Psi_i = 2(P - P_0)/\rho \omega^2 r^2$

Ψ_s Coefficient de pression statique, $\Psi_s = 2(p - P_0)/\rho \omega^2 r^2$

ω Vitesse angulaire, pour une grille fixe $\omega = 0$

INDICES

n Rang d'une roue.

$4n-1$ Plan d'entrée à l'extérieur de la roue.

$4n$ Plan d'entrée à l'intérieur de la roue.

$4n+1$ Plan de section de passage minimale.

$4n+2$ Plan de sortie.

1 - INTRODUCTION

Les bureaux d'études des constructeurs de moteurs d'aviation disposent actuellement de nombreuses méthodes pour le calcul de l'écoulement à travers les compresseurs et les turbines. Mais comme l'a montré récemment la journée de confrontation des méthodes d'évaluation de performances organisée par l'AGARD [1], ces calculs ne permettent pas de prévoir avec une précision suffisante le fonctionnement hors adaptation de ces machines et notamment leur limite de décrochage. La principale difficulté provient d'une connaissance insuffisante des pertes, mais aussi de ce que les divers programmes de calcul sont mal adaptés par leur structure même à l'amélioration des schémas prenant en compte les irréversibilités.

C'est pourquoi un programme de calcul simplifié a été établi à l'ONERA en vue surtout de permettre la prévision de l'effet des modifications géométriques apportées à un compresseur ou à une turbine lorsque les performances au point d'adaptation sont connues. Un tel programme, désigné sous le nom de "simulateur numérique des turbomachines" se trouve spécialement bien adapté à l'étude des performances des compresseurs à calage variable, c'est pourquoi sa présentation a été incluse dans ce cycle de conférences sur les "Turbomachines à géométrie variable et à cycles multiples".

Limité à l'approximation d'écoulement moyen comme la méthode matricielle [2] et la méthode des courbures [3], le simulateur numérique présente l'avantage sur le schéma plus ancien de l'actuateur [4] de considérer successivement les plans d'entrée, de section de passage minimale et de sortie de chaque grille, fixe ou mobile, d'un compresseur ou d'une turbine. L'hypothèse fondamentale du programme est que les perturbations induites par chaque grille sont convectées par le fluide mais qu'elles n'influencent pas les sections se trouvant en amont de celle où elles prennent naissance. Dans ce cadre, les irréversibilités qui peuvent être dues soit à l'attaque non adaptée des aubages, à la diffusion de l'écoulement dans les canaux interaubes divergents, au frottement sur les parois fixes ou mobiles, au mélange des sillages dans les espaces inter-roues ou aux phénomènes secondaires près des parois, pour ne citer que les principales causes d'irréversibilités, sont assimilées à des générations d'entropie entre plans discrets. Il est alors aisé, par comparaison des relevés expérimentaux aux résultats du calcul numérique, de corriger celui-ci et notamment si l'on dispose de relevés locaux (répartition radiale de pression d'arrêt, de température d'arrêt ou de toute autre grandeur) de corriger successivement les paramètres utilisés pour la description de l'écoulement dans chaque roue et obtenir ainsi un ensemble de données numériques qui sont parfois plus cohérentes même que les résultats expérimentaux proprement dits.

2 - FORMULATION GENERALE DES EQUATIONS DU SIMULATEUR NUMERIQUE

2.1 - Hypothèses fondamentales

Le simulateur numérique utilise l'hypothèse de l'écoulement axisymétrique moyen [5] avec la restriction supplémentaire d'aubes à ligne moyenne radiale, ce qui élimine, comme on le verra plus loin, l'introduction de forces volumiques dans les équations [6].

Nous nous limitons ici au cas d'un compresseur axial comprenant n étages composés chacun d'une roue mobile et d'un redresseur fixe. Pour chacune de ces roues nous nous bornerons à l'étude du plan d'entrée, du plan de section de passage minimale et du plan de sortie. Entre chacun de ces plans, une génération d'entropie représentera les diverses irréversibilités.

D'une façon générale nous supposons que le fluide remplit complètement la veine qui lui est offerte, le décrochage du compresseur correspondant justement à l'impossibilité d'obtention d'une telle solution pour une valeur donnée du débit. Dans le plan de la section de passage minimale nous tiendrons compte de l'épaisseur des aubes au moyen d'un facteur d'obstruction géométrique.

Enfin la compatibilité radiale des divers filets de courant est assurée par la condition d'équilibre radial, dont l'écriture constitue la seule hypothèse simplificatrice de ce calcul.

2.2 - Equation de continuité. Définition des rayons homologues

Si nous désignons par t la fraction de circonférence offerte au fluide à chaque rayon

- $t = 1$ dans les plans $4n-1$, $4n$ et $4n+2$

- $t = 1 - qe/2\pi r$ dans les plans $4n+1$, où e est l'épaisseur des aubes et q leur nombre.

L'équation de continuité qui, dans l'hypothèse d'un écoulement axisymétrique s'écrit

$$\frac{\partial(\rho t u r)}{\partial r} + \frac{\partial(\rho t w r)}{\partial z} = 0 \qquad (1)$$

où u est la composante radiale de la vitesse et w la composante axiale, permet de définir la fonction de courant ψ telle que

$$w = \frac{1}{\rho t r} \frac{\partial \psi}{\partial r} \qquad (2)$$

Nous désignerons par rayons homologues dans deux plans d'indices k et m différents, deux rayons r_k et r_m correspondant à la même valeur de Ψ définie par :

$$(3) \qquad \Psi = \int_{r_{0,k}}^{r_k} \rho w t r \, dr = \int_{r_{0,m}}^{r_m} \rho w t r \, dr$$

2.3 - Equation d'équilibre radial

Les aubes étant supposées radiales, la projection sur un rayon de l'équation du mouvement s'écrit dans l'hypothèse d'un écoulement axisymétrique :

$$(4) \qquad \rho u \frac{\partial u}{\partial r} + \rho w \frac{\partial u}{\partial z} - \rho \frac{v^2}{r} + \frac{\partial p}{\partial r} = 0$$

où v est la composante azimutale de la vitesse en axes absolus, ρ la masse volumique et p la pression.

L'hypothèse simplificatrice que nous ferons est d'admettre que la trace méridienne des surfaces de courant axisymétriques s'interpolent à partir de celles des carters externe et interne ce qui permet d'expliciter les deux premières dérivées partielles de l'équation (4) et donne le gradient radial de pression:

$$(5) \qquad \frac{\partial p}{\partial r} = \rho \left(\frac{v^2}{r} - w^2 \frac{dh}{dz} + 2 w^2 h \frac{r_1 h_1 - r_0 h_0}{r_1^2 - r_0^2} \right)$$

où h est la pente des méridiennes, les indices 0 et 1 se rapportant respectivement au carter interne et externe.

2.4 - Composante azimutale de la vitesse

Dans le plan d'entrée de chaque grille (indice $4n-1$ pour la grille d'ordre n) la composante azimutale v_{4n-1} de la vitesse au rayon r_{4n-1} est liée par la condition de conservation de la circulation à la composante azimutale v_{4n-2} à la sortie de la grille précédente au rayon homologue r_{4n-2} du rayon r_{4n-1} considéré.

$$(6) \qquad r_{4n-1} \; v_{4n-1} = r_{4n-2} \; v_{4n-2}$$

Notamment dans le plan d'entrée de la première grille la vitesse azimutale v_3 est nulle.

Dans les autres plans, la composante azimutale de la vitesse v est liée à la composante axiale w par la relation :

$$(7) \qquad v = \omega_n r + m w$$

où $\omega_n = 0$ pour les grilles fixes et

$\omega_n = 2\pi N/60$ pour les grilles mobiles ; N est la vitesse de rotation du compresseur en tours par minute et m la pente de la vitesse absolue dans le plan tangent à la surface de courant, mesurée par rapport à la direction axiale.

Nous admettrons que dans les plans d'indice $4n$ et $4n+1$, situés à l'intérieur de la grille (fig. 1) cette pente m est définie par la géométrie de la grille, mais que dans le plan de sortie $4n+2$

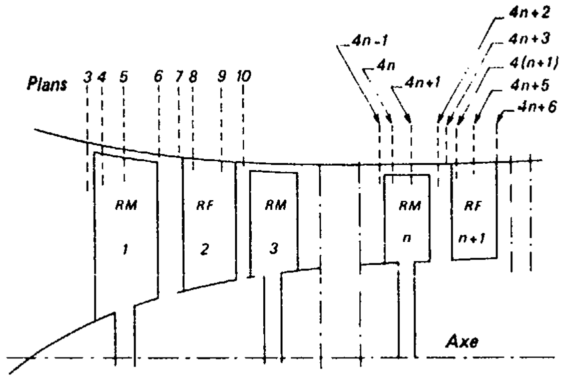

Fig. 1 – Schéma d'un compresseur multiétage, définition des plans de calcul de la roue d'ordre n.
4 n – 1 plan situé immédiatement en amont des bords d'attaque
4 n plan situé immédiatement en aval des bords d'attaque
4 n + 1 plan de section de passage minimale
4 n + 2 plan des bords de fuite.

il existe un écart flux-profil fonction du pas relatif des aubes, du calage de la grille et de l'angle d'incidence. L'estimation de cet écart flux-profil est réalisée suivant les règles classiques exposées dans le rapport NASA SP36 [7].

2.5 - Température d'arrêt du fluide

Tous les calculs s'effectuent en axes absolus et la température d'arrêt doit être calculée dans ce même système de référence, même lorsque physiquement ce calcul n'a pas de signification.

Dans tous les cas elle se calcule par référence au plan d'entrée :

$$(8) \qquad c_p T_{i_{4n+k}} = c_p T_{i_{4n-1}} - \omega_n r_{4n-1} v_{4n-1} + \omega_n r_{4n+k} \left(\omega_n r_{4n+k} + m_{4n+k} w_{4n+k} \right) \qquad k = 0, 1 \text{ et } 2$$

les rayons r_{4n+k} et r_{4n-1} étant homologues. La température statique s'en déduit immédiatement.

2.6 - Pression d'arrêt et pression statique

En l'absence de pertes la pression d'arrêt en un point pourrait être directement déduite de la température d'arrêt par la relation de transformation isentropique :

$$(9) \qquad \left(P_{4n+k}\right)_{is} = P_0 \left(T_{i_{4n+k}} / T_{i_0} \right)^{\gamma/\gamma-1}$$

En fait, cette valeur $(P_{4n+k})_{is}$ doit être corrigée pour tenir compte des irréversibilités. Malheureusement il n'existe pas encore de corrélations précises entre les grandeurs géométriques de l'aubage, les grandeurs aérothermodynamiques de l'écoulement et les facteurs de corrections donnant les valeurs réelles de la pression d'arrêt. C'est pourquoi nous utilisons dans le simulateur numérique une relation empirique pour déterminer l'accroissement d'entropie au cours de la transformation, $\sum_0^{4n+k} \Delta s$, lequel est relié aux grandeurs cinématiques de l'écoulement par :

$$(10) \qquad P_{4n+k} = P_0 \left(\frac{T_{i_{4n+k}}}{T_{i_0}} \right)^{\gamma/\gamma-1} e^{-\sum_0^{4n+k} \frac{\Delta s}{R}}$$

où $\mathcal{R}$ est la constante massique du gaz.

Les corrélations empiriques de pertes qui nous ont donné les confrontations les meilleures entre théorie et expérience consistent à décomposer l'accroissement d'entropie dans chaque grille en :

i) accroissement d'entropie dû à la non adaptation entre plans $4n-1$ et $4n$

ii) accroissement d'entropie par diffusion, entre plans $4n-1$ et $4n+2$

iii) accroissement d'entropie par mélange de sillage entre plans $4n+2$ et $4n+3$

iiii) accroissement d'entropie dû aux pertes secondaires.

Connaissant enfin la pression d'arrêt, les températures statique et d'arrêt, on obtient sans difficultés la pression locale et la masse volumique du gaz, nécessaire pour le calcul du débit masse.

3 - UTILISATION PRATIQUE DU SIMULATEUR

La mise en oeuvre du simulateur numérique nécessite la connaissance :

- de la géométrie du compresseur (lois de variation des carters externe et interne, équations des squelettes des aubes fixes et mobiles, repartitions des épaisseurs des aubes) ;

- de la vitesse de rotation ;

- des conditions de température et de pression à l'infini amont.

Comme nous l'avons indiqué, la déflexion donnée par chaque grille s'obtient à partir de sa définition géométrique par la prise en compte de l'écart flux-profil, mais l'accroissement de pression d'arrêt correspondant à l'accroissement de température que l'on peut ensuite calculer doit encore être corrigé pour tenir compte des irréversibilités.

Pour un débit masse fixé $\dot{m}$, le calcul s'effectue de l'amont vers l'aval, la conservation globale du débit devant être assurée dans chaque section.

$$\dot{m} = 2\pi \int_{r_{0_{4n+k}}}^{r_{1_{4n+k}}} \rho w t r \, dr \tag{11}$$

Le domaine de variation de ce débit pour une vitesse de rotation donnée est limité par deux des conditions suivantes :

i) aux débits élevés par un accroissement insuffisant de la pression ne permettant plus l'évacuation du flux aspiré par le compresseur dans l'enceinte avale ;

ii) aux vitesses de rotation élevées, la section de passage minimale (section $4n+1$) peut être trop petite pour permettre le passage du débit imposé (généralisation au cas d'un écoulement stratifié du blocage sonique des tuyères), le débit du compresseur est alors indépendant de la contrepression pour un certain domaine de variation de celle-ci, limité aux faibles valeurs de la contrepression par l'impossibilité d'évacuer le débit (cf. (*i*) ci-dessus) et aux plus fortes valeurs de celle-ci par une réduction du débit et un retour au fonctionnement normal à débit fonction de la contrepression ;

iii) aux faibles débits, la compatibilité radiale des divers filets de courant gouvernée par l'équation (5) peut cesser d'être possible et la veine offerte au fluide ne peut plus être complètement remplie avec le débit masse donné ; le débit maximal pour lequel apparaît une telle impossibilité peut être considéré comme le débit de décrochage du compresseur, les configurations de l'écoulement obtenues pour des débits plus faibles comportant alors des zones de fluide mort stationnaires (décollement pariétal) ou instationnaires (décollement tournant) suivant la géométrie du compresseur, comme nous le verrons plus loin sur un exemple expérimental ;

iiii) il peut également arriver que la pression statique à la sortie du compresseur passe par un maximum lorsque le débit du compresseur décroît sans que se pose le problème de la difficulté de remplissage de la veine ; un tel fonctionnement conduit nécessairement à des instabilités longitudinales de basse fréquence (pompage) et limite également le domaine de variation de la pression.

4 - ETUDE THEORIQUE ET EXPERIMENTALE D'UN ETAGE DE COMPRESSEUR AXIAL A CALAGE VARIABLE

A titre d'exemple nous présentons les données expérimentales et l'interprétation théorique du fonctionnement d'un compresseur axial subsonique à faible vitesse de rotation, le fluide pouvant être alors considéré comme incompressible, dont la roue mobile, très éloignée du redresseur aval, comporte des aubes dont le calage peut être modifié à l'arrêt. Sur les huit calages étudiés, nous en analyserons trois correspondant au calage de base (n° 2), à un calage plus fermé (n° 5) et à un calage plus ouvert (n° 0). Une coupe du compresseur d'essai est donnée sur la figure 2. La figure 3, qui représente l'évolution en fonction du

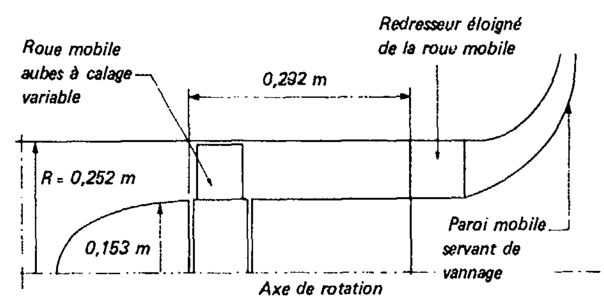

Fig. 2 – Coupe du compresseur d'essai. Vitesse de rotation N = 3000 tr/mn.

rayon réduit ξ des angles en sortie de roue mobile en axes relatifs, montre qu'ils se déduisent simplement des angles du calage de base par addition d'un angle constant.

La figure 4 représente pour ces trois calages le coefficient de perte de charge, rapport à la pression dynamique au rayon r de la vitesse d'entraînement, de la différence entre l'accroissement théorique de pression d'arrêt $\rho \omega r v$ et l'accroissement réel δP

$$\delta \psi_i = \frac{2(\rho \omega r v - \delta P)}{\rho \omega^2 r^2} \tag{12}$$

Des points expérimentaux nous avons déduit la corrélation théorique :

$$\delta\Psi_{i\,th} = 0{,}01\,(1+\varphi^2) + 0{,}25\,(1-\varphi\,tg\,\beta_*)^2 + (1-\varphi\,tg\,\beta_0)^2 + c_{f_p}\,\varphi^2 \qquad (13)$$

où $\varphi = \overline{w}/\omega r$ est le coefficient de débit moyen au rayon r ($\overline{w}$ est la demi-somme des vitesses débitantes amont et aval),

c_{f_p} est un coefficient, fonction du rayon, caractérisant les pertes secondaires et dont nous avons représenté l'évolution sur la figure 5,

β_0 est l'angle d'attaque nominal, et

β_* l'angle de sortie de l'écoulement compte tenu de l'écart flux-profil.

Utilisant avec ces données le simulateur numérique, nous avons pu comparer les valeurs calculées du coefficient local de débit aux valeurs mesurées au cours des essais (fig. 6). On notera une bonne concordance entre calcul et essais sauf aux plus grands débits pour lesquels d'ailleurs les essais deviennent imprécis du fait du faible accroissement de la pression d'arrêt.

En revanche le calcul montre qu'il existe dans chaque cas étudié un coefficient de débit moyen limite $(\varphi)_L$ pour lequel la courbe $\varphi(\xi)$ comporte une tangente verticale. Le débit correspondant limite vers les faibles valeurs des débits le domaine de fonctionnement pour lequel le fluide remplit complètement la veine.

L'expérimentation du compresseur a montré que pour les calages inférieurs au calage "2" il apparaissait alors un décollement sur le carter externe et que le fonctionnement global du compresseur restait apparemment sain pour des débits nettement plus petits que ce débit limite, malgré l'importante bulle de décollement pariétal.

En revanche pour les calages "2" à "5" un phénomène de décollement tournant apparaissait pour des débits inférieurs à cette valeur limite.

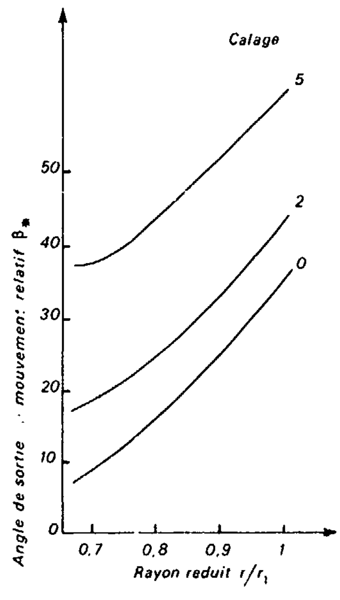

Fig. 3 – Variation de l'angle de sortie avec le rayon et l'angle de calage.

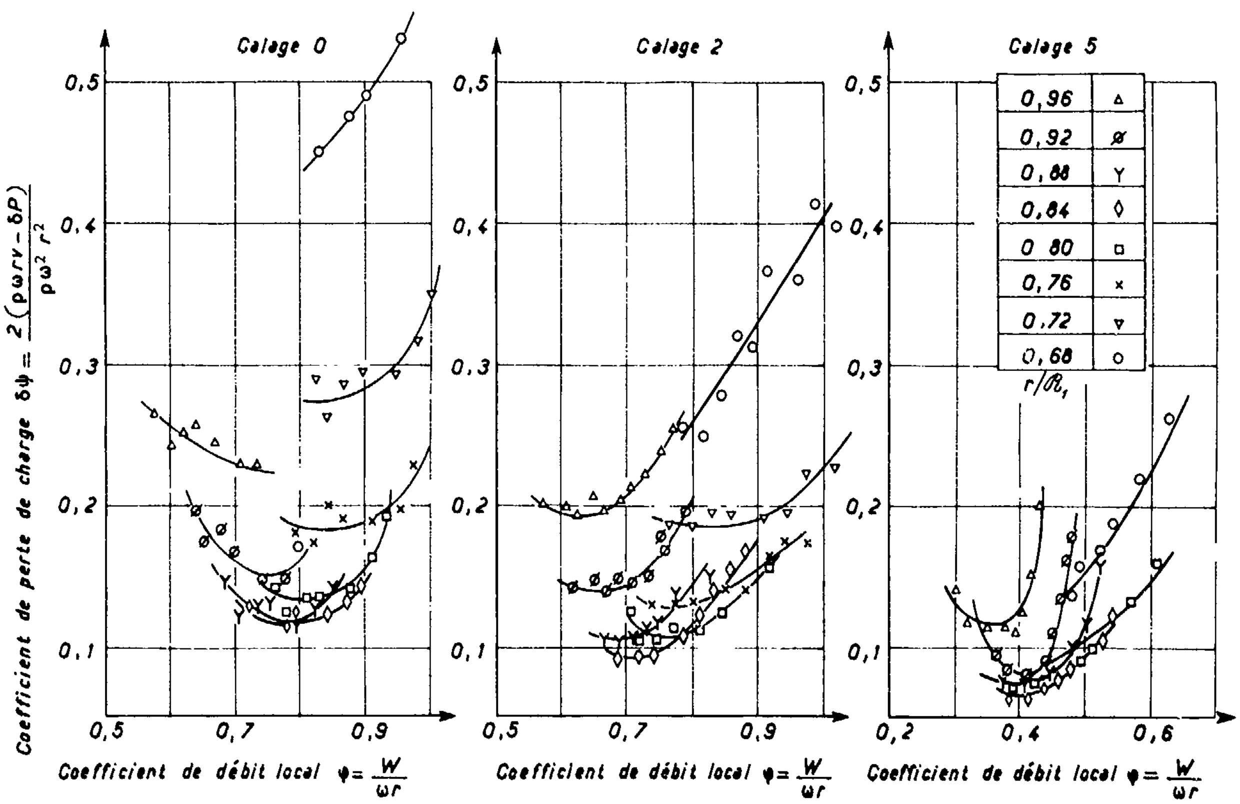

Fig. 4 – Influence du coefficient de débit local et du calage des aubes sur le coefficient de perte de charge à divers rayons réduits r/r_1.

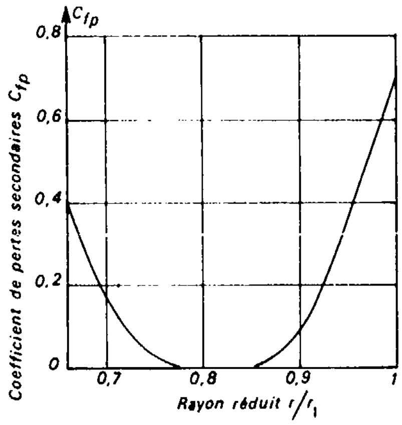

Fig. 5 — Variation en fonction du rayon réduit du coefficient de pertes secondaires C_{f_p}.

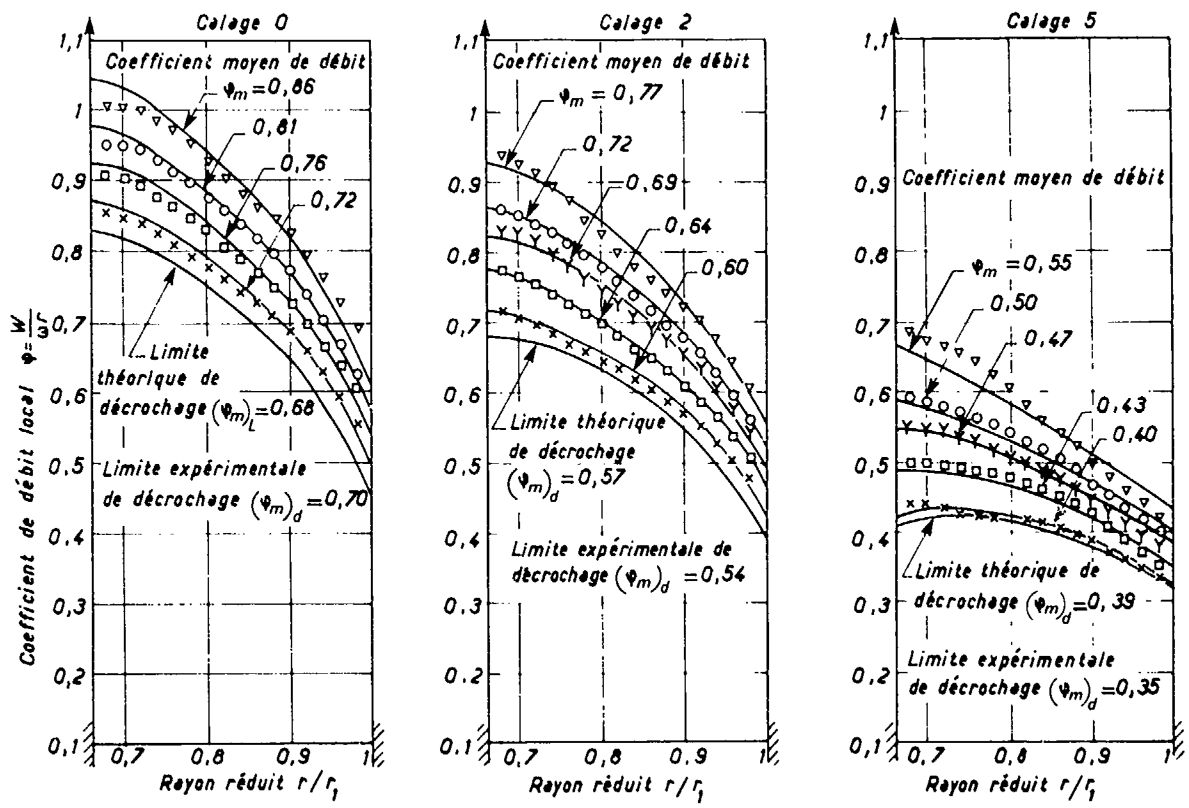

Fig. 6 — Comparaison des coefficients de débit calculés au moyen du simulateur (traits continus) et des points expérimentaux.

5 - PREVISION DE LA LIMITE DE DECROCHAGE ET DES REGIMES INSTATIONNAIRES

Permettant d'obtenir la répartition radiale de la vitesse à la sortie de la roue mobile, le simulateur numérique peut également définir le début de décrochage comme limite inférieure du débit pour lequel la veine est complètement remplie.

Se référant alors à des travaux antérieurs [9] où le phénomène de décollement tournant était décrit comme un régime où une zone de fluide mort animée d'une vitesse angulaire constante devait trouver un équilibre radial et azimutal, on a pu montrer que ce régime ne pouvait être stable en veine cylindrique que si, à des débits supérieurs ou au moins égaux au, débit limite, il existait un fonctionnement pour lequel le coefficient de pression statique:

$$\Psi_s = \frac{2(p - P_0)}{\rho \omega^2 r^2} \qquad (14)$$

était stationnaire en au moins un point.

Lorsque le redresseur est éloigné de la roue mobile, ce qui est le cas du compresseur étudié ci-dessus, on montre facilement que cette condition de stationnarité est satisfaite en tout point où a valeur de Ψ_s s'exprime par:

$$\Psi_s = \left(\frac{v}{\omega r}\right)^2 - \frac{dP_0}{\rho \omega^2 r \, dr} \qquad (15)$$

et n' en diffère que peu dans le cas où l'équilibre radial de la veine doit tenir compte du déplacement des surfaces de courant.

La figure 7 sur laquelle nous avons porté les courbes théoriques de Ψ_s montre qu'effectivement au calage "5" la condition (15) était déjà satisfaite pour un débit nettement supérieur au débit de décrochage, qu'au calage "2" ce n'est qu'au

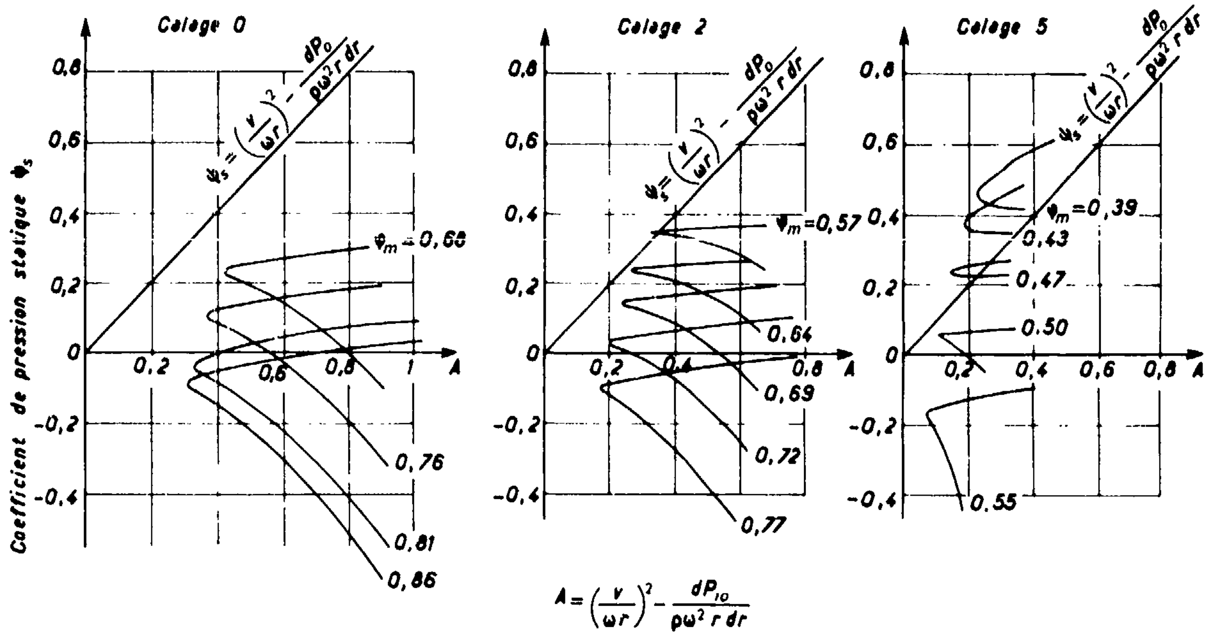

Fig. 7 – Recherche par le calcul de la condition de stabilité du régime de décollement tournant.

débit de décrochage que cette condition est satisfaite et qu'elle ne l'e.t pas du tout dans tout le domaine de fonctionnement pour le calage "0".

L'analyse de l'écoulement au moyen du simulateur numérique permet donc de prévoir dans un cas simple comme celui étudié ci-dessus, la limite de décrochage et la nature des régimes discontinus qui s'instaurent pour des débits inférieurs ou égaux au débit de décrochage.

6 - CONCLUSION

Nous avons essayé de montrer que, par la facilité d'introduction de diverses lois de pertes qui lui est propre, le simulateur numérique de turbomachines est un outil commode pour un bureau d'études pour analyser ou prévoir les performances d'un compresseur à géométrie variable. Son emploi optimal correspond au cas où l'on connaît un point de fonctionnement et que l'on recherche l'influence de la variation de divers paramètres géométriques et notamment de l'angle de calage.

La confrontation des prévisions théoriques aux résultats expérimentaux obtenus sur un compresseur à faible vitesse de rotation et à aubes à calage réglable à l'arrêt est très satisfaisante et la généralisation de la méthode au cas d'un compresseur fonctionnant dans le domaine compressible a déjà été entreprise.

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CHAMBRE DE COMBUSTION ANTI-NO_x
A ECOULEMENT AERODYNAMIQUE VARIABLE POUR TURBOREACTEUR

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RESUME

L'évolution des cycles des moteurs de TSS conduirait, avec des chambres de combustion fonctionnant de façon identique au décollage et en croisière à une élévation des émissions d'oxydes d'azote.
La SNECMA a défini dans des études à long terme une chambre à "écoulement aérodynamique variable" qui traite séparément la croisière et le décollage.
Cette chambre dite "à deux modules" comprend un module de ralenti et un module de croisière à bas niveau d'oxydes d'azote, utilisant la technique de combustion d'un prémélange prévaporisé.
Un module bi-dimensionnel conçu pour fonctionner sous haute pression a été expérimenté.
Des modifications de la géométrie de l'ensemble ont permis de surmonter le problème des remontées de flamme et de fonctionner jusqu'à une pression de 4.10^5 N/m^2 et une température d'entrée de 773°K.
Les résultats obtenus sont encourageants, mais des problèmes ardus restent à résoudre et il ne peut être envisagé d'appliquer la technologie proposée avant une dizaine d'années dans le meilleur des cas.

PLAN

1. INTRODUCTION

La technique de la chambre de combustion des turboréacteurs a beaucoup évolué dans les dernières années. Cette évolution a d'abord été due à l'évolution du cycle thermodynamique des nouveaux moteurs civils subsoniques, du type JT 9D, CF 6, RB 211, où l'utilisation de grands rapports de dilution ($\lambda \sim 5$) implique simultanément des rapports de pression et des températures entrée turbine élevées ($\pi \sim 25$ ou 30, $Tt_5 \sim 1400$°K).
Les moteurs militaires et les moteurs d'avions civils supersoniques suivent la même tendance.
Ce type de cycle thermodynamique a fort rapport de pression et température entrée turbine élevée dont la richesse nominale (rapport du débit masse carburant sur débit masse de l'air) est également augmentée, car la température sortie compresseur croît moins vite que la température entrée turbine, entraîne une modification sensible des performances requises de la chambre de combustion.

Du point de vue tenue mécanique et fiabilité de la machine, les conditions sont naturellement beaucoup plus sévères pour la chambre et pour la turbine. Pour la chambre, ceci résulte des niveaux élevés de pression absolue et de température sortie compresseur, puisque c'est la température de l'air qui sera utilisée en refroidissement. Pour la turbine, niveau élevé de température sortie compresseur et entrée turbine sont simultanément à l'origine du problème. C'est pourquoi la qualité de la répartition de températures sortie chambre joue un rôle essentiel dans ces nouvelles machines.
Parallèlement, un effort a été fait dans le domaine de l'encombrement. Les chambres de combustion actuelles sont plus poussées en rapport l/h (longueur sur hauteur du tube à flamme), ce qui est favorable du point de vue de refroidissement, mais défavorable du point de vue qualité de la répartition de températures entrée turbine.
La mise au point de telles chambres de combustion n'est pas allée sans problèmes. Encore aujourd'hui, on constate sur ces moteurs, que la tenue de la chambre est la raison la plus fréquente des taux d'arrêt en vol ou de dépose prématurée des moteurs. C'est pourquoi, cette technique doit être considérée avec une certaine prudence.
Enfin les problèmes d'environnement, d'abord apparus avec les émissions de fumées et de bruit, sont maintenant plus largement pris en compte, puisqu'on assiste à l'apparition de réglements limitant les émissions des moteurs d'avions à basse altitude. Des réglements limitant les émissions à haute altitude sont probables dans un avenir a moyen terme. Les nouvelles contraintes qui en résultent ou en résulteront pour les chambres de combustion, risquent d'avoir des conséquences profondes sur la conception de ces dernières.

2. LE PROBLEME DE LA POLLUTION DES MOTEURS D'AVIONS

Les gaz d'échappement des moteurs d'avions sont composés en majorité de produits considérés comme non polluants (O_2, N_2, H_2O, CO_2) et en faible quantité de produits qui peuvent être nocifs s'ils sont présents en quantité suffisante. Ce sont CO, les hydrocarbures imbrûlés, les oxydes d'azote, les aldéhydes, SO_2. A celà, s'ajoutent les particules de carbone, qui, parcequ'elles apparaissent sous forme de fumée noire, furent les premières à attirer l'attention du public, bien que n'étant pas nécessairement les plus dangereuses.

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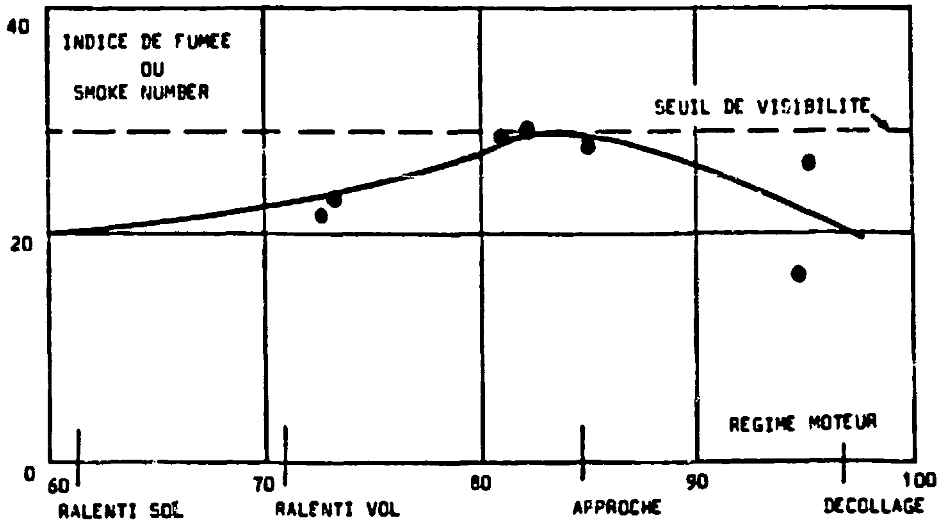
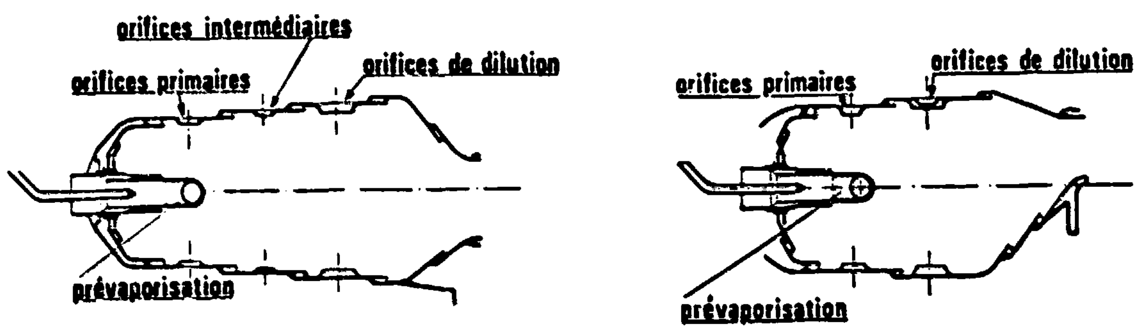


EVOLUTION TYPIQUE DU NIVEAU DE FUMEES
AVEC LE REGIME DU MOTEUR

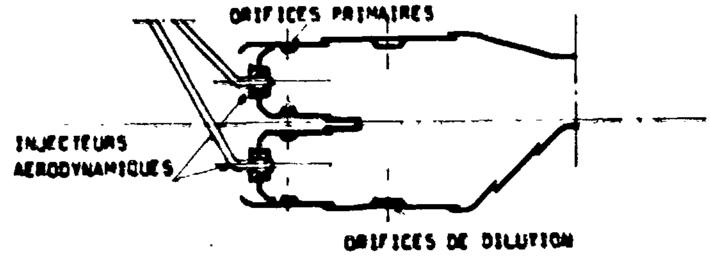
FIGURE N° 1:



COUPE DE CHAMBRE STANDARD SNECMA

COUPE DE CHAMBRE AVANCEE SNECMA

FIGURE N° 2:



COUPE D'UNE CHAMBRE A DEUX TETES

FIGURE N° 3:

Les règlements actuels ou futurs portent sur des limitations d'émissions de particules de carbone, CO, hydrocarbures et NO_x. Les aldéhydes et SO_2 ne sont pas pris en compte, les premiers parce qu'en très faible quantité et difficiles à mesurer, le SO_2 parce que dépendant uniquement de la teneur en soufre du carburant, n'étant donc pas de ce fait un problème de combustion.
Les particules de carbone se forment dans les zones riches de la zone primaire où règnent une température et une pression élevées. Pour éviter ou réduire leur formation, il faut obtenir un mélange air carburant le plus homogène possible dans la région où règnent les températures de flamme les plus élevées.
La figure 1 représente une courbe d'évolution typique du niveau d'émissions de fumée d'un moteur en fonction du régime. Le niveau maximum est généralement atteint vers le régime en plein gaz, ce qui n'est pas étonnant puisque la température sortie compresseur, la richesse, donc la température de flamme sont alors les plus forts.
Il est évident, compte tenu de ce qui vient d'être dit, que la solution optimale est celle du prémélange total air carburant en proportions plutôt pauvres, en amont de la combustion.

Les émissions de CO et hydrocarbures, sont liées au niveau du rendement de combustion, car ces produits sont des composés intermédiaires qui apparaissent dans le processus complexe de réactions chimiques de combustion.
Les paramètres qui influencent le rendement, donc les émissions de CO et hydrocarbures imbrûlés sont de deux natures différentes :
- Les conditions aéro-thermodynamiques dans lesquelles s'effectue la combustion, c'est-à-dire, pression, température, Mach moyen d'écoulement et richesse du mélange.
- La technique de combustion :
  structure de l'aérodynamique de la zone de stabilisation de flamme et des conditions de mélange air carburant.

Les conditions thermo-dynamiques sont d'autant plus défavorables qu'elles s'écartent du régime maximum.
Pression et température ont une influence directe sur la vitesse de réaction, de même que la richesse qui donne les vitesses de réaction les plus grandes quand les proportions du mélange sont stoechiométriques. Ce sera le cas, si la chambre est adaptée au décollage. Aux régimes partiels, les proportions seront d'autant plus sous-stoechiométriques qu'on approchera du ralenti.
Quant au second paramètre, l'optimum est représenté par un prémélange total du carburant, dans les proportions stoechiométriques pour les conditions les plus défavorables, c'est-à-dire, vers le ralenti.

Il existe deux processus de formation du NO_x :
- l'un dit du NO_x précoce (prompt NO_x)
- l'autre dit du NO_x thermique.

Le premier de ces deux processus est d'autant moins important par rapport au second que le niveau de température et de pression est élevé. Le second, comme son nom l'indique, dépend du niveau de température de flamme et du temps de séjour dans les zones chaudes. Le niveau de température étant le paramètre le plus important, il est nécessaire, pour réduire sensiblement les émissions de NO_x, d'abaisser le niveau de température maximum atteint dans la zone de combustion. Dans une chambre classique, l'adaptation est faite de manière à ce que la richesse dans cette zone primaire soit stoechiométrique pour des régimes proches du régime maximum, si ce n'est au régime maximum lui-même. Ceci dans le but d'avoir le meilleur rendement de combustion grâce aux vitesses de réaction les plus élevées.
De ce fait, il existe toujours quelque part des zones locales à richesse stoechiométrique, donc des zones où la température est très élevée, bien que la richesse moyenne puisse être différente. Pour les éviter, il faut créer un mélange homogène, dans un rapport largement sous-stoechiométrique, de manière à ce que nulle part le niveau de température puisse approcher la température stoechiométrique. La réduction en NO_x sera d'autant plus importante que la stabilisation de la flamme se fera à richesse plus basse, c'est-à-dire, aussi près que possible de la limite d'extinction pauvre. Toutefois, ceci ne doit pas se faire au prix d'un rendement exagérément bas.
On voit donc, que pour réduire le niveau de fumées, de CO, d'hydrocarbures imbrûlés, ou de NO_x, il faut tendre vers le prémélange carburant air de combustion. Pour CO et les hydrocarbures, il faut que le rapport de mélange soit stoechiométrique dans la zone de combustion pour les conditions pour lesquelles on cherche à obtenir, le rendement optimum. Par contre, pour les NO_x, il doit être aussi loin que possible du stoechiométrique.

3. PROGRAMME DE TRAVAIL SNECMA

Le problème de la réduction des NO_x est bien entendu le problème pollution le plus difficile à résoudre. Il peut être abordé sous trois aspects différents :
- Par l'injection d'eau dans la zone primaire de la chambre,
- Par la réduction du temps de séjour des gaz dans les régions chaudes du tube à flamme, c'est-à-dire, en amont de la zone de dilution.
- Par l'utilisation de la combustion en prémélange pauvre, comme cela a été expliqué précédemment.

3-1- Injection d'eau

Ce procédé n'est valable que pour des phases très limitées dans le temps de la mission de l'avion, décollage par exemple pour un avion subsonique. Et encore est-ce au prix de bien des inconvénients (installation des réservoirs, pompes, circuits spéciaux à bord des avions, etc...). Il est totalement exclu pour la croisière d'un avion supersonique, car la quantité d'eau à emporter serait du même ordre de grandeur que la quantité de carburant.
N'étant pas directement confrontés au problème de la pollution à basse altitude des avions subsoniques, nous avons décidé de ne pas approfondir cette technique qui d'ailleurs, du point de vue combustion, ne semble pas poser de problèmes majeurs.

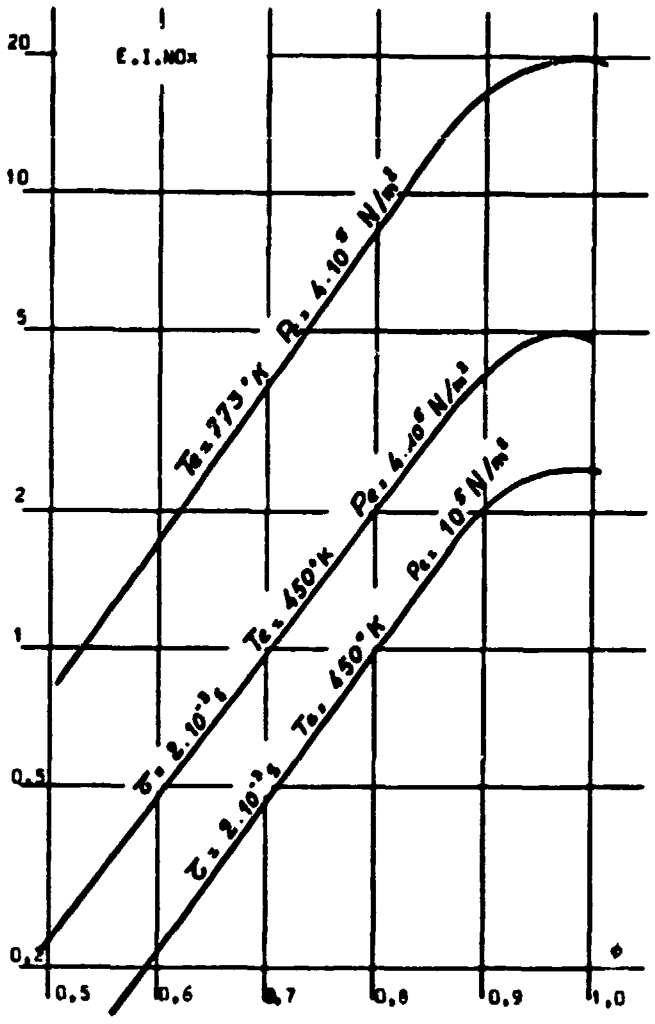


FIGURE N° 4: EVOLUTION DES INDICES D'EMISSION DE NOx AVEC LE RAPPORT D'EQUIVALENCE ϕ

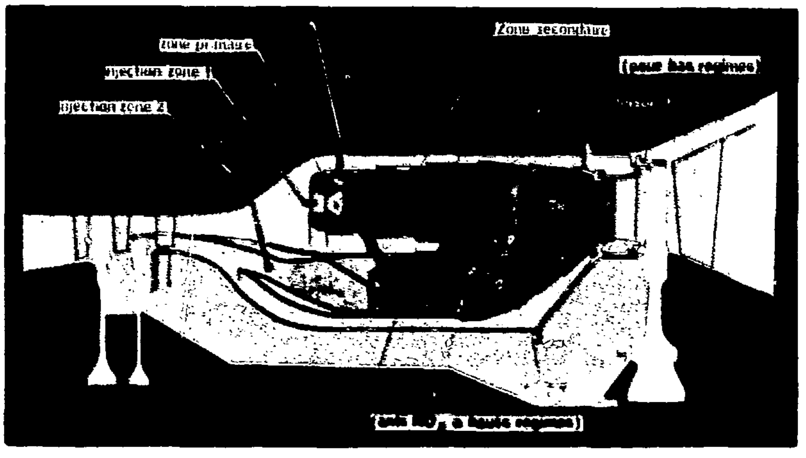


FIGURE N° 5: CHAMBRE A DEUX MODULES

3-2- Réduction du temps de séjour dans les zones chaudes du tube à flamme

C'est une technique applicable sur les chambres de combustion "classiques". Sur la figure 3 nous avons schématiquement représenté la coupe d'un tube à flamme standard à côté de la coupe d'un tube à flamme faisant appel à cette technique.
On peut voir qu'elle s'accompagne d'un raccourcissement sensible du tube à flamme, ce qui correspond à un état plus avancé de la technique, les autres performances (répartitions de températures, rendements, allumage et stabilité en altitude, etc..) ne devant pas être sensiblement altérées. La réduction possible de l'indice d'émission de NO_x se situe entre 10% et 30% aux régimes maxi. Nous poursuivons le développement de ce type de solution.
Une autre façon d'appliquer cette technique est, pour les chambres de suffisamment grande taille de dédoubler la zone primaire en deux parties superposées, autrement dit, de faire une chambre à double tête (voir fig. 3). Les proportions géométriques restant homothétiques, il s'ensuit une division par environ deux de la longueur de la zone primaire, donc du temps de séjour dans cette zone.
Ce type de chambre, nettement plus complexe que le précédent, pose en plus des problèmes de répartitions de températures, d'allumage et de stabilité en altitude, pour des gains en NO_x correspondant à une division par deux environ. N'ayant pas d'application directe en vue pour cette technique (type JT 9D, CF 6 par exemple) nous n'avons pas entamé pour le moment de programme de recherche dans cette voie, qui ne nous paraît vraiment rentable que si la technique du prémélange pauvre est appliquée pour une de ces deux têtes.

3-3- Technique du prémélange pauvre

Son intérêt a été expliqué dans un paragraphe précédent.Etant donné ses limites de fonctionnement restreintes par rapport à une chambre classique, il est nécessaire d'adjoindre à une telle chambre adaptée aux régimes élevés, une autre chambre adaptée pour les autres conditions de fonctionnement (ralenti, régimes partiels). Il s'agit donc de faire une chambre de combustion à adaptation variable, ce qui pourrait être réalisé par une géométrie variable qui modulerait les débits d'Air entre les différentes zones du tube à flamme selon les régimes, mais qui dans ce cas peut être réalisée dans une géométrie fixe en jouant sur les adaptations différentes de deux modules différents. Les problèmes que pose une telle technique sont analogues à ceux de la chambre à double tête évoquée dans le paragraphe précédent, avec en plus, les problèmes propres liés à l'utilisation du prémélange prévaporisation totale en amont de la zone de stabilisation de flamme.
Ce sont ces derniers problèmes auxquels nous avons décidé de nous attaquer d'abord, afin de situer les limites de réduction de NO_x possibles avec cette technique, à priori, la plus prometteuse de toutes celles qui viennent d'être évoquées, et aussi les limites dues à des problèmes qui lui sont propres, comme la sécurité, la fiabilité, la complexité technologique, etc...

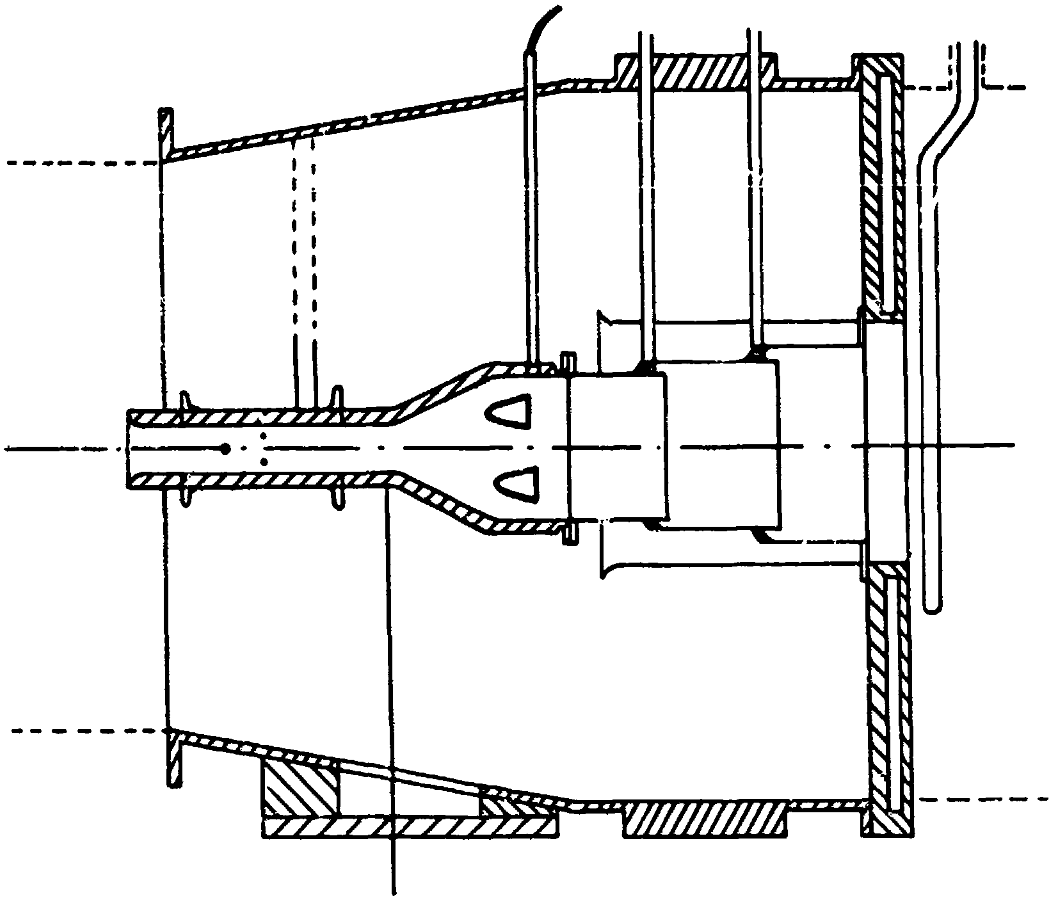
4. CONCEPTION DE LA CHAMBRE A DEUX MODULES

Les considérations précédentes ont montré l'intérêt d'une combustion en mélange pauvre d'un prémélange d'air et de carburant, avec un temps de séjour modéré afin de réduire les émissions d'oxydes d'azote.
La figure 4 illustre ce principe ; elle montre la variation de l'indice d'émission d'oxyde d'azote résultant de la combustion d'un prémélange air kérosène stabilisée par une grille perforée en fonction du rapport d'équivalence du mélange.
De plus, au cours du processus de combustion de gouttelettes de combustible, il existe toujours une zone de réaction stoechiométrique au voisinage de ces gouttelettes.Il ne suffit donc pas de réaliser un mélange à richesse uniforme d'air et de gouttelettes de carburant, encore faut-il prévaporiser le carburant.
Il faut donc réaliser, en amont du dispositif de stabilisation de la combustion un prémélange uniforme d'air et de carburant prévaporisé.
Par ailleurs, comme nous l'avons vu plus haut, il faut réduire les émissions d'oxyde d'azote, réduire le temps de séjour à haute température, donc concevoir un foyer à faible temps de recirculation et faible temps de séjour global.

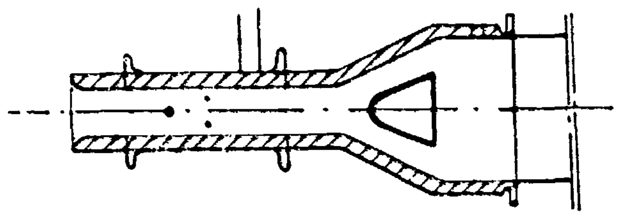
Une chambre de combustion d'architecture classique réalisée suivant ces principes est convenable, mais elle présenterait les inconvénients suivants :
- elle ne pourrait pas fonctionner de manière satisfaisante dans toute la gamme de richesses imposées par les variations de régime d'un turboréacteur, du fait de ses médiocres limites d'extinction pauvre.
- elle ne pourrait satisfaire aux impératifs de réduction de l'oxyde de carbone et des hydrocarbures imbrûlés pour lesquels il est nécessaire, au contraire, de rechercher des temps de séjour élevés et une richesse de combustion assez voisine du stoechiométrique.

En conséquence, nous avons pensé,
à concevoir une chambre dans laquelle l'écoulement d'air est divisé en deux parties à la sortie du compresseur HP ;
une partie alimente un module ralenti (module A) de type classique dans lequel la combustion s'effectue avec un temps de séjour long et à une richesse élevée au ralenti,
l'autre partie alimente un module de croisière (module B) du type à prémélange dans lequel la combustion d'un prémélange air combustible prévaporisé s'effectue dans un temps de séjour bref et à richesse faible.

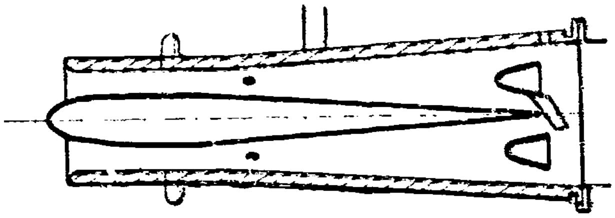
CONFIGURATION A (SCHEMA DU MONTAGE D'ESSAI)



CONFIGURATION B



CONFIGURATION C



DIVERSES CONFIGURATIONS DU MODULE BIDIMENSIONNEL

FIGURE N° 6:

Une telle chambre se présenterait comme le montre la figure 5.
On distingue : le module A, muni d'un système d'injection de type classique, dont la zone primaire est adaptée à une richesse proche du stoechiométrique au ralenti, le complément d'air étant introduit par des orifices de dilution.
Le module B dans lequel la combustion pauvre est stabilisée par un ou plusieurs stabilisateurs en V et alimenté en carburant par des rampes du type de celles utilisées en rechauffe. Le mélange des deux écoulements est réalisé par mélange des jets issus du module A avec l'écoulement du module B.
Les modules A et B fonctionneraient alors par exemple dans les phases suivantes :

Phase de vol	Ralenti	Décollage	Montée	Croisière	Approche
Module A	X	X	X		X
Module B		X	X	X	

Des conditions de fonctionnement typique du module B, correspondant à un cas de vol en croisière supersonique à haute altitude sont les suivantes :

$T_{entrée} = 800°K \quad P_{entrée} = 10.10^5\ N/m^2 \quad \phi = 0,5$

5. RESULTATS EXPERIMENTAUX

Dans le but d'étudier les niveaux d'émissions en NO_x et le rendement de combustion que peut permettre d'atteindre un tel module à prémélange air combustible prévaporisé ainsi que les problèmes liés à son fonctionnement, la SNECMA a entrepris la fabrication et les essais d'un élément du module B.

5-1- Montage d'essai

Le matériel expérimenté est constitué d'un ensemble bi-dimensionnel comprenant :
- une entrée d'air de section rectangulaire 320 x 33 mm dans laquelle le nombre de Mach est voisin de 0,24 et où sont placées les rampes d'injection de carburant.
- un diffuseur ralentissant l'écoulement jusqu'à un nombre de Mach d'environ 0,08 en aval duquel sont placés les stabilisateurs de flamme réalisant un blocage de 0,5.
- un tube à flamme refroidi par film cooling.

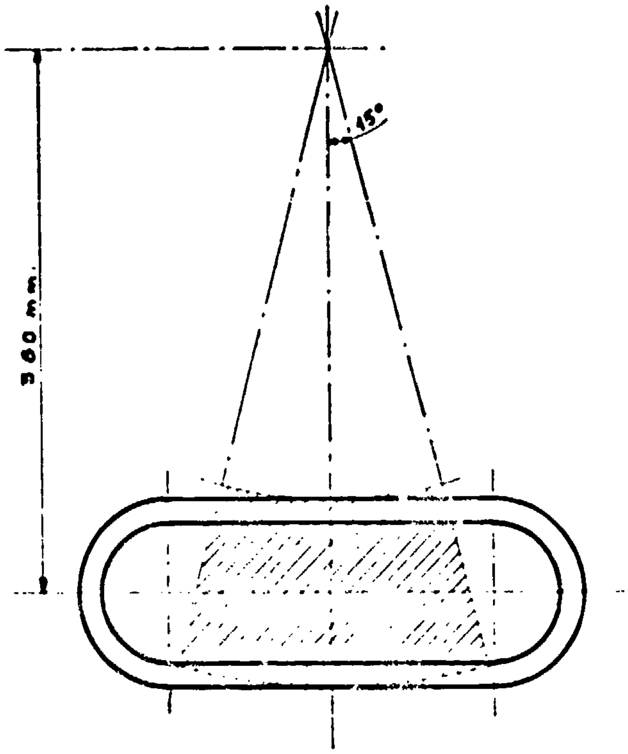
Compte tenu des problèmes rencontrés, trois configurations principales ont été expérimentées qui sont présentées figure 6.
L'injection du carburant se fait à environ 180 mm en amont des accroche-flammes par une ou plusieurs rampes à section droite circulaire ou elliptique.
Diverses configurations d'injection, équi-courant, contre-courant et injection perpendiculaire à l'écoulement d'air ont pu être testées.

5-2- Dispositifs de mesure

Les prélèvements de gaz à la sortie du tube à flamme sont effectués par une sonde refroidie par eau comportant quinze orifices de prélèvement moyennés ; la sonde pivote autour d'un axe situé à 380 mm au-dessus du centre de la veine et le balayage angulaire peut s'effectuer soit en continu soit tous les 10°.
La zone explorée par rapport à la surface totale de sortie du tube à flamme est représentée figure 7.
Les gaz sont aspirés par une pompe à membrane qui refoule dans les analyseurs en continu, permettant le dosage de CO et CO_2 par absorption infrarouge, des hydrocarbures imbrûlés par ionisation de flamme et des oxydes d'azote par chimiluminescence.

5-3- Résultats d'essais

L'un des problèmes les plus délicats résultant de l'utilisation d'un écoulement prémélangé d'air et de carburant prévaporisé est la difficulté d'éviter toute auto-inflammation ou remontée de flamme en amont des stabilisateurs de flamme.
De fait, avec la configuration A ayant un diffuseur à fort angle au sommet, nous avons observé à pression atmosphérique pour une température d'entrée de 773°K, des remontées de flamme qui se stabilisent à la paroi du diffuseur dans les zones décollées.
La configuration B dans laquelle le stabilisateur a été déplacé vers l'amont, a permis d'éviter ce phénomène à pression atmosphérique, mais il est apparu à une pression de $2,7.10^5\ N/m^2$. Seules certaines configurations d'injecteurs donnant par ailleurs de médiocres résultats en oxydes d'azote, donc réalisant sans doute un mauvais prémélange ont permis de fonctionner dans ces conditions.
La configuration C, conçue dans le but de supprimer les zones décollées, se caractérise par deux diffuseurs de faible angle au sommet superposés et séparés par un corps central. Cette configuration a effectivement donné toute satisfaction à $T_e = 773°K$ jusqu'à une pression de $4.10^5\ N/m^2$.



ZONE D'EXPLORATION DE LA SONDE EN SORTIE DE LA "DEUX MODULES"

FIGURE N° 7:

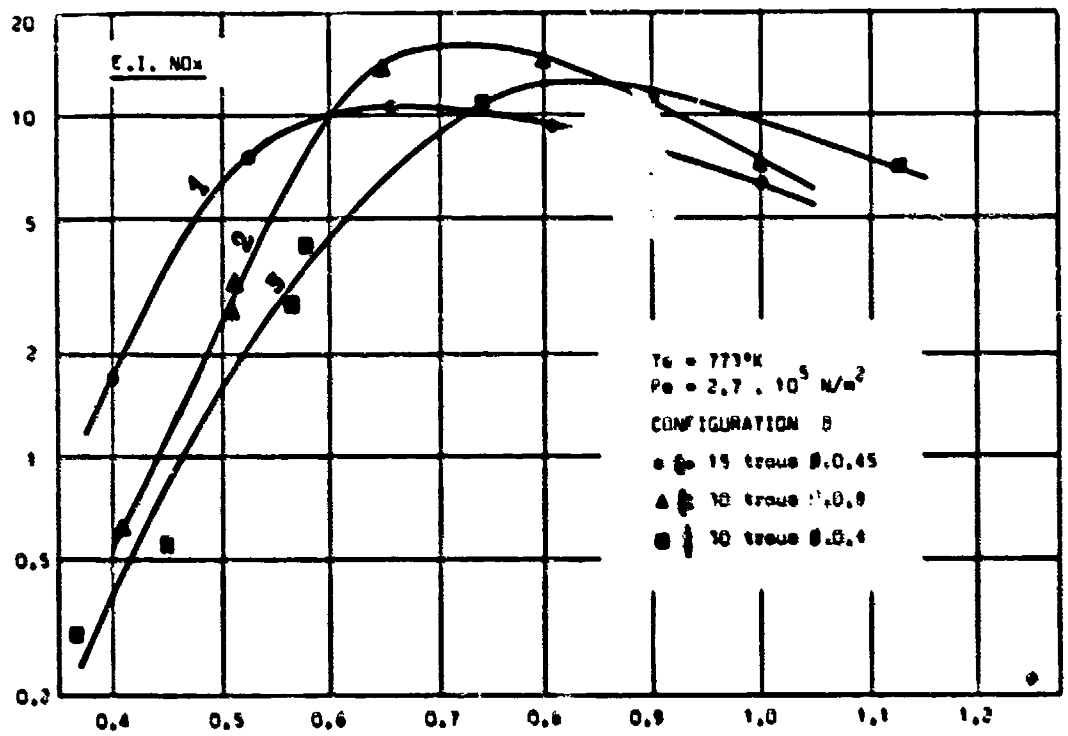


FIGURE N° 8: INDICES D'EMISSION DES NOx MESURES (A & B)

Indices d'émission et rendements de la configuration B -

Les indices d'émission d'oxyde d'azote obtenus dépendent dans une large mesure du système d'injection utilisé.
La figure 8 montre les résultats obtenus pour trois types d'injection dans les conditions de fonctionnement Te = 773°K, $Pe = 2,7.10^5$ N/m^2.
En abscisse est porté le rapport d'équivalence ϕ col calculé en rapportant le débit de carburant injecté au débit d'air entrant dans le conduit de mélange ; en ordonnée sont portés les indices d'émission d'oxyde d'azote mesurés en intégrant les valeurs de concentrations gazeuses mesurées par l'exploration en continu effectuée a la sortie du tube à flamme.
Les types de rampes expérimentées sont les suivants :

Courbe 1 : 1 rampe d'injection injectant à contre-courant et comportant 15 orifices de 0,45 mm de diamètre.
Courbe 2 : 2 rampes d'injection à contre-courant et comportant chacune 15 orifices de 0,8 mm de diamètre
Courbe 3 : 2 rampes d'injection injectant perpendiculairement à l'écoulement et comportant chacune 15 orifices de 0,4 mm de diamètre.

On constate sur cette planche que pour un même stabilisateur et un même temps de séjour (2,3 ms environ à ϕ = 0,5) l'indice d'émission d'oxyde d'azote varie de 1,4 à 6 à la richesse nominale ϕ = 0,5 suivant le type d'injection, donc suivant la qualité du prémélange.
Les inefficacités enthalpiques obtenues avec la configuration B et les dispositifs d'injection décrits ci-dessus à Te = 773°K et $Pe = 2,7.10^5$ N/m^2 sont présentés figure 9.
On constate que les rendements de combustion des configurations 2 et 3 sont très voisins pour ϕ variant entre 0,4 et 0,6.
Les rendements de combustion sont calculés suivant la formation classique

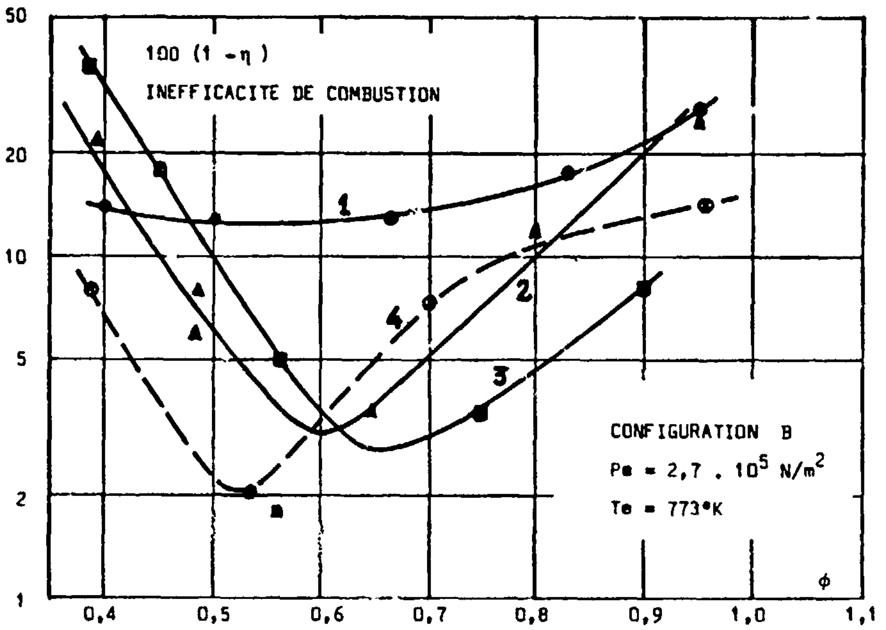
$$\eta_H = 1 - (0,24\ EICO + 1,16\ EICH_4 + 0,047\ EINO_x).\ 10^{-3}$$

qui se rapporte à une combustion idéale ne produisant que du CO_2 et de l'eau. En fait, pour un degré d'avancement de la réaction de 1 qui représente la limite accessible, c'est l'état d'équilibre qui est atteint, le rendement η_H est alors loin d'être égal à 1 lorsque ϕ tend vers la valeur unité.
Ceci pourrait expliquer la remontée des rendements lorsque ϕ croît au-delà de 0,6.
D'autre part, les valeurs relativement faibles de rendement obtenus ne doivent pas surprendre, compte tenu de la faible longueur du tube à flamme (distance entre bord de fuite des stabilisateurs et plan de mesure voisine de 280 mm).
Nous avons cherché à améliorer les rendements obtenus et dans ce but, expérimenté un stabilisateur de même blocage mais muni de bras radiaux. Les résultats obtenus sont représentés par la courbe 4 et une valeur de 97,5 % a été mesurée à Te = 773°K et $Pe = 2,7.10^5$ N/m^2 ce qui doit permettre d'obtenir un rendement satisfaisant dans les conditions nominales de fonctionnement.

Indices d'émission et rendement obtenu avec la configuration C

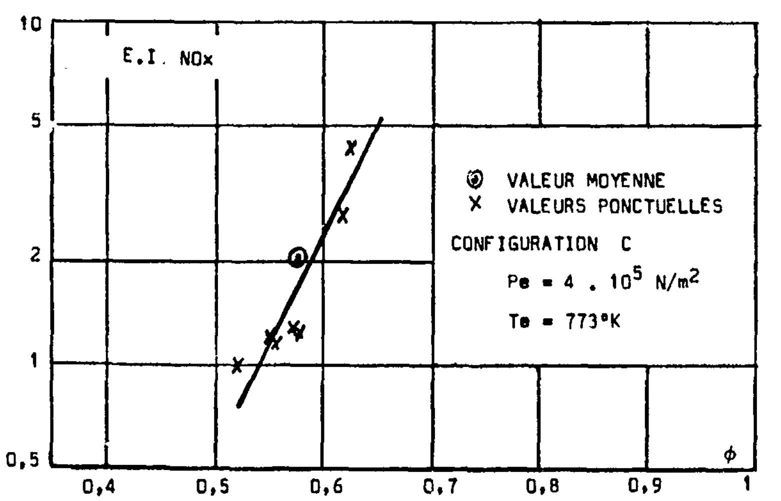
La configuration C qui a permis de résoudre le problème des remontées de flamme a été testée jusqu'à $Pe = 4.10^5$ N/m^2 à Te = 773°K et a donné les résultats suivants.
Au point de fonctionnement réalisé $Pe = 4.10^5$ N/m^2, Te = 773°K et ϕ = 0,53 un indice d'émission d'oxyde d'azote de 2 a été obtenu avec un rendement enthalpique de 96,5 % ceci avec deux stabilisateurs sans bras radiaux et deux rampes d'injection injectant perpendiculairement à l'écoulement et comportant chacune 30 orifices de 0,4 mm de diamètre.

Si l'on se refère à la courbe de la figure 4 tracée à partir de résultats expérimentaux d'un foyer à prémélange et stabilisateur par grille perforée extrapolée à la pression de 4.10^5 N/m^2 et à Te = 773°K (par $EINO_x\ (T_1\ P_1) = EINO_x\ (T_o\ P_o) \times e^{\frac{T1 - To}{230}} \cdot \frac{P1}{Po}$)
on constate qu'il serait théoriquement possible d'atteindre dans ces conditions un indice d'émission d'oxyde d'azote d'environ 0,8. Il est vraisemblable qu'un meilleur mélange permettrait sans doute d'approcher ce résultat.
Les essais réalisés ont d'ailleurs révélé l'existence d'un gradient de richesse non négligeable d'un bord à l'autre du tube à flamme en raison de la variation de la pression d'injection depuis l'entrée de la rampe d'injection jusqu'à son extrémité, compte tenu de la faible section droite des rampes utilisées.
L'exploration tous les 10 degrés du plan de sortie du tube à flamme a permis de déceler ce phénomène et nous a permis de tracer l'évolution de l'indice d'oxyde d'azote en fonction de la richesse locale mesurée. La figure 9 montre les résultats obtenus. Sur cette figure sont présentées d'une part, les valeurs locales pour lesquelles le rapport d'équivalence mesuré varie de 0,53 à 0,62 et l'indice d'oxyde d'azote de 1 à 4,5 et la valeur moyenne déduite de l'analyse en continu ϕ mesuré = 0,575 $EINO_x$ = 2,1, la valeur du ϕ col mesurée étant de ϕ = 0,535, l'écart de 7% entre ces deux dernières valeurs de ϕ provenant vraisemblablement de l'exploration incomplète de la section de sortie, compte tenu de la géométrie de la sonde. (fig. 7)
On peut donc espérer avec une homogénéité plus grande atteindre un indice d'oxyde d'azote voisin de 1 pour ϕ = 0,5 soit une valeur très proche de la valeur théorique estimée.
Néanmoins, le résultat acquis à l'heure actuelle sur le module à bas niveau d'oxyde d'azote est très encourageant, puisqu'il permet d'espérer atteindre un indice d'émission d'environ 3,2 à 4 dans les conditions nominales représentatives dans le cas de la croisière supersonique d'un avion TSS.



RENDEMENTS ENTHALPIQUES MESURES (A & B)

FIGURE N° 9:



INDICES D'EMISSION DES NOx (C)

FIGURE N° 10:

6. PROBLEMES A RESOUDRE

Hormis les problèmes de remontée de flamme dont nous avons déjà parlé et qui ont été résolus, du moins jusqu'à une pression de fonctionnement de 4.10^5 N/m^2 et le problème de l'homogénéité de la richesse du mélange évoqué plus haut, les problèmes à résoudre sont nombreux :

Problèmes liés au seul module de croisière

- Risques d'auto-inflammation du mélange air carburant en amont des stabilisateurs
- Tenue thermique des stabilisateurs

Problèmes liés à l'association du module de ralenti et du module de croisière.

- Répartition de température à l'entrée du distributeur turbine et variations en fonction des régimes moteur.
- Régulation de l'injection du carburant entre les deux modules
- Performances de réallumage en altitude

Il est certain que ces problèmes vont se poser au fur et à mesure de l'avancement des travaux et ce d'autant plus que l'utilisation d'une telle chambre de combustion serait envisagée pour des moteurs à taux de compression élevé.
Une première approche du problème du mélange par jets des écoulements issus des deux chambres a été réalisée avec la collaboration de l'ONERA. Les résultats obtenus ont montré la difficulté de réaliser une répartition de température satisfaisante en amont du distributeur turbine.
L'étude de l'inflammation spontanée d'un mélange air carburant entreprise par l'ONERA à notre demande dans le cadre d'un autre programme, a permis d'obtenir des indications précieuses sur le comportement à pression et température élevées d'un prémélange prévaporisé au kérosène.
Il est vraisemblable qu'un compromis sera nécessaire aux pressions élevées entre le temps nécessaire pour réaliser un mélange aussi parfait que possible et le temps de séjour limite admissible, compte tenu des phénomènes d'auto-inflammation.

7. CONCLUSION

Nous venons d'exposer la manière dont a été abordé à la SNECMA le problème de la réduction des émissions de NO_x par les chambres de combustion de turboréacteurs.
Nous avons montré que des possibilités potentielles de réduction de ces émissions pouvant aller jusqu'à des divisions par 4 existent, grâce à la technique du prémélange prévaporisé.
Les travaux ne sont pas actuellement dans un état suffisamment avancé pour permettre de situer les limites exactes d'utilisation de cette technique. Il est possible qu'elle ne puisse pas être envisagée sur des moteurs à taux de compression très élevé. Le temps nécessaire pour que son utilisation sur moteur d'avion de transport civil, avec les conditions de sécurité et de fiabilité requises est difficile à évaluer aujourd'hui.
Néanmoins, nous ne pensons pas qu'elle puisse être envisagée avant une dizaine d'années. D'ici là, bien des travaux de recherche et de développement seront nécessaires.

G.Winterfeld

Since the combustion efficiency of your two-module combustor is very high, being well above 99% in a wide range of mixture ratios, there arises the question of measuring or calculating the combustion efficiency. Could you please comment on how the figures have been obtained?

Author's Reply

En fait nous utilisons une formule de calcul du rendement enthalpique qui se réfère à une combustion idéale ne produisant que du CO_2 et de l'eau; de ce fait pour des richesses de l'ordre de 1, l'état d'équilibre est tel que le rendement calculé de cette manière n'est pas proche de 1 mais inférieur à 1. Ceci explique, d'ailleurs, que sur la vue que je vous ai montrée les inefficacités remontent pour des richesses de 0,8 à 1 compte tenu de la méthode de calcul du rendement basée sur les indices d'émission de CO, C_xH_y et de NOx. Nous sommes conscients qu'il y a là un problème de calcul du rendement dans les conditions de richesse élevée.

D.Kretschmer

(1) Pourriez-vous me donner des détails supplémentaires sur la distance entre les injecteurs et les stabilisateurs ainsi que le temps de résidence et le temps nécessaire pour l'évaporation des gouttelettes?

(2) Est-ce que le carburant est préchauffé?

Author's Reply

(1) La distance entre le nez des stabilisateurs et l'injection est d'environ 180 mm. Le temps de vaporisation et de mélange disponible doit être de l'ordre de la milliseconde, le mélange air-carburant ayant une vitesse d'environ 150m/s dans la veine d'entrée.

(2) Non, le carburant n'est pas préchauffé. Il est chauffé et vaporisé par l'air. D'après un calcul approché nous estimons qu'un pourcentage d'environ 70% à 80% du carburant injecté est prévaporisé.

D.R.Hounslow

You appear on the second cycle chamber to have a relatively narrow band of efficient operation with respect to changes in second cycle mixture strength. In view of this, how do you see the problems of fuel staging – over the flight envelope between primary cycle only operating at idle through off-design points before the second stage operation has been fully developed?

Author's Reply

Nous n'avons pas fait d'études complètes du fonctionnement des deux modules. A priori, la répartition du carburant doit se faire de la façon suivante: au ralenti, seul le module de ralenti fonctionne. Au décollage et en montée, compte tenu de la richesse nécessaire les deux modules, de ralenti et de croisière, sont en fonctionnement simultané; en croisière le module de ralenti est éteint et le module de croisière anti NOx est seul en fonctionement à une richesse d'environ 0,5 de telle sorte que la richesse moyenne à la sortie soit la richesse requise pour les conditions de vol en croisière.

DESIGN FEATURES FOR A PRE-MIXED VARIABLE AREA COMBUSTOR.

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1. INTRODUCTION

The feasibility of a pre-mixed, variable area combustor has been justified by previous work [1 & 2] both in terms of construction and its effectiveness for pollution control. The theoretical aspects were discussed in [1], whereas the second paper [2] dealt with some more practical aspects and also, some of the design considerations.

It is the objective of the present work to justify some of the design features by presenting them in more detail, especially with respect to the construction of the primary zone and its ancilliary parts. In addition, a short section has been included to emphasize the practicability of the control of the three major pollutants, oxides of nitrogen, carbon monoxide and unburnt hydrocarbons.

2. GENERAL DESCRIPTION OF THE COMBUSTOR

The combustor has been designed to meet the requirements of a small, single-can engine having a reverse flow, tubular combustor. The size and general geometry are indicated in Fig. 1. From both manufacturing and performance viewpoints, the entry section of the chamber demands very careful attention. The following features are the most critical ones -

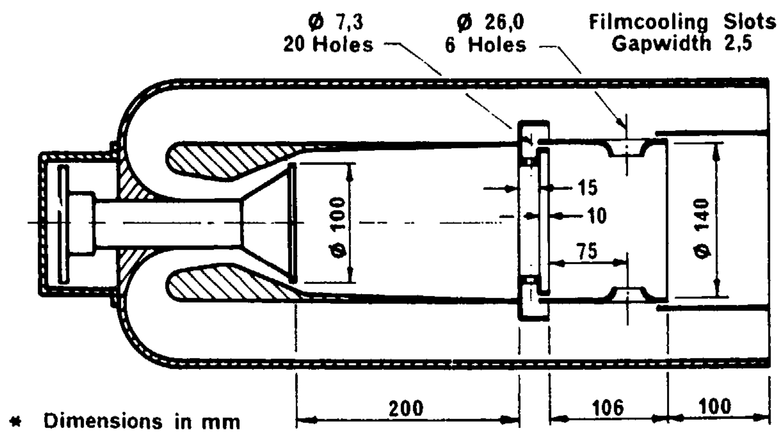

Fig. 1 Layout of the Selected Combustor

a) The air entry requires a smooth aerodynamic profile.

b) Until it spills over the baffle, it is essential that the airflow should always be accelerating. This will minimise the possibility of flashback of the flame from the primary zone to the fuel injectors.

c) In order to meter accurately the flow of air over the baffle, the conical profile will require an accurate finish (±0,025 mm).

d) To ensure a good strong recirculation downstream of the baffle, the latter must be finished with a sharp edge.

e) The combustion performance of a chamber is very dependent upon the pressure drop available to create a stable aerodynamic flow pattern. A minimum such value is about 2,5 % of the inlet pressure. For the purpose of this design, a 3,0 % pressure drop has been selected.

f) To facilitate a smooth light-up, a centrally positioned pilot fuel atomiser will be located within the downstream face of the baffle.

g) The materials of construction will be those used for existing conventional combustion systems (e.g. stainless steels, nickel-base alloys etc.).

The air admitted to the primary zone is controlled solely by the movement of the baffle; the remaining air ports (secondary holes, dilution holes, film cooling devices) have constant cross-sectional areas. Thus the overall pressure drop through the system is not independent of operating conditions. Hence, the 3 % pressure drop committment can only be met at a single design condition (No. 4 in Table I). The full range of engine operating points selected for design consideration is given in Table I, condition No. 8 being representative of the ground idling case.

A wide-cut kerosene has been selected as the fuel and its relevant properties are listed in Table II. For calculation purposes, the air has been taken as being dry, free from CO_2, and having a molecular weight of 29. The ratio inerts/oxygen has been assumed to be 3,76.

3. DESIGN FEATURES OF THE COMBUSTOR

In a similar manner to conventional ones, the combustor may be considered as comprising three zones, primary, secondary, and dilution. The preliminary sizing of the chamber was described in [2], as was the assessment of the air distribution.

3.1 Primary Zone Considerations

The size and critical dimensions of the primary region were established in the former work; relevant data are listed in Table III. The zone is designed to operate at a near-constant equivalence ratio of 0,8, and stops are placed so as to restrict the baffle movement such that the primary zone mixture will maintain combustion at all conditions, even if the movement mechanism fails.

TABLE I

OVERALL OPERATING CONDITIONS OF COMBUSTOR

Cond. No.	Chamber Inlet Press. P_2	Chamber Inlet Temp. T_2	Fuel Flow $\dot{m}_f$	Equiv. Ratio ϕ_0	Air/Fuel Ratio A/F	Chamber Air Mass Flow $\dot{m}_a$	$\frac{\dot{m}_a\sqrt{T_2}}{P_2}$
	bar	K	gmol/s	-	(mass)	gmol/s	$\frac{\text{gmol.K}^{\frac{1}{2}}}{\text{s.bar}}$
1	6,68	735	0,247	0,234	62,86	75,05	304
2	7,88	779	0,326	0,274	53,62	84,49	299
3	9,00	815	0,407	0,311	47,33	93,12	295
4*	9,60	810	0,437	0,313	47,02	99,32	294
5	10,68	803	0,508	0,323	45,55	111,85	296
6	12,99	789	0,611	0,323	45,56	134,55	291
7	14,99	787	0,709	0,324	45,36	155,44	291
8	1,24	294	0,026	0,176	83,63	10,51	145

* Note: This condition has been arbitrarily chosen as the Design Condition

TABLE II

FUEL PROPERTIES

Composition $C_{10}H_{20}$	Molecular Weight 140	S.G. @ 289 K 0,762
Aniline Gravity Constant 7030 °F	Calorific Value (nett) 43210 J/g	

Distillation				
Distillation	IBP 330 K		Average B.P. (volume)	446 K
	10% 389 K		(molal)	439 K
	50% 441 K		Normal Bubble Point	389 K
	90% 487 K		Normal Dew Point	470 K
	FBP 510 K		Stoichiometric A/F	14,71 by mass

			Kinematic Viscosity	
Critical Temperature	625	K	311 K	1,00 mm^2/s
Critical Pressure	34,5	bar	298 K	1,16
Surface Tension @ 298 K	2,30	J	263 K	1,97
Latent Heat (Vap.) @ 1,0 bar	37660	J/gmol	240 K	2,97
Smoke Point	26	mm		

Specific Heat	Vapour	Liquid
273 K	206 J/gmol	270 J/gmol
373 K	260	323
473 K	312	377

TABLE III

PRIMARY ZONE COMBUSTOR DATA

Cond. No.	% of $\dot{m}_a$ for ϕ = 0,8	$\dot{m}_{a,PZ}$	$T_{max,PZ}$ ϕ = 0,8	Air Injection Area Total	Air Injection Area Baffle	Duct Dia. at Baffle	$\Delta P/P_2$ ϕ = 0,8
		gmol/s	K	mm^2	mm^2	mm	%
1	29,38	22,06	2415	7243	2128	112,74	4,34
2	34,50	29,13	2449	7809	2694	115,89	3,61
3	39,00	36,31	2465	8385	3270	119,01	3,04
4	39,25	38,98	2460	8419	3304	119,19	3,00
5	40,36	45,14	2452	8576	3461	120,03	2,94
6	40,36	54,62	2450	8576	3461	120,03	2,82
7	40,70	63,30	2447	8626	3511	120,29	2,79
8	22,00	2,31	2094	6558	1443	108,80	1,21

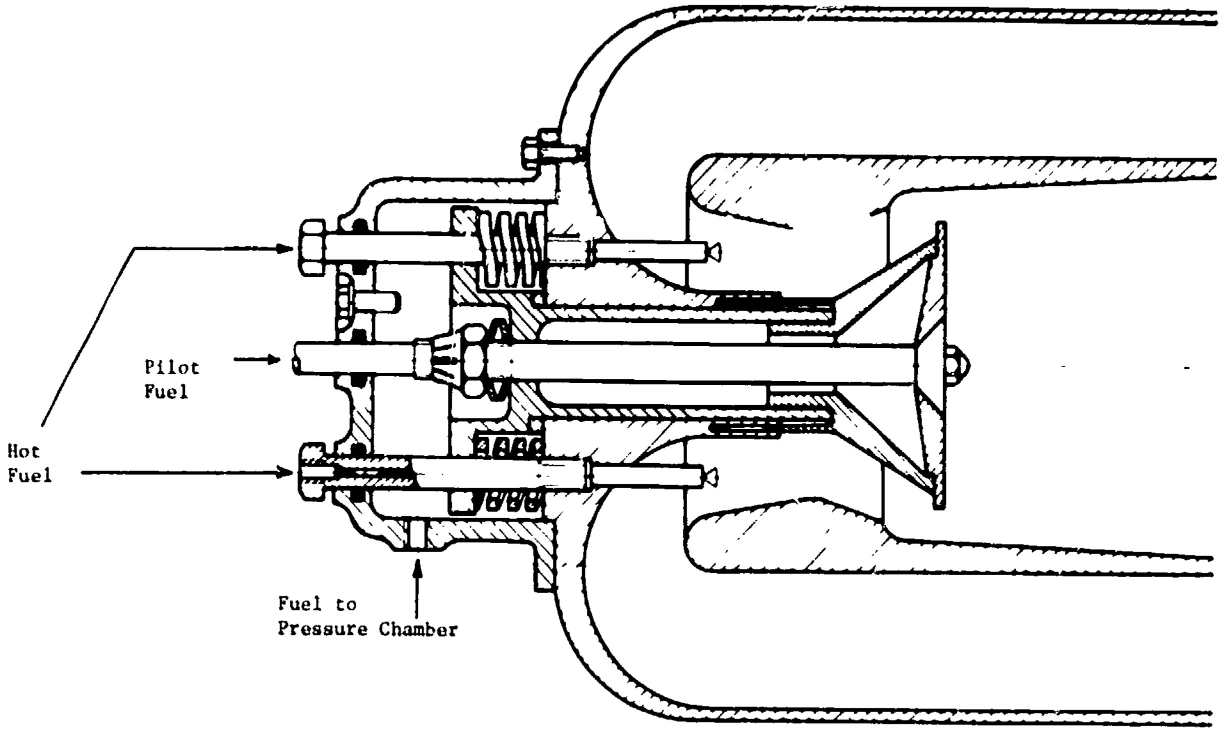

Fig. 2 Combustor Head

Figure 2 indicates the general arrangement of the combustor head, and includes the modifications stemming from the present work. These latter have been necessitated by the requirement for adequate performance and response at the various operating conditions.

Discussions following publication of the earlier work, indicated concern about the functioning of the fuel heater, and also the response characteristics of the baffle to throttle changes. Consideration of such discussions has resulted in the removal of the fuel heater from the outer annulus of the primary zone and the use of cold fuel within the baffle operating mechanism. The justification of the heat exchanger removal will be given in a later section.

The objections to the use of hot fuel within the baffle mechanism are two-fold. At deceleration conditions there is a large quantity of fuel within the mechanism at temperatures and pressures above the new operating condition. During the course of deceleration the fuel pressure is reduced, and there is the grave risk of boiling within the pressure chamber which is situated behind the piston. The implications of such boiling require no further comment. Secondly, a considerable amount of fuel still requires to be exited from the system in order to permit the return of the piston. As originally conceived, this could only exit via the fuel nozzles, and at whatever rate they dictated. This rate might, or might not, be sufficient to cater for the required deceleration. In the new scheme (Fig. 2) the fuel injectors are fed independently of the pressure sensing mechanism, and hence the fuel within the pressure chamber can be removed through a spill device in the control system. Such a device can be of any size, and the baffle movement would then only be limited by the natural frequency of the system. Consideration of the present mechanism suggests that this frequency is of the order of 150 Hz. As a design 'bonus', the removal of the hot fuel from the baffle mechanism also results in a simplification of the sealing problems, and the spring mechanisms.

As in the previous design, the linear movement of the baffle (with a complex wall profile) has been retained in preference to a simple wall and a complex baffle movement. The baffle profile dimensions are listed in Table IV.

TABLE IV

BAFFLE PROFILE DIMENSIONS

Distance from Stop 1	0	1,31	2,90	5,02	5,94	7,78	12,63	17,37	mm
Duct Diameter	108,80	112,74	115,89	119,01	119,19	120,03	120,03	120,29	mm
Gap Width	4,40	6,37	7,95	9,51	9,60	10,02	10,02	10,15	mm

3.2 Intermediate and Dilution Zones

Even though the fuel and air are premixed, it is still considered that an intermediate zone is desirable. This is to provide a 'burn-out' zone within a region which will always operate at a sufficiently weak condition so as to ensure that equilibrium conditions will not give rise to sensible amounts of pollutants, and also to prevent abrupt quenching of any partially completed reactions of the gases emerging from the primary zone. The only unique feature of the proposed system (Fig. 1) is that the intermediate air is introduced through the secondary baffle. The height of the secondary baffle has been chosen as sufficient to stabilise the reciculation behind the primary baffle. However, the theory of design, with respect to baffle position and size, is so sketchy, that optimisation could only be achieved by a short developement programme.

The dilution zone is entirely conventional and requires no further comment. At the design point (Cond. 4, Table I) the air distribution is that given in Table V. For comparison, the estimated air distribution for a conventional combustor has been included (operating at the same condition).

TABLE V

AIR DISTRIBUTION CHARACTERISTICS AT DESIGN POINT

	Premixed Combustor	Conventional Combustor	
Condition No. 4			
Primary Zone Air	39,12	16,19	%
Secondary Zone Air	10,0	6,52	%
Dilution Air	35,89	40,34	%
Film Cooling Air	14,99	36,95	%

One of the noticeable features is the reduction in the required amount of film cooling air. The major reasons for this are (a) the operation of the primary at a weak premixed condition (i.e. lower peak flame temperature) and (b) the fact that premixed flames are non-luminous.

It might be anticipated that this saving would be reflected in an increased availability of dilution air, but this is not so, largely because most of the air thus saved is required to ensure weak mixture operation within the primary zone, and to a lesser extent, the secondary zone. Due to the premixed state of the gases, it is anticipated that the exit temperature distribution will be better than that of a conventional chamber. This may be attributed to the reduction in peak temperatures of the gases leaving the primary zone, and by the further pre-dilution within the secondary zone. Because of these characteristics, the concept of a pre-mixed system should be a good design for combustors operating with a very high turbine inlet temperature, a feature which may become more important in the near future.

3.3 Fuel System Considerations

The changes in the proposed fuel system have been occasioned to facilitate the control of the baffle movement and also to improve the control of the heat transferred to the fuel, whilst at the same time giving better response characteristics to changes in engine conditions.

In the former work [2], it was demonstrated that there was a need to have flash vapourisation of the fuel, since the times required for the complete evapouration of the droplets at inlet conditions were about an order of magnitude greater than the mean residence time between the commencement of injection and the entry to the combustion zone. If the injectors were to be spaced sufficiently far upstream to ensure evapouration, then the dwell times would be of the same order as the estimated spontaneous ignition delays.

An alternative system was devised which preheated the fuel (under pressure) such that immediately upon escape from the nozzle, vapourisation occurred. In order to obtain a reasonable fuel/air distribution in the shortest possible distance, a conical pintle would be inserted into the nozzle to direct the fuel with a cone angle which might range from 60° to 110°. A somewhat similar distributor has been used successfully in other pre-mixed studies [3].

There are possible problems which might occur due to such heating. The major ones are :

a) the possibility of lacquer deposition due to thermally unstable substances within the fuel. Such deposits could bring about blockage of small apertures and/or decrease the efficiency of the heat exchanger due to 'scaling',
b) the possibility of true thermal decomposition (cracking) caused by overheating at the walls,
c) the possibility of vapour-lock,
d) the possibility of corrosion due to the hot fuel.

Of these four factors, that of lacquering is the most difficult, due to the uncertainty of predicting which fuel will lacquer, and at what conditions. The mechanism of lacquering is generally accepted as being the result of some form of oxidation polymerisation of the impurities within the fuel, the oxidation being brought about by diiolved oxygen. Some metals accelerate the process (e.g. copper), others do not (e.g. stainless steel). Of some sigificance is the fact that deposits of this type can be reduced by appropriate treatment at the refinery, although the latter could increase the cost of the fuel by about 20 %. Over the years there has accumulated a fair amount of experimental evidence, but unfortunately much of it is apparently contradictory. Szetela [5] has recently published a good review of the problem, which also demonstrates some of the contradictions. For instance, a substantial reduction in deposits can occur if the fuel is deoxygenated. For the range of wall temperatures anticipated for the present work (320 to 600 K), a potential reduction of the rate of deposition of about two orders of magnitude seems possible. On the other hand, it is also claimed that substantial reductions in deposits can result from aerating the fuel. As would be anticipated, increased fuel velocities (i.e. low residence times) lessen the fuel deposits. Currently, there are insufficient data to be able to predict the rate of deposition for any fuel at any operating condition; one may only attempt to incorporate design features within the fuel heater which will minimise the possible effects.

The lowest temperature at which the fuel commences to crack is about 630 K, which is sufficiently above the scheduled bulk fuel temperature so as to preclude the possibility of cracking other than (possibly) at the walls. Since the air temperature never exceeds 815 K, it seems unlikely that (gas side controlling) the wall temperatures will be excessive. Calculations indicate that the maximum inside wall temperature would be of the order of 600 K.

Vapour lock may be suppressed providing that the fuel pressure is kept above the bubble point pressure corresponding to the fuel temperature.

Corrosion by the fuel can be eliminated by selecting the appropriate stainless steels for construction of the fuel system.

3.3.1 Design Features

The fuel system will be sited in the annular section connecting the compressor discharge and the combustor. For the engine application here, this annulus has the following dimensions - O.D. 340 mm, I.D. 230 mm. To facilitate control of the air passing over the heat exchanger, this annulus will be subdivided into two concentric annuli having the following gap widths - inner annulus 12 mm, outer annulus 43 mm. Air flow to the two annuli would be controlled by an iris-diaphragm type of valve.

The heat exchanger will be fitted inside the outer annulus and will take the form of a stainless steel coil, bearing fins. For this example, the coil will be considered to be a tube (15,9 mm O.D., 12,5 mm I.D.), having circular fins of diameter 41 mm and spaced at 2,5 mm intervals. This heater will be used to preheat the fuel to temperatures corresponding to the dew point temperatures at the operating pressures within the combustor. The calculations indicate that a tube length of 3100 mm should be appropriate (i.e. approx. 3 1/3 coils). The equations used to estimate the bubble and dew points are -

$$\lg P_B = 4{,}0726 - 1584{,}25 \div T \qquad (1)$$

$$\lg P_D = 6{,}2008 - 2914{,}41 \div T \qquad (2)$$

These equations have been derived by consideration of the critical data and the normal bubble and dew points reported in Table II. The estimated performance of the fuel heater is listed in Table VI.

TABLE VI

PERFORMANCE OF THE FUEL HEAT EXCHANGER

Cond. No.	Air Press. $P_2 = P_D$	Dew Point Temp. at P_2 ($= T_f$)	Bubble Pt. Pressure at T_f (P_B)	Actual Fuel Pressure P_f	Air Temperature Preheater		Total amf	% of amf to Heat Exchanger
					Inlet	Outlet		
	bar	K	bar	bar	K	K	gmol/s	
1	6,88	542	14,11	10,62	735	689	75,05	20,6
2	7,88	549	15,38	18,62	779	724	84,49	21,1
3	9,00	555	16,52	28,96	815	755	93,12	22,3
4	9,60	558	17,12	33,44	810	758	99,32	26,1
5	10,68	563	18,14	42,33	803	764	111,85	36,4
6	12,99	570	20,31	65,50	789	767	134,55	65,7
7	14,99	580	21,93	87,97	787	772	155,44	100

If the results given above are reviewed in the light of the previous discussion, one may deduce -

a) the final fuel temperatures have values corresponding to the region where substantial lacquering may be experienced. However, the tube diameter is comparitively large, and blockage may not be a problem. Loss of heat transfer efficiency may necessitate a somewhat larger heater than that given above. Since there is no means of predicting the behaviour of any given fuel, it is anticipated that some separate developement work will be required in this area. Previous test rig experience with a similar heater gave no indication of problems, although he operating time was only a total of about fifty hours.

b) Fuel cracking should be insignificant, since the highest predicted wall temperature was estimated at 590 K.

c) There is a possibility of vapour lock at Condition 1, but this could be overcome by a small reduction of flow number of the main fuel atomisers. However, the uncertainty of the bubble and dew point data suggests that such a change should not be introduced prior to the preliminary testing of the unit.

It may thus be concluded that a heat exchanger of the above type is feasible from every aspect, except possibly, that of thermal instability of the fuel. Even with respect to this latter, the chances of successful developement must be rated as fairly high.

As before, the light-up and idling condition would be met by means of a Simplex pilot atomiser. The advantages and disadvantages of such a system have been discussed elsewhere [2]. The use of such a pilot minimises the possibility of flashback over the baffle to the main fuel injectors. The experience of Roy et al [3] confirms the suitability of the use of a pilot.

4. COMBUSTOR PERFORMANCE

The methods used to predict the combustor performance were discussed in the former paper [2], and need not be repeated here. However, a further review of the available data [3,6] has shed a little more light on the performance of premixed systems. The chamber used for the reported work, was a 90° sector of a small annular chamber (radii 163 and 101 mm) fabricated with ceramic walls, so as to minimise heat losses and prevent quenching at the walls. Gas samples may be taken at several planes spaced from the combustor exit (corresponding to the normal primary zone exit) to a plane immediately downstream of the recirculation zone associated with the primary baffle. This chamber operates with a nominal inlet Mach No. of 0,04 and with an overall pressure loss of 4 %.

The combustion efficiency data from [3] (combustion in air, unadultered by steam) may be represented by an equation of the Odgers-Carrier type 7, and for weak mixtures this is -

$$\lg \lg 1/\eta = 1{,}699 \lg(\dot{m}_f / V' P^{2\phi}) - 1{,}191\ \phi + 0{,}9022 \qquad (3)$$

Units : lb/s, ft^3, atm

Using Eqn. (3) to predict the end efficiencies for the present combustor yields for all conditions (except idling No. 8, not applicable) an end efficiency of 99,9+ %. On the basis of this, it would seem that the combustor would exhibit no pollutant problems due to incomplete combustion products (CO & UHC).

Recently, measurements have been made of NOx, using the premixed sector of Roy et al. [6]. The new data are plotted against the simple correlation curve proposed by Kretschmer et al. [8]. The data show sensible agreement with the scatter band (Fig. 3) and lend a measure of confidence to the predictions made in [2].

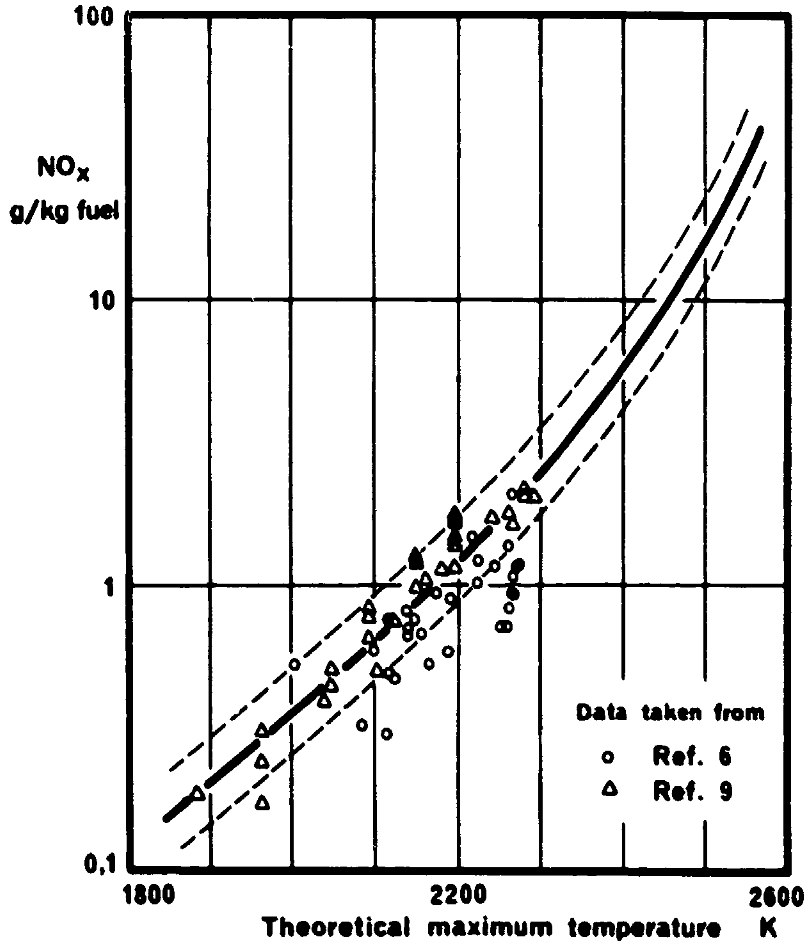

Fig. 3 NOx Found in Gas Turbine Exhaust (as NO_2)

TABLE VII

NOx PREDICTIONS FROM [2]

Cond. No.	NOx Premix	NOx Conventional
1	3,0	20
2	4,2	30
3	4,9	47
4	4,8	45
5	4,7	42
6	4,5	40
7	4,5	40
8	2,1	2,1

in g/kg fuel

4.1 Performance at Idling

There has been no change in the concept of the design for the idling condition, and the pilot system will be used. Since there is no easy means of preheating and flash-vapourising this fuel, the combustor performance at this condition will be similar to that of a conventional chamber. However, the injector has only to cater for a single operating condition, and thus it should be possible to develope the system to yield an acceptable combustion efficiency. This was indicated in [2].

5. WALL COOLING

None of the modifications proposed in the present chamber will alter the air distribution from that given previously [2]. Hence the wall temperatures will remain as they were (Table VIII).

TABLE VIII

PREDICTED WALL TEMPERATURES

Condition No.		1	2	3	4	5	6	7	8	
Maximum Wall Temperatures	Primary Zone	1179	1225	1266	1258	1256	1250	1263	1116	K
	Secondary Zone	1025	1110	1195	1202	1198	1203	1202	630	K
	Dilution Zone	1033	1092	1173	1176	1198	1188	1193	634	K

6. TRAVERSE QUALITY AT THE COMBUSTOR OUTLET

The dilution system used is quite conventional and one might, therefore, expect a similar temperature distribution to a none-premixed chamber operated at the same conditions. Due to the premixed entry conditions to the primary zone, and to the lean mixture operation, the gases leaving a premixed primary zone will be much more homogeneous than those within a standard chamber. For this reason, the duty of the secondary and dilution mixing devices are not so difficult, and one might confidently expect a better traverse for the premixed chamber. In the conventional chamber, a big proportion of the air is used for the purpose of film cooling and this also detracts from the mixing quality. The premixed chamber only requires about half the air for film cooling, and should consequently be better mixed.

7. CONCLUSIONS

The present results confirm the previously reported study [2] as to the viability of a premixed combustor with variable area control of air distribution.

The predicted performance, based upon experimental data from a premixed (but not variable area) combustor, suggests that all pollutants may be controlled to meet environmental restrictions, except possibly at idling, where the combustor functions in a conventional manner.

By using a moveable baffle, actuated by the fuel pressure, the air distribution must follow the engine response to changes of operating conditions. The inclusion of stops limits the baffle movement, and provide a fail-safe action by ensuring combustion at all operating conditions.

The need to vapourise the fuel immediately upon injection (to ensure adequate mixing) necessitates the use of a flash vapourising system.

In order to ensure adequate response of the baffle movement to changes of engine conditions, it is desirable that the fuel which operates the piston, (which governs the baffle movement), be both cold and independent of the injectors. This has resulted in the fuel heating system being placed in the annular section between the compressor discharge and the combustor. To facilitate control of fuel temperature, this annulus will be needed to be sub-divided, and the air flow into each annulus will be metered by means af a single iris-diaphragm valve.

The fuel heater can be designed to eliminate cracking, vapour lock and fuel corrosion, but there is no guarantee that thermal instability will not result in lacquering. An attempt has been made to minimise this latter by (a) using a wide bore tube and (b) inserting a filter after the heat exchanger. However, it is anticipated that some developement might be needed to meet this problem.

ACKNOWLEGEMENTS

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NOMENCLATURE

A/F	Air/fuel ratio by mass		-
P	Pressure	(Eqn. 3: atm)	bar
T	Temperature		K
V'	Combustion volume	(Eqn. 3: ft^3)	n.a.
ϕ	Equivalence ratio		-
η	Combustion efficiency		-
$\dot{m}$	Mass flow	(Eqn. 3: lb/s)	gmol/s

Indices:

a	Air
B	Bubble point
D	Dew point
f	Fuel
0	Overall
2	Inlet

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DISCUSSION

H.Ahrendt

Do you think that in further development you can reduce pressure losses to those of conventional combustors?

Author's Reply

The value of 3% pressure loss is that of a conventional combustor, although it rises to 4% at one operating condition. This is because with variable area baffle, the pressure loss must change with conditions due to the change in restrictions. Many conventional combustors still operate with 5% loss, so I don't think we are excessive in accepting 4% at a single condition.

G.Winterfeld

In the conclusion in your paper it is stated that the fuel for operation of the baffle will be cold. Could you please specify what temperature range has to be used?

Author's Reply

Approximately 300 K was assumed to be the delivery from the fuel pump though it would pick up a small amount of heat from the inlet air steam.

G.Winterfeld

Then, if the figure is 300 K, how would you sustain this figure in operational environment? The engine heats up if it runs for several hours and the temperature of the fuel may well come above 300 K.

Author's Reply

I think that it still would be acceptable with any normal engine pump or any wing temperature rise. What I am implying is that it cannot be permitted to approach the flash vaporisation temperature.

D.R.Hounslow

Do you feel that at the relatively low design air casing Mach number you have chosen $M = 0.03$ I believe, that the simple baffle/secondary baffle configuration is the optimum flame stabilisation device to chose?

Author's Reply

I agree that a Mach No. of 0.03 is not the best operating number necessarily. This was selected because this gave us approximately the 3% pressure loss we wanted. Personally, I would prefer to see a somewhat higher Mach number (about 0.05), but this would have to pay the penalty of an increased pressure loss. It is a matter of weighing the balance between pressure loss and what you would like in terms of combustor velocity. In fact, generally speaking a somewhat higher Mach number would be desirable, at least from the combustion viewpoint.

R.M.Denning

Have you considered that the pressure drop in the combustion chamber may be defined by the requirement to cool the HP turbine stator and that the figure of 3 ~ 4% loss may be as high as 5 ~ 5½%?

Author's Reply

Obviously, if such an event happened, one would require a design modification but in fact (as I have indicated in the previous discussion) I would be prepared to accept the higher pressure loss because it has a higher mixing potential etc, and it would be beneficial with regard the combustion processes.

THE VARIABLE GEOMETRY COMBUSTOR

by

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R. C. Adkins - Senior Research Officer

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SUMMARY

An attempt is made to review both the concept of variable geometry as it applies to the control of emissions from gas turbine combustors, and also to the potential of a novel fluidic device to put the concept into practice. This method is based upon the use of a bled-vortex generated to control and vary the air distribution characteristics within the combustor liner, and experimental data are given from the first 'cold' tests carried out with the device.

The conclusions made in the report are that

- although significant emissions reductions can be achieved by the use of variable geometry, it is unlikely that the level of reduction attained will be sufficient to exceed the appropriate US Federal Standards for civil aircraft engines.

- the vortex-controlled, variable geometry combustor has potential for further development.

1. INTRODUCTION

The pollutant emissions levels from aircraft gas turbine engines must be significantly reduced in the future if they are to satisfy the current United States Federal Standards[1]. The 1981 Standards require that the oxides of nitrogen be reduced to approximately one-third, and the carbon monoxide to below one-twentieth of the levels which are exhausted from today's engines. Efforts to produce such low-emission combustors have taken many different paths in recent years and the subject of this report is that path which has become referred to as variable geometry. The term 'variable geometry' implies that the air distribution characteristics within the gas turbine combustor liner can be made to vary with changes in engine power level, and its name has been derived due to the fact that these changes were first effected by making modifications to the liner geometry by, for example, mechanical variation of the size of the liner holes.

This report attempts to present the basic reasons for the effectiveness of variable geometry, to describe the limitations of the method and also to present the first 'cold' results obtained from an experimental programme aimed at producing a system which employs a special fluidic device to control and vary the air distribution characteristics in the liner.

2. POLLUTANT EMISSIONS AND THE VARIABLE GEOMETRY COMBUSTOR

2.1 Formation of Pollutant Species

The sources of the pollutants of interest in an aircraft gas-turbine combustor are reasonably well understood, and are directly related to the temperature-time-composition histories of all the fluid elements in the combustor. Carbon monoxide, CO, is an incomplete product of combustion, formed in substantial quantities in the primary zone. It undergoes oxidation to CO_2 in the primary and intermediate zones, and this rate-limited step greatly influences the concentration of CO in the exhaust gases. CO emissions are at their maximum at low-power conditions when the combustor is operating close to the weak-extinction limit, the bulk temperatures are low, and the oxidation rates are slow. The presence of hydrocarbons (HC) in the exhaust is due largely to unburned fuel and partially oxidised products which find their way into the cooling films within the combustor liner, and like CO, these emissions are also highest at low-power conditions.

Engine smoke, mostly carbon, is formed in the primary zone in fluid elements with rich fuel-air ratios, and consequently are at a maximum at high-power conditions. The oxides of nitrogen, which comprise mostly nitric oxide (NO), are formed in the high temperature regions of the primary and intermediate zones, and in fluid elements with equivalence ratios near unity; their formation rate is characterized by a very strong dependence upon temperature. Due to the relatively slow reconversion process of NO to N_2 and O_2, once formed, it remains in the exhaust gases and is most prevalent at high-power conditions where the bulk temperatures are maximum.

Accurate predictions of the emission levels of all the species considered above cannot be made, but an indication of the dependance of the species of primary concern, namely NO and CO, upon the combustor operating conditions can be obtained from the data in Tables 1 and 2 below.

TABLE 1

NO FORMATION CHARACTERISTICS

AT FULL-POWER CONDITION

Equivalence Ratio ϕ	Flame Temp. (Typical) ^oK	Concentration in g/kg Fuel NO Equil.	Formed
0.4	1610	90	0.0
0.6	1989	180	0.1
0.8	2310	200	16
1.0	2500	110	40
1.2	2470	20	6

TABLE 2

CO FORMATION CHARACTERISTICS

AT IDLE CONDITION

Equivalence Ratio ϕ	Flame Temp. (Typical) ^oK	Concentration in g/kg Fuel CO Equil.	Formed
0.4	1380	0	2000
0.6	1750	2	1726
0.8	2080	21	29
1.0	2300	197	197
1.2	2250	784	942

In both tables, a comparison is made between the concentration levels that are formed at the exit of the primary zone of the combustor with those that would exist if equilibrium conditions were to be achieved. The data in Table 1, calculated by the method given in Reference 2, clearly indicates two important factors in NO formation; firstly that the concentration levels formed are well below equilibrium conditions, and secondly that there are two orders of magnitude increase in NO formed as the fuel-air equivalence ratio changes from 0.6 to 0.8 (or as the temperature changes from 2000 to 2300^oK). The CO formation data in Table 2 was calculated on the assumption that the fuel is instantly oxidised to CO and that the CO is further oxidised by reaction with the OH specie. The results indicate that low CO levels can only be achieved in a narrow range of equivalence ratios from approximately 0.7 to 0.9 (i.e. for temperatures from 1900 to 2100^oK). It is important to note that at ratios below 0.7 the CO levels are high due to the slow rate of oxidation whereas at ratios above 0.7 they are high because they reach equilibrium conditions. These CO equilibrium values take no account of the oxidation reactions that occur beyond the primary zone of the combustor so they must be used with care. Experimental evidence shows for example, that CO levels from combustors operating at high-power levels where the mass-mean primary zone equivalence ratio is near unity, are usually less than 10g/kg fuel rather than at the 200 equilibrium value which corresponds to the primary zone condition. Hence the oxidation reactions must play a very significant role at this condition. Perhaps the most important conclusion that can be made from these approximate calculations are that emission levels of both NO_x and CO are extremely sensitive to the temperature histories within the combustor. A low emission combustor can only be developed, therefore, if the design permits close control of the temperature histories within the combustor at all engine operating conditions.

2.2 Variable Geometry Combustion

The practical difficulties in developing the low-emission combustor are substantial, both in achieving the homogeneous, or near-homogeneous conditions implied by the results given above, and also in maintaining a limit to the equivalence ratio(s) within the combustor over the wide range of operating conditions through which the aircraft gas turbine must operate. The homogeneous state can only be approached if complete pre-mixing of the fuel with primary zone air takes place before combustion, and the full potential of the variable geometry concept is only realised if this is carried out.

Present-day combustors, however, operate with fuel injection directly into the combustor which results in combustion taking place in numerous fluid elements over a wide range of equivalence ratios. A combustor, or combustion region, can be characterised by the mass-mean equivalence ratio, $\bar{\phi}$, (or $\bar{\phi}_p$ for the primary zone) but it is clear from the discussion above that the distribution of equivalence ratios about this mean are as important as the mean itself in influencing emission levels. As a result of this factor, the rate of change of NO and CO with $\bar{\phi}_p$ for such systems cannot be expected to be nearly so rapid as the data in the two tables would imply, and in addition neither specie can be expected to go to the zero concentration level as there will always tend to be some hot regions to generate NO, and cold regions to produce CO. A basic feature of gas turbine combustors is that if they are operated with conventional fuel systems, they then exhibit a characteristic CO-NO_x relationship similar to that shown in Figure 1 (see reference 3), and even if variable geometry is employed with these systems then relatively little departure from this characteristic curve can be anticipated. Consequently the potential of this approach to pollution control relates to the position of this characteristic relative to the emission standards required. For aircraft engines the present standards correspond to emission indices of approximately 8g/kg fuel for both pollutant species. These criteria tend to fall close to the characteristic of most aircraft gas turbines, as is shown in Figure 1, which would indicate that they may be achieved by the use of variable geometry providing very close control is maintained on the combustion zone conditions. In practice this is not feasible because although the high and medium power levels may be operated near to the 8:8 point, it is not possible to reach this point during the idle condition due to the limit in flame temperatures imposed by the low combustor inlet temperature. If the limitations imposed by the CO-NO_x characteristic are accepted, therefore, it must be considered necessary to incorporate an additional control technique such as fuel-air premixing, if the emission levels are to be comfortably below the 1981 Aircraft Standards.

Despite these limitations, it is clear, however, that substantial reductions in NO_x and CO emissions can be achieved by the application of the variable geometry concept. Its potential has been fully confirmed experimentally by many workers[4] in the last few years but more ingenuity and development is required in the design and control systems necessary for the concept before it can find wide application in practical gas turbine engines.

3. THE APPLICATION OF VORTEX FLOW CONTROL

Many techniques have been proposed and developed for varying and controlling the air flow distribution characteristics in combustors but nearly all have involved the use of moving parts either adjacent to the hot walls of the combustor, or in the air inlet stream.

One novel approach that has recently been studied at Cranfield involves the use of the vortex flow control principle which was pioneered by Adkins[5]. A simple piece of hardware has been built involving this principle with the objective of producing a system with a range of primary zone air flows from 28 to 38 percent. This range was selected on the basis that it represented the minimum range to produce the lowest emissions from a specific combustor with known CO-NO_x characteristics*. The study was the subject of the student dissertation project[6] and will be described together with a brief explanation of the principle of vortex flow control.

3.1 Vortex Flow Control

Over the past few years the principle of vortex flow control has been applied to a range of short diffusers (with an included angle of 30°) each with a differing area ratio and geometric configuration. In all cases, the problems of substantial pressure loss and flow instability that normally results due to such short lengths have been avoided by the presence of the vortex, and the measured static pressure recoveries have been higher than those which may be obtained using conventional aerodynamic designs. The application of the principle does, however, require a small fraction of the mainstream air to be bled from the vortex chamber to maintain stability and further research is in progress to

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* The data given in Section 2.1 indicates that homogeneous combustion systems require a wider range of air distribution characteristics. To achieve a fixed ϕ of 0.7, for example, would require a range between 20 percent PZ air at idle to 40 percent PZ air at full power according to the schedule given in Figure 2.

minimise the level of air bleeding required. The results obtained to date clearly show the bleed levels to be well below that required by the more conventional diffusers which operate with boundary layer suction to reduce diffuser length.

The simplicity of construction required to produce a tubular vortex controlled diffuser is demonstrated by the design shown in Figure 3. The amount of bleed required from this method of approach has been found to be very sensitive to the position and size of the vortex fence located just downstream of the diffuser throat. A full description of the flow structure postulated to occur at the vortex is given in Reference 5, where the case is also made that the structure is inherently stable. A typical performance characteristic of a VFC diffuser is shown in Figure 4, which relates the static pressure effectiveness η (defined in the figure), to the quantity of bleed air taken from the vortex chamber. The characteristic is almost step-shape, with the big improvement in performance occuring from points (b) to (c) in the figure, and the quantity of bleed at the point (c) has been defined as the "minimum bleed requirement, B_C". Correlations have been obtained from these research studies which permit values of B_C and η to be estimated for any new applications. These correlations indicate that both quantities are dependant primarily upon diffuser configuration and the flow distortion characteristics at the inlet, and that the application of the technique in a typical gas turbine engine will require less than three (3) percent bleed in order to achieve an effectiveness of approximately 85 percent.

It is the ability of this small quantity of bleed air to effect a significant variation in the flow behaviour in the diffuser which may possibly be used, with careful design, to control the air distribution characteristics in a variable geometry combustor.

3.2 Design Details of Test Equipment

The vortex flow control device can be used in a number of different ways to produce the desired flow control. The configuration used for this study, shown in Figures 5 and 6, was chosen because the required variation in primary zone flow was modest, namely from 28% at Idle to 38% at Full-power, and also to ensure that the rate of bleed-off would be a maximum at the Idle condition where its effect on engine performance is less critical. Since an appreciable amount of the primary zone air is obtained by re-circulation from the secondary holes, it was decided to separate them from the dilution air by means of a shield. A vortex flow device had therefore to control the distribution of flow between the shielded and dilution-air passageways. The purpose of the model was to determine the vortex fence position and size, and the position of the primary zone splitter shield to achieve the required range in flow distribution.

The model, fabricated out of 1.6mm sheet metal, was tested at near ambient pressure. The air was supplied from a small fan capable of developing a diffuser inlet Mach number of 0.22. The combustor was operated at pressures in excess of atmospheric so that the bleed gases needed no suction, and the bleed flow was controlled by a valve. Orifice plates, installed according to B.S.1042, were used to measure the flow rates at the model inlet and at the bleed pipe exit. A 10-point rake, designed to give equal area weightings across the diameter, was used in conjunction with wall static tappings to measure the mass flow entering the shielded primary zone area. The testing procedure consisted of a systematic variation of the variables, X, Y, Z (see Figure 7), and each configuration was subjected to a range of bleed rates from zero up to the maximum attainable. The ranges of variables investigated were as follows:-

Fence axial gap,	X from 2.74 to 6.2mm
Fence radial gap,	Y from 2.03 to 3.81mm
Distance from throat to shield,	Z from 15.24 to 43.2mm

The bleed flow rates were limited to a maximum of approximately 3.5 percent in most cases due to the pressure limitations of the fan used to supply the test air.

3.3 Discussion of Experimental Results

The results obtained from a systematic variation of the three main design variables, X and Y for the control fence, and Z for the position of the primary zone splitter relative to the diffuser throat, are shown in Figures 8, 9 and 10 respectively. All show the change in PZ, the percentage of air that enters the primary zone, versus the percentage of air bled from the diffuser exit. The general shape of these graphs can be anticipated from the discussion given above, and in particular from the typical characteristic of vortex controlled diffusers given in Figure 4. It is to be expected that at low bleed values, a to b in the figure, there will be little change in the airflow distribution until at the critical point b, a decrease in the primary zone airflow will occur and continue to occur until the critical point C is reached. After this point, further increases in bleed flow rate will only have a marginal effect upon the airflow distribution.

The important criteria for the application under study are the rate of change of PZ with bleed as it must be sufficiently sensitive to permit ease of control, and the range in airflow distribution that can be achieved with any one geometry. Unfortunately, the lower limit of the range was not determined in all cases due to the economies imposed in operation of the equipment, as it was not possible to apply suction to the bleed tube. The lower limit was controlled, in most cases, therefore, by the level of the pressure drop between the bleed chamber and the atmosphere.

Figure 8 shows the influence that the position of the fence has upon the operation of the combustor. The value of PZ is shown to decrease from 36 to 30 as the bleed increases from zero to 3.5 percent, but it is clear that a wider range could be obtained with additional bleed. The optimum position of X, in terms of the sensitivity of PZ with bleed, appears to be within the range of 3.66 to 4.37mm but, in general, the influence of fence position is not shown to be of great significance.

The other critical dimension for the fence is the value of the radial gap, Y, between the diffuser throat and the top of the fence. This parameter is shown in Figure 9 to exert a significantly greater effect upon the performance characteristics. The best performance in terms of both range and sensitivity is that obtained at a value of Y = 2.5mm.

Finally the results in Figure 10 show the influence that the position of the primary zone splitter shield has upon the performance when the fence position and size are at the optimum values of X = 3.66 and Y = 2.54mm respectively. As the distance from the diffuser throat reduces from 43.2mm to 15.2mm the characteristics change very little; the maximum value of PZ increases from 35 to 36 and its minimum value from 29 to 30. Clearly, this range could be increased by reducing the distance below 15.2mm, and by increasing the bleed flow rate either by applying suction or by operating the combustor at higher pressure levels.

In general, these results are considered to be encouraging and the concept worthy of further pursuit particularly as it involves no moving parts and, with development, it is anticipated that the range in PZ values can be extended significantly. It should also be stressed that the concept in its present form can operate without bleed at the high power level conditions, and thus will have a very limited effect upon the overall engine performance.

4. CONCLUSIONS

The following main conclusions can be made:-

1. Significant reductions in the levels of all emissions species can be achieved by the use of variable geometry.

2. The absolute level to which emissions can be reduced from gas turbines by the use of variable geometry alone is probably not sufficient to meet the 1981 U.S. Federal Aircraft Gas Turbine Standards.

3. The application of the principle of vortex flow control to achieve changes in air distribution characteristics with engine operating conditions, appear feasible and worthy of further pursuit.

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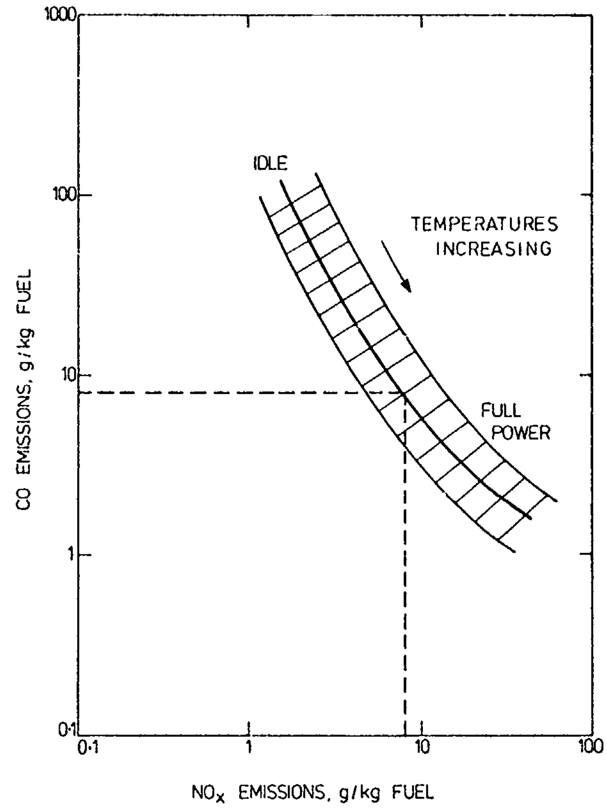


FIGURE 1 TYPICAL EMISSION CHARACTERISTICS OF A CONVENTIONAL GAS TURBINE COMBUSTOR

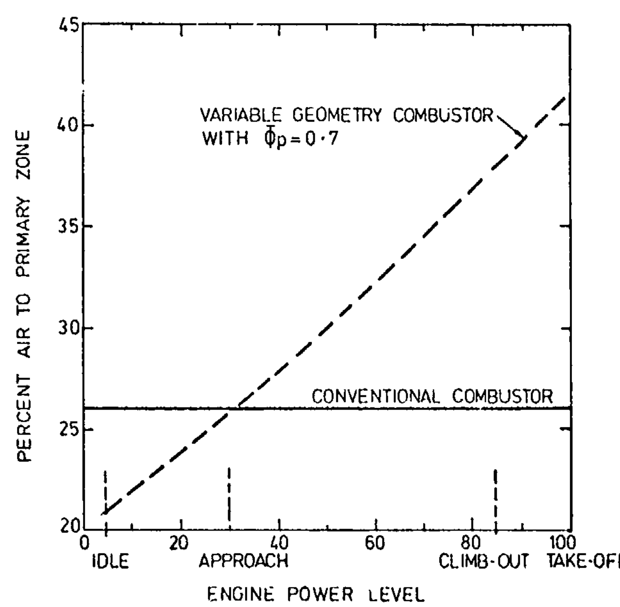


FIGURE 2. VARIATION IN PRIMARY ZONE AIR FRACTION WITH ENGINE POWER LEVEL

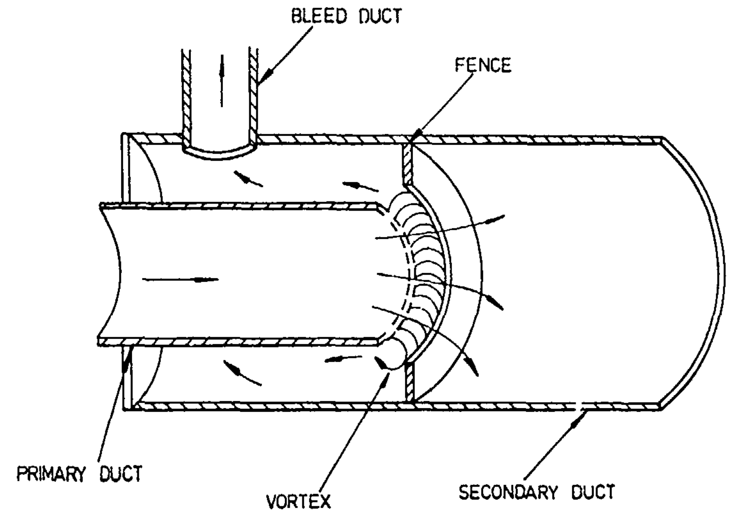
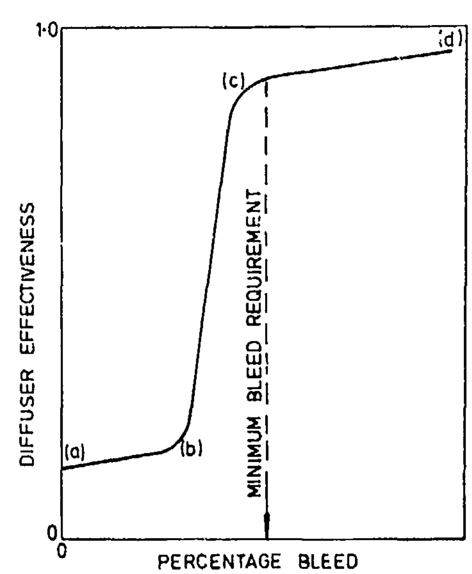


FIGURE 3. A TYPICAL TUBULAR VORTEX FLOW CONTROLLED DIFFUSER



EFFECTIVENESS IS THE MEASURED STATIC PRESSURE .ISE DIVIDED BY THE IDEAL STATIC PRESSURE RISE

FIGURE 4. TYPICAL DIFFUSER OPERATING CHARACTERISTICS

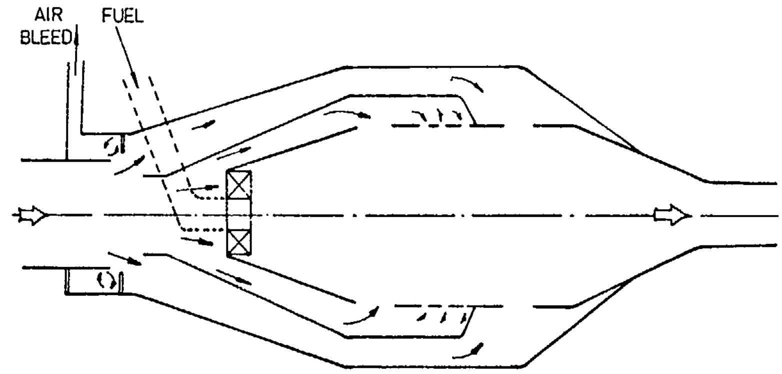
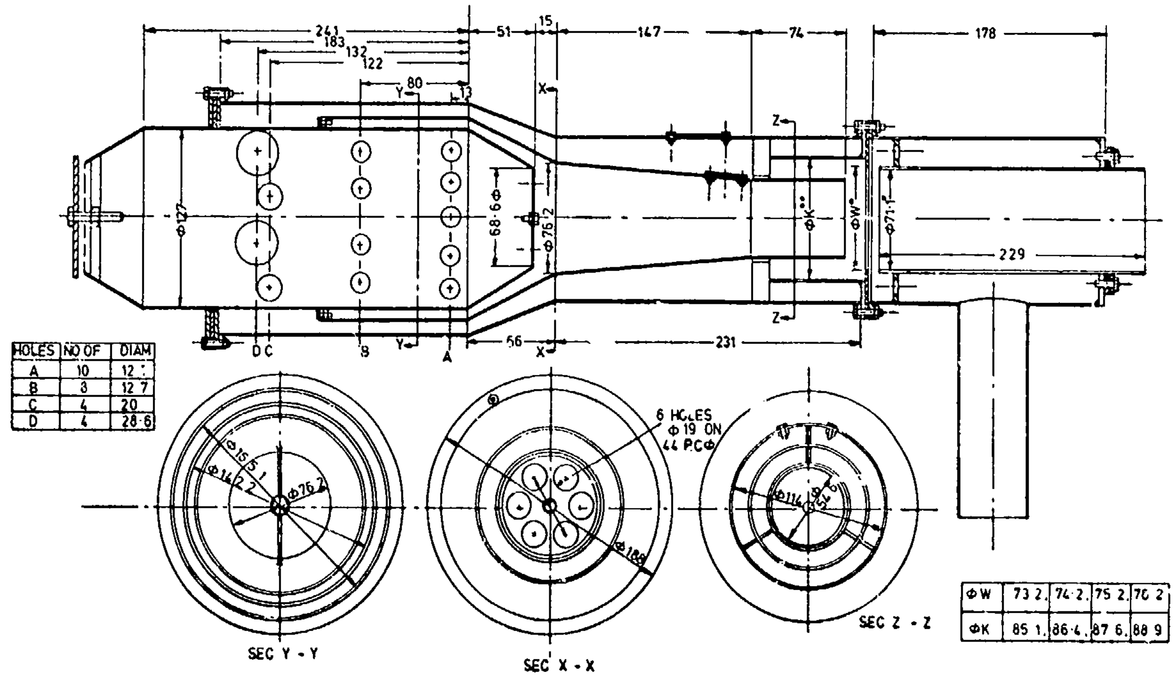


FIGURE 5. VORTEX FLOW CONTROLLED 'VARIABLE GEOMETRY' COMBUSTOR



HOLES	NO OF	DIAM
A	10	12 :
B	3	12 7
C	4	20
D	4	28·6

ΦW	73 2.	74·2.	75 2.	76 2
ΦK	85 1.	86·4.	87 6.	88 9

FIGURE 6. GENERAL ARRANGEMENT OF TEST COMBUSTOR [Dimensions in mm]

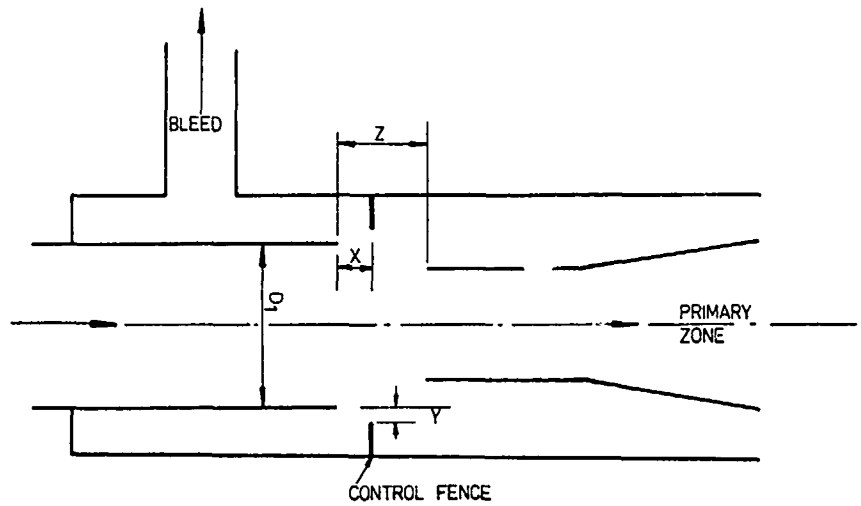


FIGURE 7 DEFINITION OF DESIGN VARIABLES

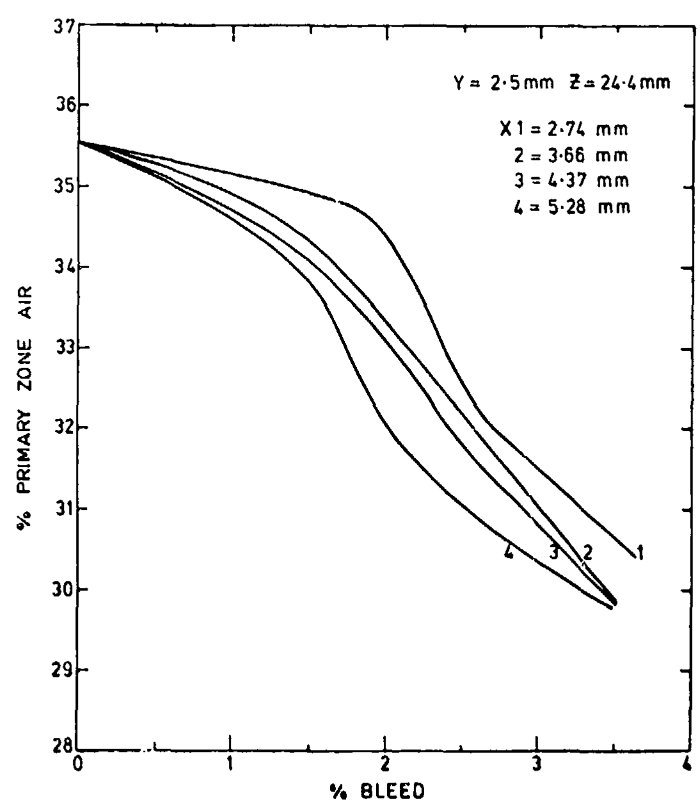


FIGURE 8 PERFORMANCE CHARACTERISTICS WITH CONSTANT Y AND Z

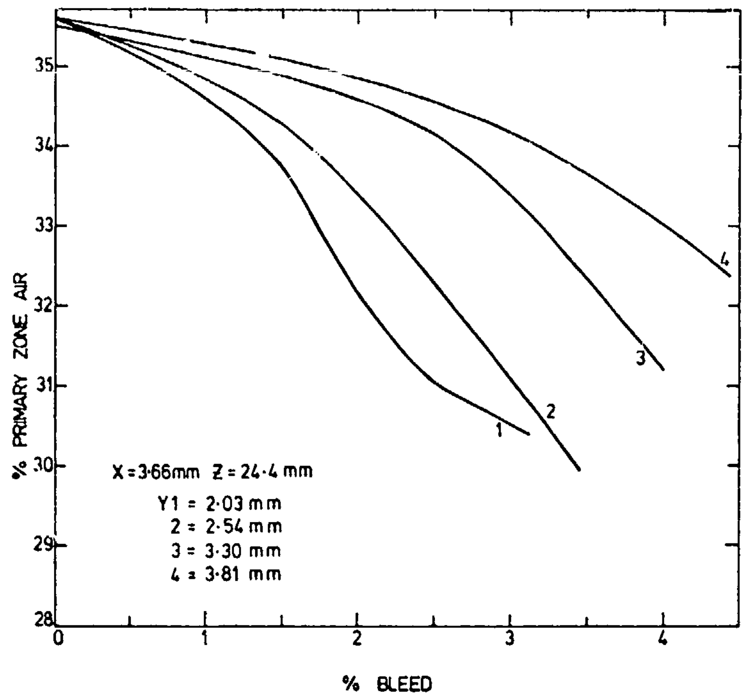


FIGURE 9. PERFORMANCE CHARACTERISTICS WITH CONSTANT X AND Z

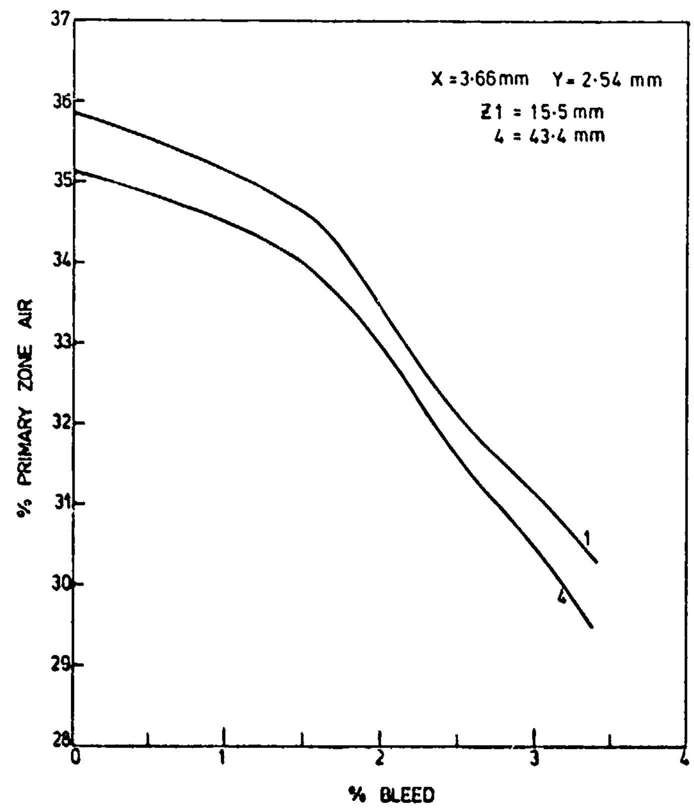


FIGURE 10. PERFORMANCE CHARACTERISTICS WITH CONSTANT X AND Y

G.Winterfeld

The concept of varying the combustion air distribution by a bleed-air stabilized diffuser seems to be very encouraging. However, bleed is a penalty with respect to s.f.c. Can you give some indication about the increase of s.f.c. to be expected?

Author's Reply

We have done sums on this matter although not for the variable-geometry combustor considered in this paper, but for the case in which the bleed vortex is used solely to control the flow in precombustor diffusers. The results show the concept to be advantageous if:

(1) we use the 3% bleed air for turbine cooling, or

(2) we can reduce the rate of bleed to lower levels.

I know I am not answering your question in a satisfactory manner, but I have no numbers at hand. I am just giving you an indication of the conclusions that we have come to. You will note that in the concept in this paper the 3% bleed is required only at idle and not at full power, so the s.f.c. penalty is really very modest indeed.

M.Pianko

On figure 2, the objective shown is to vary the percentage of the primary air from 20 to 40%. However, on the last figures 8, 9 and 10, it appears that by modifying the parameters x, y, z, you obtain a variation of the primary air only from 25 to 35%. Is it possible with this concept of variable combustor to achieve the goal of variation from 20 to 40%?

Author's Reply

At the moment we do not know but may I make three quick comments:

Figure 2 is meant to represent the extreme limit to the design goal and is beyond the goal we would need to attain the current emission standards.

We really do not know the limit to which this concept can be pursued. There are many variables, not only the x, y, z that we showed, but also in the basic geometry that we have employed.

Finally, as I understand it, emission standards that are currently in force for 1981 are going to be substantially relaxed and, in particular, for the oxides of nitrogen. The problem then reduces to one of CO control at idle and I am confident that the range required for such control could be achieved by this sort of approach. Of course the difficulty today is that in designing for a low emission aircraft gas turbine combustor we are shooting at a moving target. Until the standards are firmly set it is very difficult to say which is the optimum method of approach to reach the standards.

D.Kretschmer

Have you ever experienced aerodynamic instabilities?

Author's Reply (R.S.Fletcher)

The answer is yes if we do not design it correctly, but perhaps I can refer you to Mr Adkins on this one as he has a history of some eight years in duration which he can use to reply to that question.

Author's Reply (R.C.Adkins)

We have found that there is a limiting area ratio for stable diffusion which is set by the degree of non uniformity of the flow at the diffuser inlet. We are able to predict this limit, and by selecting an area ratio of just less than this value, we can ensure that our diffuser will run within the stable flow regime.

Author's Reply (R.S.Fletcher)

Or alternatively, we may say, as the inlet flow distortion increases, the maximum area ratio of the diffuser which can be achieved with this concept is reduced. This is particularly important, of course, as the velocity profiles just upstream of the combustor are never flat as the combustor designer always assumes in the design process.

J.Odgers

What precautions have you taken to ensure a fail-safe mechanism?

Author's Reply

Well it is on the grounds of safety, I think, that the proposed approach has much appeal because it is on this criteria that moving valves or other mechanical controls give cause for concern. Our design approach is such that it would only require significant bleed at idle. In an emergency case, if the bleed valve failed to open, excess air would be fed into the primary zone and blow-out may occur. At the opposite end of the spectrum, if the bleed valve leaked at full power there would be less air in the primary zone than we designed for. Having chosen a burner of say 0.70 equivalence ratio, the worse that could then happen is for the combustor to operate at a condition which exists in today's combustors. This is, of course, simplifying the issue somewhat as there are other problems such as wall cooling. But taking all factors into account, we think the approach has potential for aircraft gas turbines.

THE PROS AND CONS OF VARIABLE GEOMETRY TURBINES

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Definition of variable geometry

The concept of "variable geometry" (v.g.) may have an extensive as well as a restrictive signification. We must therefore first suitably define this concept. "An (airborne) gasturbine engine is said to have variable geometry if the off design behaviour can be influenced to a degree exceeding the limit set by the RPMs of the controlled rotor." This lecture will cover the jet engine, i.e. the airborne gasturbine engine.

Twospool and multispool units without mechanical coupling of the rotors, which are nearly exclusively used today, belong then to the type of engine with variable geometry even though they possess no regulatory devices. Due to the fact that the rotors are free rotating, which leads to a variable speed ratio ($n_{LP}/n_{HP} \neq$ const) through the whole range of operation, an "elasticity" in the mode of operation (partial load, acceleration etc.) is achieved which can normally only be attained through variable geometry (for example adjustable compressor guide vanes). The comparison of the two turbojet engines RR-Olympus and GE J-79 can be cited as an example. For the engines the total pressure ratio is about 15 respectively 12. The Olympus engine achieves the compressor adaption at partial load through free rotating of both rotors. With the single rotor concept of the J-79 (two mechanically coupled rotors would here be equivalent) this was achieved through the arrangement of 7 adjustable stators.

The above definition emphasizes the off design point operation of the propulsion system which should be improved through the use of variable geometry. The improvement could pertain to the efficiency of a component and thereby reduce the specific fuel consumption at a definite load condition of the engine. It could also mean that two or more components attain a higher degree of compatibility. In this case the result can also be an improvement in fuel consumption or an increase in thrust of the whole propulsion system. Improvements can furthermore have an influence on performance alternation through the acceleration time; excessive RPMs and excessive temperatures can be avoided and so on. Improvements in noise and exhaust emission can likewise be attained through variable geometry. In summary, the mode of operation is implicated here in its most vast sense.

Occassionally it is said that the so called adaptive adjustable geometry (for example adjustable guide vanes in the compressor) is in principle something different than the type of geometry that affects the range of operation depicted in performance maps for turbo engines, and which thereby influence the thermodynamic cycle (example: change in area of the exhaust nozzle). This differentiation can ultimately only point out gradual differences and offers no fundamental supplement. The purpose of the variable guide vane adjustment in the compressor of most of the engines built today is to adapt the characteristics of the compressor stages at partial load. On the other hand, stator adjustment (of course with different angle changes) allows also the variation of flowrate and pressure at a given compressor speed. In this case the thermodynamic cycle is also influenced.

Employment of variable geometry

Fig. 1 a depicts a visionary engine. The purpose of the disclosed difference between the upper and lower engine halves is to point out possible localities for variable geometry. In spite of the monstrous construction, there is no claim of completeness. The numbers are so arranged that they give the chronological order of the engine sections passed by the gasflow. In other words, the lowest numbers represent the inlet diffusor, whereas the highest numbers represent re-heating and the outlet section.

The left side of the table in fig. 1 b shows those sections (localities) with variable geometry that are todays practice, whereas the right side shows those that have been tested in prototypes, discussed in technical bureaus and have been applied for patent. Even after considering the diverse alternative solutions that are shown on the left side, still about 10 adjusting devices exist which are functionally distinguishable.

Fig. 2 shows the 3 propulsion categories that have had a marked influence on the development of aeronautical and space engineering from its very beginnings until today. The time span ranges from 30 to 40 years respectively. This time span covers the beginning and endphase of the piston-powered engines, whereas the development of the turbojet engines and chemical rockets will hopefully continue in an unrestrainable pace.

The trend towards complexity in propulsion systems is obvious. If one also takes into account all of the regulatory devices (not all pictured in the figures), it is easy to see the risk involved for the reliability of the propulsion system, for example due to the possible disorderly movement by quick throttle action (danger of oscillations or even inadvertent effects arising from the variable geometry).

This remark is not to document a negative attitude towards variable geometry, but to point out the necessity of considering the system as a whole. In order to reach an evaluation within the scope of the program (flight missions) in question, the advantages of the theoretical thermodynamic gain (for example improvement of fuel consumption at certain load conditions) must be compared with the disadvantages such as complexity, reduction of safety and increase in cost.

From the just mentioned we can draw three conclusions:

1) The chosen system in which the pros and cons of a turbine with adjustable guide vanes is to be investigated should be relatively simple; we chose a twospool turbojet engine. Strictly speaking, with exception of an adjustable thrust nozzle, there is no variable geometry. We should make as little use as possible of those variants listed in fig. 1. The safety risk with respect to control-oscillations or similar phenomena, even with an additional section with variable geometry, is in this case acceptable.

2) If adjustable guide vanes are being employed in the hot section of the engine, then those should be chosen which incorporate the minimum of technological risk. The guide vanes should be adjusted and not the rotor blades, and this should not occur in the hot high pressure turbine, but in the low pressure turbine which operates at a lower temperature. Besides this, only a single stage should be considered.

3) Lastly, a sophisticated engine with many free parameters was not chosen, because with the numerous functional factors everything could be proven. With bypass or three-stream engines with variable thrust nozzles in the bypass, variable load distribution in primary and bypass flow, duct burners, variable thermodynamic process through mixed or direct jet propulsion, jet-interpenetration through special valves etc. it is possible, solely through adequate selection of the efficiencies in compressors (and/or turbines) to have, for example, the economy of the turbo engine improved or impaired when adjusting the turbine stator.

Example for the utilization of an adjustable turbine

Problem

After these more or less general remarks concerning the pros and cons of an adjustable turbine, we want to discuss now a specific case of utilization. We have given the following hypothetical flight mission:

A military aircraft with a single engine must be able to reach Mach 0.85 at an altitude of 11 km (~ 36,000 ft) without the use of the possibly existent afterburner. This condition represents the optimum lift-drag-coefficient. On the other hand, the aircraft should be able to fly for a long time at sea level. This would for example be the case when the aircraft has to stay in a holding pattern outside the range of enemy radar.

The engine is a twospool turbojet with an adjustable stator in the low pressure turbine (fig. 3). For the design condition $H_0 = 11$ km, $M_0 = 0.85$, a total pressure ratio of $\Pi_{tot} = \Pi_{LP} \cdot \Pi_{HP} = 3.5 \cdot 4.5 = 15.75$ is given.

The low pressure turbine entry temperature was given to 1250 K, from which a combustion chamber temperature of 1450 K results.

Fig. 4 depicts a schematic view of an adjustable turbine stator. An (additional) radial gap has to be considered, which is dependent on the hub-tip-ratio and the adjustment range $\Delta A/A$. This causes a drop in efficiency of $\Delta\eta_{LPT} = 1.5 - 2.5$ % in the low pressure turbine.

Results for fixed geometry

Fig. 5 shows the results for flight time and range during holding with a given constant fuel mass for an engine without variable geometry. The Mach numbers and the lift coefficients are given for the individual points. The optimum lift coefficient is $c_{L,opt} = 0.347$, which gives a flight time at minimum drag of 1.37 hours.

Results for adjustable turbine stator

Fig. 6 displays the same results as in fig. 5; the dashed line pictures additionally those for an adjustable low pressure turbine. The additional losses due to radial gap were accounted for by a 2 % diminuation in efficiency. The curve shows the results for the best adjustment of the turbine stator. It may seem surprising at first that this adjustment is the same as at the design condition. The diminuated efficiency of the low pressure turbine results in an insignificantly shorter flight duration when compared with the non-variable turbine.

What is the reason for the best adjustment of the turbine stator being the same as its position at the design condition? Regarding this question, fig. 7 displays a few parameters of the engine at Mach 0.35 during holding. The flight duration is a maximum at this flight speed (see also fig. 5).

One notices at first that the flight time varies by only 1/2 % when the stator area varies by about 13 %. The reason for this is that the high pressure turbine entry temperature, the pressure ratio and the "mean" effiency level of turbo components remains practically constant. The result is merely a redistribution of the pressure ratio between the low pressure and high pressure compressor. When opening the stator, there is a decrease in the RPMs of the low pressure shaft and an increase in the high pressure rotor. The same tendency is reflected by the pressure ratios. The final result is that it is best not to adjust the low pressure turbine due to the fact that the loss tendencies, i.e. the partial efficiencies, compensate each other.

The change in the mode of operation in relation to the stator adjustment is also depicted in those performance maps, which were the basis of the calculation (fig. 8 and 9). The scales in these diagrams were modified with the aid of correction factors, so that they correspond with the investigated engine.

Results for adjustable thrust nozzle

As can be seen in fig. 6, the adjustment of the turbine stator, as well as that of the thrust nozzle were investigated. When the thrust nozzle was opened 20 – 30 %, an increase in flight duration in the whole field was the result. The corresponding values for pressure, temperature and RPM are shown in fig. 10.

The total pressure ratio and the entry temperature in the high pressure turbine change greatly. Even the compressor massflow, which was constant over the adjustable range in fig. 7, is now variable.

The change in the specific fuel consumption can actually be derived from the curve for the flight duration, which, by the way, has a much more coarse scale than in fig. 7. Thrust and Mach number are constant, and the fuel mass is 3000 kg in all cases.

The corresponding operation conditions are plotted again in the performance maps of fig. 8 and 9, and thereby round up the data given by fig. 10.

It was also investigated what advantages an adjustable turbine stator would have with an open nozzle. For the flight duration, the result was the same as the before discussed: It is best not to adjust the stator.

As can be seen from fig. 8, opening the stator results in a greater safety span with respect to the surge line of the low pressure compressor. Though this effect is of actual advantage, the expense of an adjustable stator in the turbine for this reason is not justified.

Summary

An adjustable stator in the low pressure turbine of a twospool turbojet is of no explicit advantage with regard to the discussed problem.

As was mentioned above, the shape of the compressor performance map could be so altered (manipulated), that, even in our very simple example, a stator adjustment would result in an improvement of the fuel consumption. In other words there can be engines having such a concept, that stator adjustments prove to be of advantage. Such a result cannot be considered as generally valid, because only the characteristic of one specific engine can be altered, and then only in the form of a remedial measure. Variable geometry can be of advantage for those flight missions in which long periods of time are run at very low partial load. This may be comparable to the situation of an automobile in inner city traffic. The advantage with respect to an automotive gas turbine engine is quite considerable. But flight missions with comparable conditions are very seldom.

Literature

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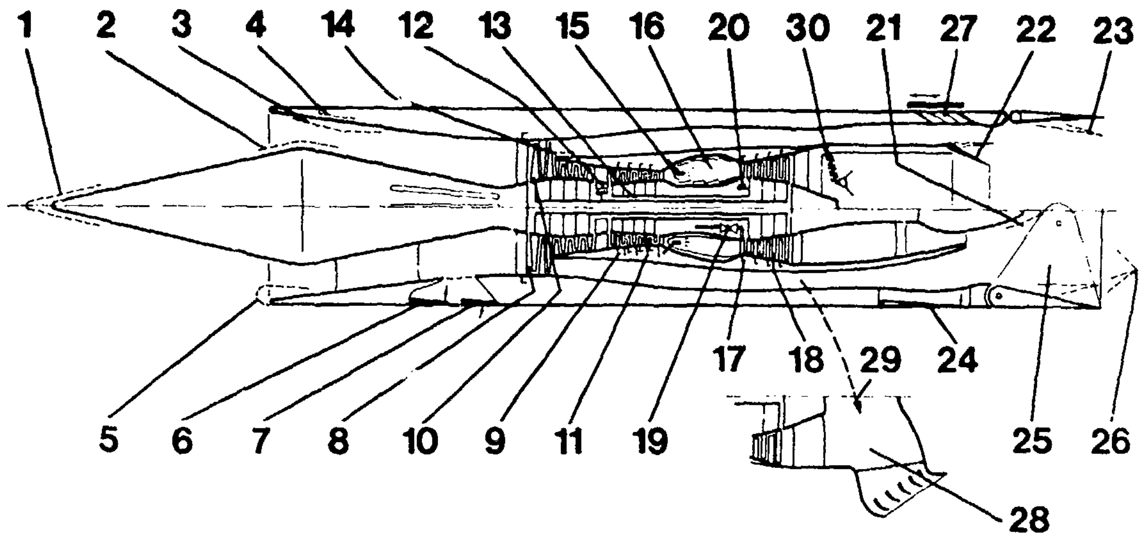

Fig.1a: Visionary jet engine with possible localities for variable geometry

Explanations to variable geometry

	realized in production engines		prototype test / proposal / patent pending etc.
1	Inlet cone displacement in direction of engine main axis	5	Inflatable inlet lip
2	Inlet cone adjustment perpendicular to engine main axis	8	Retractable shutter in front of first compressor blade
3	Angle change of inlet lips	12	Pneumatic or mechanic fit adjustment of labyrinth seals
4	Displacement of inlet lips	14	Splitter adjustment on bypass engine
6	Adjustable auxiliary inlet	16	Combustion chamber with 2 or more primary zones and variable mixture
7	Blow-off between supersonic inlet and engine	17	Turbine with adjustable guide vanes
9	Compressor with adjustable stator	18	Turbine with adjustable rotor blades
10	Compressor with adjustable rotor	19	Throttling or shut-off cooling air channel for hollow blades
11	Controlable blow-off orifices between compressor stages	20	Fit adjustment of labyrinth seals
13	(Free rotating) multispool engine	25	Pneumatic thrust reversal
15	Combustion chamber with switchable injection nozzles (main nozzles - auxiliary nozzles)	27	Shift blade row for thrust reversal
21	Adjustment of minimal thrust nozzle cross-section through shifting of axial core	30	Re-heat with extendible flame holder (mechanically or pneumatic)
22	Adjustment of minimal thrust nozzle cross-section with flaps		
23	Angle change in thrust nozzle expansion section		
24	Rear-end ventilation of expansion section		
25	Mechanical thrust reversal		
26	Shutting and opening the relief channel for thrust reversal		
28	Turnable nozzles for thrust vector control		
29	Dual thrust systems with variable exhaust deflection		

Fig.1b: Explanations to variable geometry

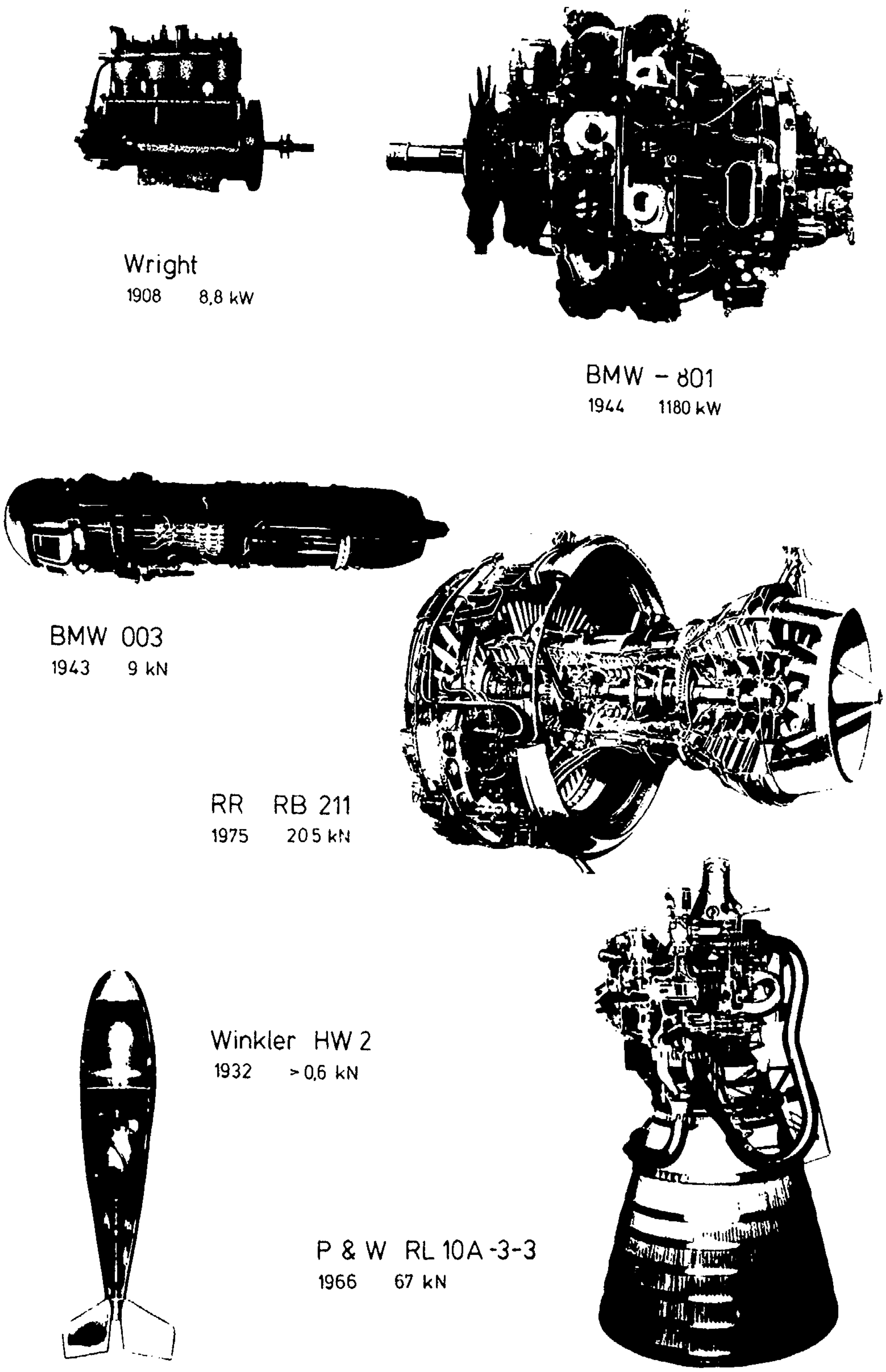

Fig. 2: Development of propulsion systems

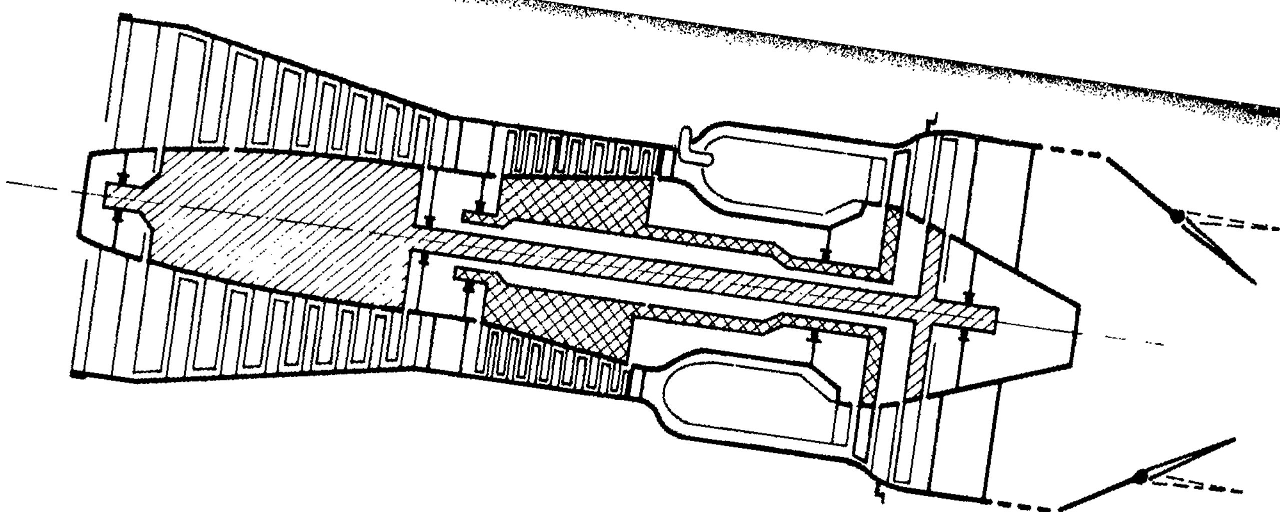

Fig. 3: Twospool turbojet engine

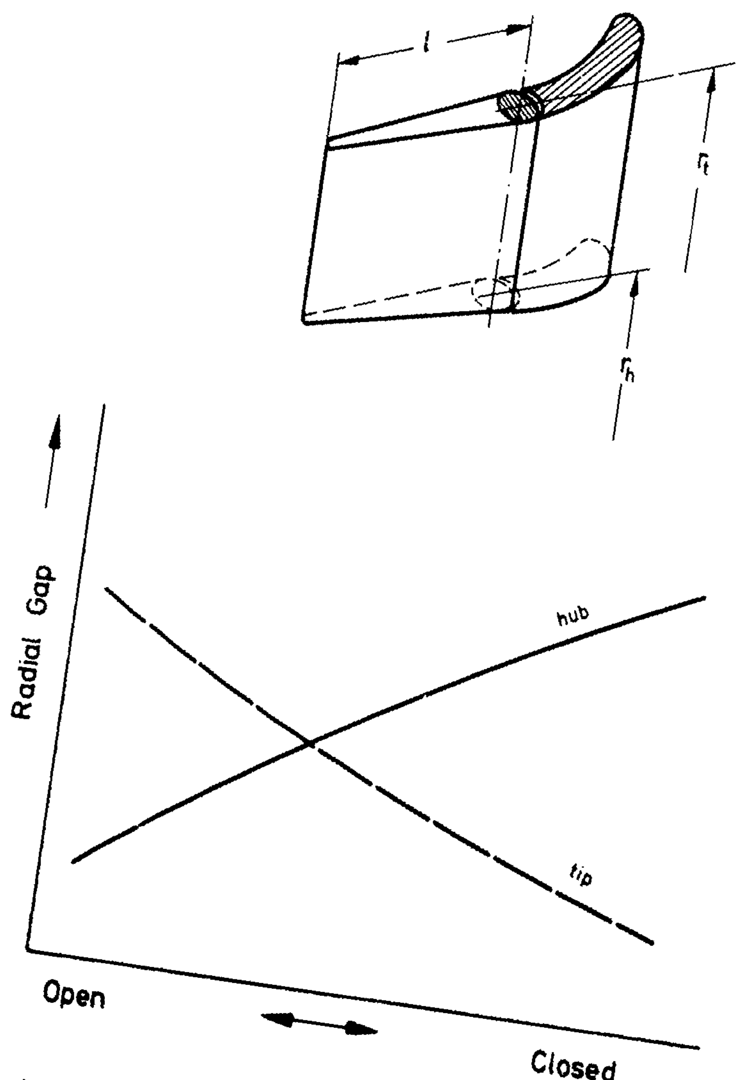

Fig. 4: Kinematics of adjustable turbine guide vane

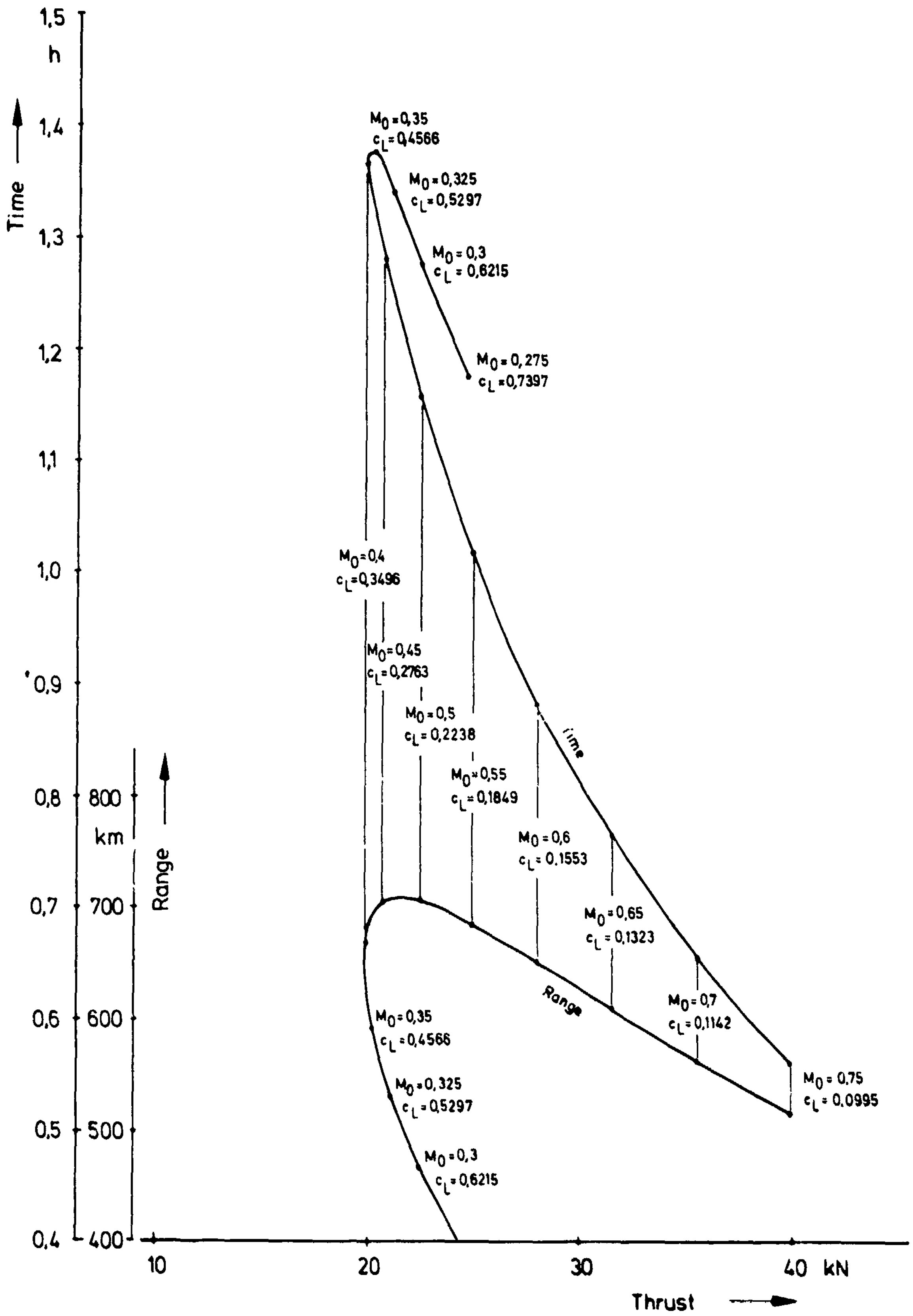

Fig. 5 : Flight duration and range with non-adjustable geometry
(invariable geometry)

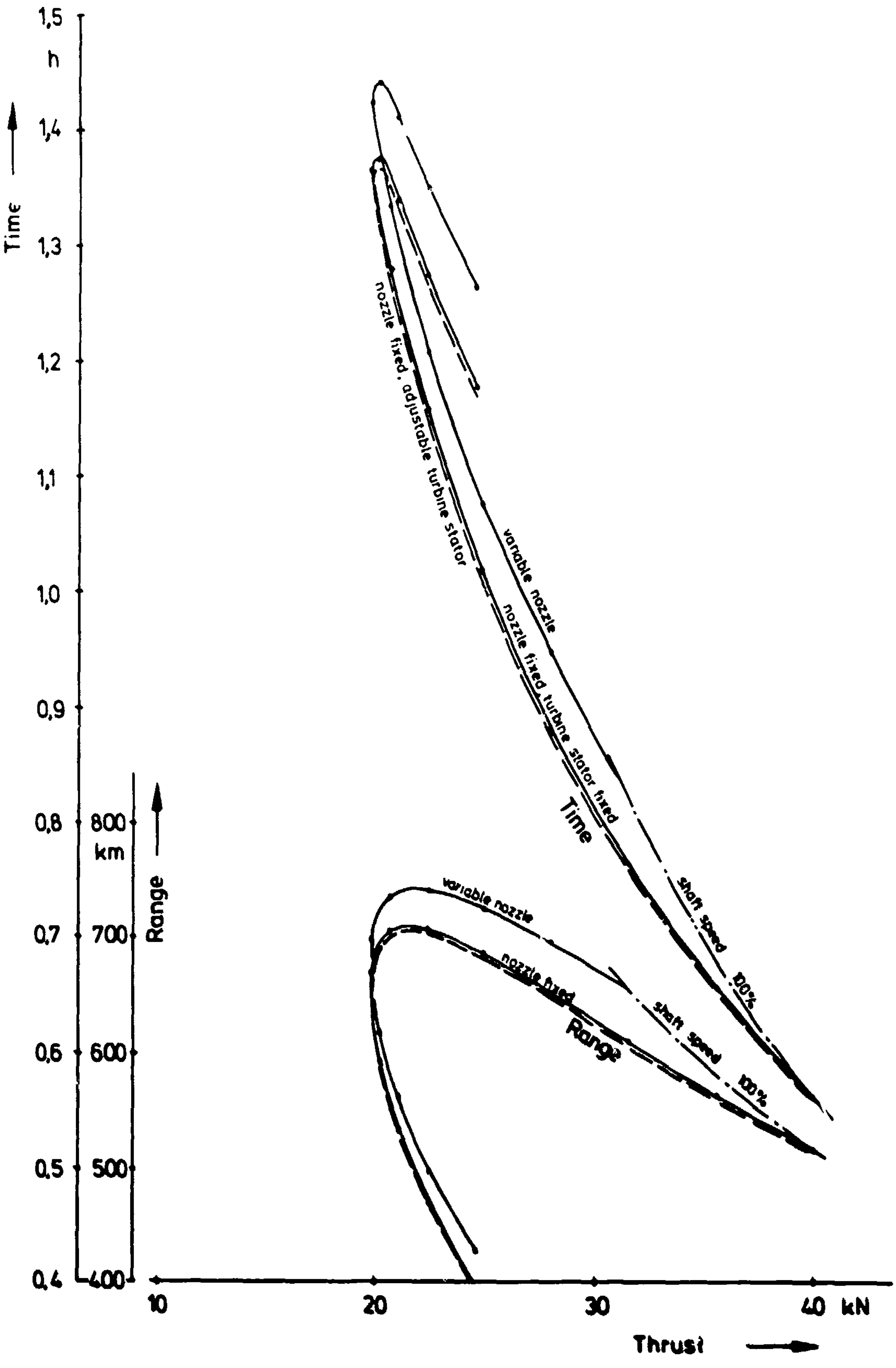

Fig. 6: Flight duration and range with adjustable geometry

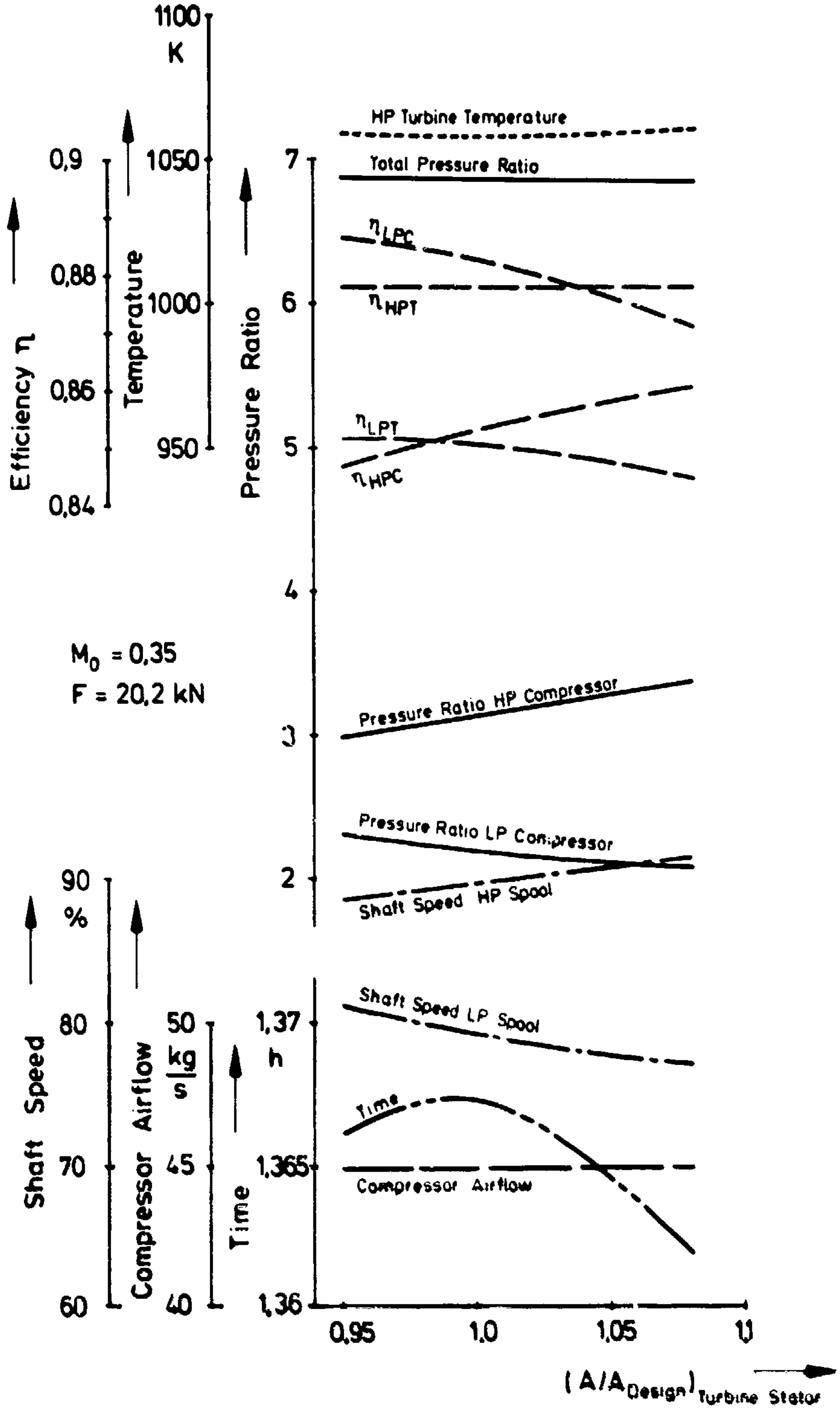

Fig. 7: Engine parameters with variable stator of low pressure turbine

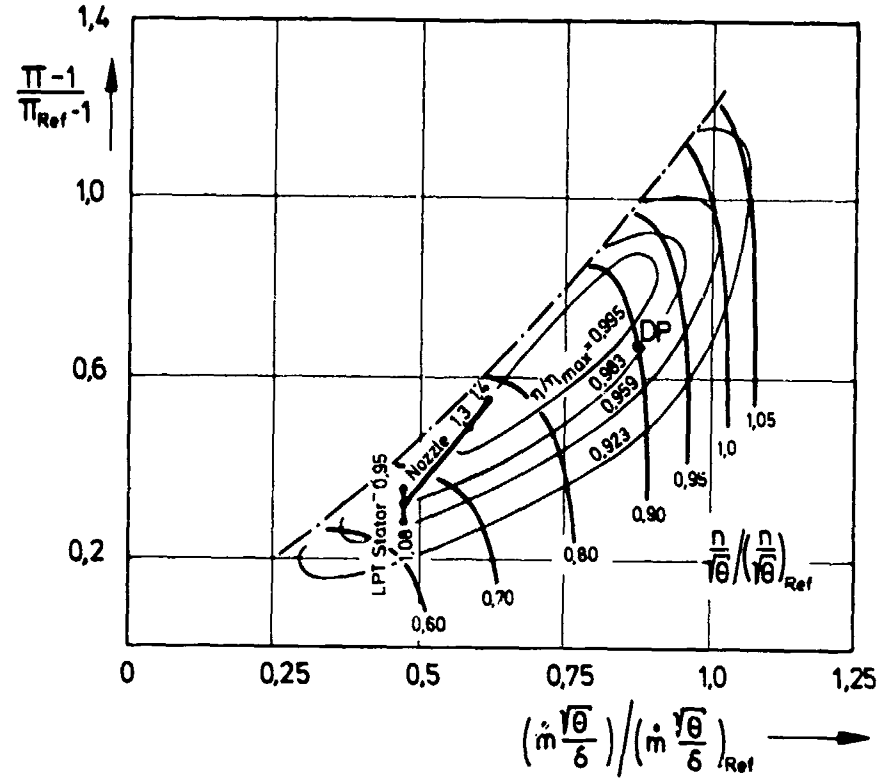

Fig. 8 : Performance map of low pressure compressor

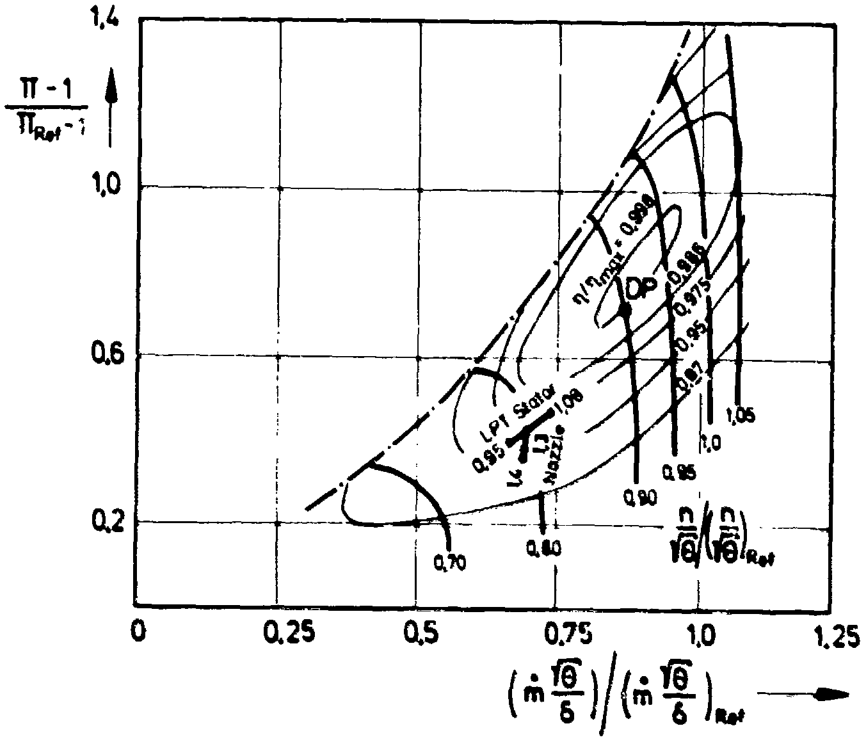

Fig 9 Performance map of high pressure compressor

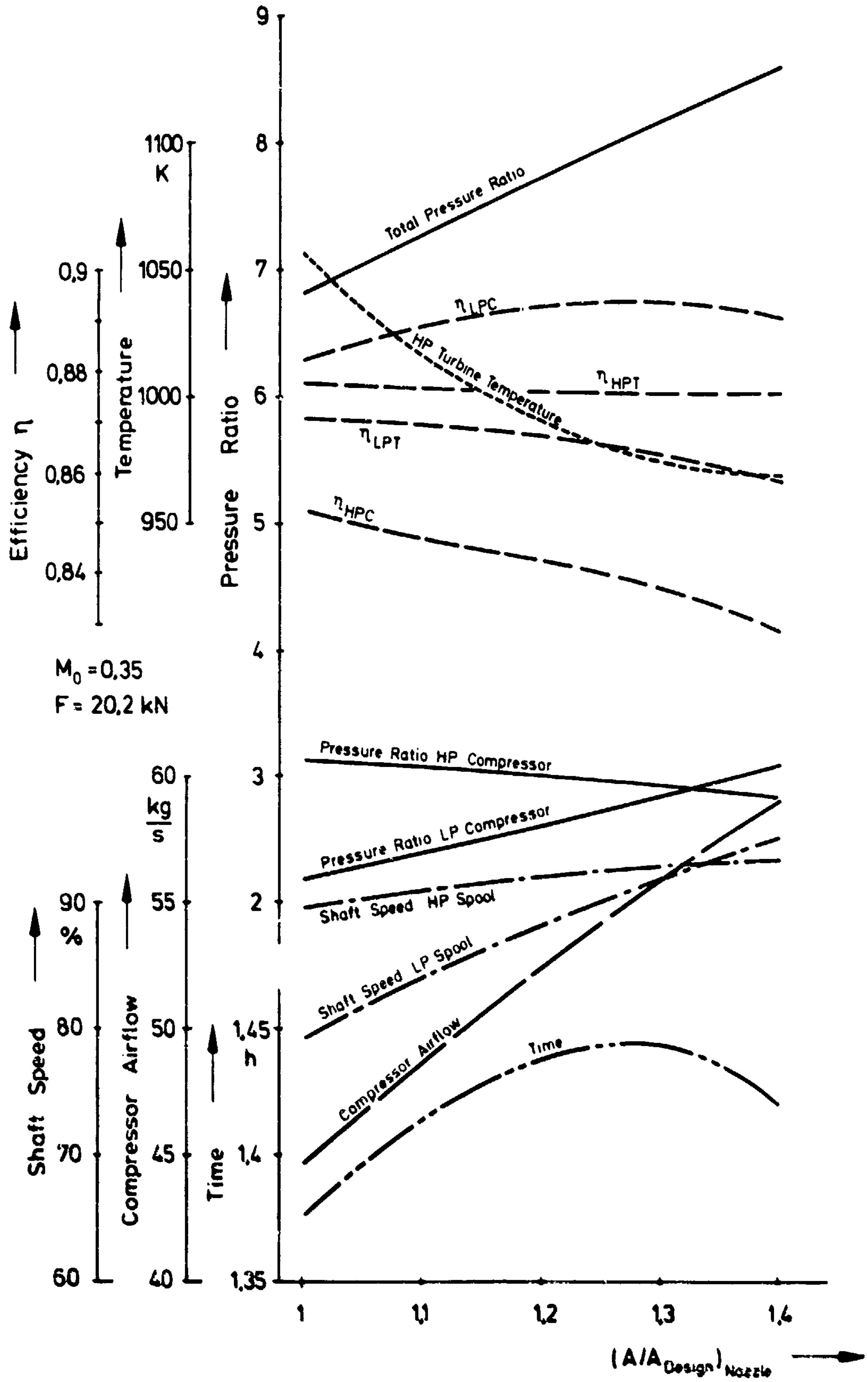

Fig. 10: Engine parameters with variable thrust nozzle

DISCUSSION

H.GRIEB

In the written paper is mentioned a LP turbine efficiency penalty of 2% at design point due to the variable geometry. In our studies we found a strong influence of turbine off-design performance (off-design efficiency) not only on SFC, but also on the range of engine performance flexibility achieved. Did you introduce turbine off-design performance characteristics in your study?

Author's Reply

I did not show the performance maps of the variable turbine. The calculations were based on five different low pressure turbine maps for five different angles of the stator adjustment. These performance maps were not measured but computed. In the different performance maps the efficiency level is also different.

S.Boudigues

Je suis d'accord avec les conclusions des auteurs sur l'efficacité relativement médiocre du distributeur de turbines BP, mais malgré les difficultés technologiques que cela présente il serait bon de regarder l'effet de la variation de la section du distributeur HP. En effet, à ce moment-là on dispose de trois paramètres qui permettent de situer le point de fonctionnement au point optimum, aussi bien dans le champs de compresseurs basse pression que dans le champs de compresseurs haute pression, et à ce moment-là, au prix d'une complication technologique certaine et de difficultés de manipulation de section dans une région très chaude, on obtient quand même des gains de consommation spécifiques ou de poussées qui sont très supérieures à celles qui sont affichées par la seule variation de la turbine basse pression.

Author's Reply

Nous sommes d'accord avec vous Monsieur Boudigues, et, il a d'ailleurs été demontré, lundi ou mardi, qui le changement de la section de la turbine haute pression rendait davantage. Le but de notre étude était simplement de prouver que, même avec des hypothèses très simplificatrices telles que changer uniquement la tuyère d'une part, et le distributeur de turbine BP d'autre part, on pouvait obtenir le résultat que nous venons de démontrer. Mais nous avons également dit, et ceci est indiqué dans notre conférence, qui si l'on avait par exemple changé la position des courbes de rendement dans les compresseurs basse pression et haute pression, on aurait pu obtenir, même pour cette étude très simplifiée, une amélioration de la consommation spécifique; mais si le déplacement de ces courbes de rendement était fait d'une façon différente, on aurait même pu obtenir une altération. Donc notre but était plutôt de montrer, qu'un changement de la section de la tuyère apporte de toute façon quelque chose de positif (fait, généralement admis aujourd'hui), tandis qu'un changement de la section des distributeurs n'est pas forcément aussi efficace. En outre on devrait épuiser toutes les possibilités d'influencer le comportement de la machine avant l'introduire un distributeur orientable d'une turbine HP, dont la technologie est particulièrement délicate en cas d'un réacteur moderne à température élevée.

POTENTIAL IMPROVEMENTS IN ENGINE PERFORMANCE USING A VARIABLE GEOMETRY TURBINE

by

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SUMMARY

A trend in military aircraft is toward increasing thrust loading for improved maneuverability coupled with a requirement for extended subsonic cruise range at low power settings. Conventional turbine engines designed to meet these requirements must operate over large ranges of airflow between maximum power and cruise. As a result, the inlets and nozzles designed for these engines cannot perform efficiently with the low engine airflow rates typical of subsonic cruise operation. Variable turbine geometry offers a promising approach for obtaining both high thrust loading and efficient cruise performance by permitting large amounts of thrust modulation at constant airflow rates. Installed performance comparisons between a fixed geometry turbofan, a fixed geometry turbojet, and a variable geometry turbojet are made at typical subsonic cruise, supersonic cruise and combat flight conditions. This study indicates that for applications where both subsonic and supersonic fuel consumptions are important, a variable geometry turbine turbojet can offer substantial reductions in fuel usage.

NOMENCLATURE

A	turbine inlet area
A_c	inlet capture area
ALT	flight altitude
A_{MIN}	minimum turbine inlet area
A_8	exhaust nozzle throat area
A_9	exhaust nozzle exit area
A_{10}	maximum fuselage cross-sectional area per engine
A_{11}	fuselage cross-sectional area at airplane connection point per engine
CV	exhaust nozzle velocity coefficient
FG_{ID}	ideal gross thrust
FN	installed net thrust
FN_{ID}	ideal net thrust
$FN_{ID_{MAX_{TF}}}$	ideal net thrust for the turbofan engine at maximum afterburning power
MN	flight Mach Number
PAMB	ambient pressure
PS_9	exhaust nozzle exit static pressure
P_8	exhaust nozzle throat total pressure
q_∞	free stream dynamic pressure
SFC	installed specific fuel consumption (pounds of fuel per hour per pound of net thrust)
SFC_{ID}	ideal specific fuel consumption
$SFC_{ID_{MAX_{TF}}}$	ideal specific fuel consumption for the turbofan engine at maximum afterburning power
$SFC_{ID_{TF\ AT\ MN\ =\ 2.5}}$	ideal specific fuel consumption for the turbofan engine at flight Mach number = 2.5
$W_{A_{ENG}}$	engine demand airflow
$W_{A_{INLET}}$	inlet supplied airflow
ΔCD_{AE}	throttle dependent aft-end drag coefficient

ΔCD_{INLET} throttle dependent inlet drag coefficient

$\Delta D_{AFT-END}$ throttle dependent aft-end drag

ΔD_{INLET} throttle dependent inlet drag

ΔFN_{η_R} thrust loss due to inlet pressure losses

η_R inlet pressure recovery

I. INTRODUCTION

Fighter aircraft of today and the future must be flexible. That is, they must operate effectively over an extremely wide range of diverse mission requirements. This emphasis on versatility has a tremendous impact on the propulsion system for such an aircraft. It must be capable of efficient low power operation for loiter and extended subsonic cruise range. It must deliver very high thrust for transonic combat maneuverability and for supersonic cruise and dash requirements. In addition to the thrust level, the fuel burned at these conditions is an important consideration since it also will significantly affect the capability and size of the overall weapon system.

From an engine thermodynamic cycle standpoint alone, meeting this variety of demands is a difficult undertaking. The problem is even further compounded by the propulsion system installation losses due primarily to the changes in engine airflow demand between cruise and maximum power conditions. The inlet and nozzle must be designed to accommodate the maximum airflow rate to be encountered anywhere in the flight envelope. This results in high inlet spillage drag due to reduced airflow and high aft-end drag due to a closed down nozzle at the low power settings associated with subsonic cruise. As the trend toward increasing disparity between the maximum thrust and cruise thrust continues, this installation problem will become even more severe. One possible solution is to develop an engine cycle that will provide a range of thrust levels at a constant airflow rate. This can be achieved through the use of variable turbine geometry.

The propulsion system most often selected for missions with diverse requirements is a mixed flow afterburning turbofan. The afterburner provides the required maneuvering thrust while the inherently high propulsive efficiency of the fan provides good uninstalled performance at low power settings. Two drawbacks to this type of cycle are its high specific fuel consumption at high power conditions, and its susceptibility to high installation losses at low power conditions.

On the other hand, a turbojet cycle generally provides good performance at high thrust levels but has poor performance (installed and uninstalled) at low thrust conditions. A variable geometry turbine could be used to significantly improve the turbojet's part power fuel consumption and installation characteristics.

To illustrate the effectiveness of this design approach, both uninstalled and installed engine performance have been calculated for a variable turbine geometry turbojet, and the results have been compared with similar estimates for a fixed turbine geometry turbojet and a fixed turbine geometry turbofan at several important operating points in a typical fighter mission. All three cycles were sized such that they had approximately the same installed maximum thrust at Mach number .9, and an altitude of 30,000 feet, a typical transonic combat sizing point. The design characteristics of these engines are summarized in Table 1.

TABLE 1

CYCLE DESIGN PARAMETERS

	VARIABLE TURBINE TURBOJET	FIXED TURBINE TURBOJET	FIXED TURBINE TURBOFAN
OVERALL PRESSURE RATIO	15	15	24
BYPASS RATIO	.1*	.1*	1.0
TURBINE INLET TEMPERATURE (°F)	3000	3000	3000

The turbofan design parameters are typical of those values which several recent preliminary design studies have shown to be near optimum for an advanced fighter application. The bypass ratio represents a compromise between subsonic cruise and maximum power specific fuel consumption. Of course, changes in aircraft and mission design variables will alter the choice of engine cycle variables. The turbojet design parameters were selected to insure that this would be a relatively simple, single spool machine.

In the analysis of these cycles, it will be shown that the variable geometry turbine in a turbojet engine will improve the subsonic installed cruise fuel consumption to the point where it is <u>lower</u> than that of the turbofan while retaining the good supersonic and high thrust performance that is characteristic of a turbojet cycle.

This analysis does not represent a complete cycle selection process. To do that job properly, the mission variables, airframe variables and engine variables must be considered simultaneously. However, the results do indicate that this concept is promising and deserves further detailed study for applications where both subsonic and supersonic fuel consumptions are important.

*Air is removed from the compressor and used to cool the afterburner liner and nozzle.

II. ENGINE OPERATING CHARACTERISTICS AND UNINSTALLED PERFORMANCE

This section of the paper deals with the part power operating characteristics of a variable geometry turbine turbojet. As was previously mentioned, the variable geometry is being used to improve the fuel consumption at subsonic cruise. In order to illustrate this concept, the reduced power operating characteristics of the variable turbine turbojet are compared with those of a fixed turbine geometry turbojet at .9 Mach number and an altitude of 30,000 feet.

For both cycles, the initial decrease in thrust is achieved by reducing the augmentor fuel flow. There is no change in the rotating components. Further decreases in thrust below intermediate power are accomplished by reducing main burner fuel flow with a resulting decrease in turbine inlet temperature. This results in a rematching of the turbomachinery for a conventional engine.

The operating characteristics for the fixed and variable geometry turbojets are shown in figure 1. Compressor pressure ratio, percent of design airflow, and turbine inlet temperature are plotted as a function of percent of maximum thrust. Turbine work requirements dictate that engines operate over most of their thrust range with the turbine nozzle "choked." As a result, for a fixed turbine nozzle, as turbine inlet temperature is reduced, both compressor pressure ratio and mass flow must decrease. This is shown by the solid line in figure 1. The compressor is simply moving along an operating line on its performance map.

The behavior of the variable turbine engine is distinctly different. As turbine inlet temperature is reduced, the turbine nozzle area can be changed to allow the compressor to remain at its design point. This constant compressor match point operation can be maintained until limit loading is reached on the turbine. In an effort to postpone this limit loading to as low a power setting as possible, the afterburner liner cooling flow was turned off during non-augmented operation and consequently this air was no longer lost to the cycle. When constant match can no longer be sustained, the engine is operated in a conventional manner with only sight variations in area to maintain adequate surge margin. The dashed line of figure 1 shows that in this instance the design compressor corrected flow and pressure ratio were held constant down to a thrust of approximately 35% of the maximum available at this flight condition. This corresponds to a turbine inlet temperature of 2000°R.

Variable turbine geometry also provides an additional benefit in the transonic regime. At these flight conditions, the engine with fixed turbine geometry cannot be operated at its maximum turbine inlet temperature without exceeding compressor aerodynamic limits. Again, the variable geometry turbine can be used to allow operation at the compressor design point and at the maximum turbine inlet temperature.

The changes in turbine and exhaust nozzle areas required to operate the variable geometry engine are shown in figure 2. As thrust is reduced by reducing turbine inlet temperature, the turbine nozzle area closes down to maintain the desired flow function ($W\sqrt{T}/P_A$). At the same time, the exhaust nozzle throat area opens up in order to increase the turbine pressure ratio allowing design compressor match point operation at turbine inlet temperatures less than design. In contrast, the fixed geometry cycle makes only small variations in exhaust nozzle throat area to maintain the desired surge margin. Figure 2 also shows that operating the engine at the compressor design point by using the variable geometry turbine results in a larger exhaust nozzle exit area for ideal expansion relative to that of the fixed turbine geometry at the same thrust level. This trend will have important implications in the nozzle installation area.

The trends presented for the fixed geometry engine are for a turbojet cycle. The behavior of the fixed turbine geometry turbofan cycle is similar except that the higher augmentation ratio results in constant cycle parameter operation further into the part-power regime.

The data presented in figure 1 show that the variable geometry turbine turbojet has operating characteristics which have the potential for increasing engine life. Two factors which significantly impact engine life are time at temperature and continually changing engine rotational speeds to produce desired thrust levels. The time spent at subsonic cruise represents a significant portion of a fighter mission and the variable geometry turbine turbojet cruises at a turbine inlet temperature approximately 400°R less than the fixed turbine turbojet and 300°R less than the turbofan. Additional benefits might result because the variable geometry turbine turbojet can provide a wide range of thrust modulation without changing the compressor speed.

The uninstalled engine performance at 30,000 feet, Mach .9 is presented for the three study engines in figure 3. Ideal specific fuel consumption divided by the ideal specific fuel consumption of the turbofan at maximum afterburning power is plotted versus ideal net thrust nondimensionalized by the ideal maximum afterburning net thrust for the turbofan. Uninstalled performance is engine cycle performance with fully expanded nozzle flow and without engine installation losses. All the engines were sized to give the same maximum thrust at this flight condition.

The data shows that both turbojets have augmented power specific fuel consumption less than that of the turbofan. The ability of the variable geometry turbine turbojet to operate at the maximum turbine inlet temperature at this flight condition results in a slight reduction in afterburning specific fuel consumption and an increase in intermediate power thrust. The most striking feature of this performance plot is the substantial reductions in part power fuel consumption due to the variable turbine geometry. The ability to develop the same thrust as the fixed cycle, but at a higher airflow, (see figure 1) results in a higher propulsive efficiency. In addition, compressor pressure ratio is also substantially higher at any given thrust level (see figure 1), yielding better thermal efficiency. The net result is nearly a 13% reduction in specific fuel consumption at typical subsonic cruise conditions. However, this improvement is not sufficient to make this cycle better than the turbofan at reduced power. It is only after the installation effects are included that the variable turbine geometry turbojet becomes truly competitive.

III. INSTALLATION CHARACTERISTICS

Several recent fighter aircraft have suffered performance penalties due to high inlet and aft-end losses associated with operating the propulsion system off design at subsonic cruise. At these low power settings, a conventional engine's airflow demand is much less than the inlet can supply. Some of this air must be spilled or bypassed incurring a drag penalty. Also, at reduced power, the nozzle exit area must be closed down, to avoid large internal overexpansion losses. With the nozzle closed down, there is more aft facing area and also an increased probability that flow separation will occur on the external nozzle surfaces. These factors can and frequently do result in a drag penalty.

In contrast, figures 1 and 2 show that the variable geometry turbine turbojet exhibits quite different characteristics. This engine operates at design airflow as thrust is reduced below intermediate power. In addition, the full expansion nozzle area remains essentially constant rather than closing down. At a typical subsonic cruise power setting the variable turbine geometry turbojet will have lower inlet and aft-end installation losses than a fixed turbine cycle. This is shown schematically in figure 4.

It is extremely important that engine cycle studies be done on an installed basis, as these losses may substantially impact the final outcome. The inlet and aft-end drags can be divided into a reference drag component that is independent of power setting and the remaining throttle dependent component. The reference drag is included in the airplane drag polar. Any drag change caused by operating away from the reference conditions is charged to the propulsion system. Inlet pressure recovery effects and internal nozzle performance must also be included when calculating installed performance. The net installed propulsive force can then be defined as:

$$FN = FN_{ID} - \Delta D_{INLET} - \Delta D_{AFT-END} - \Delta FN_{\eta R} - FG_{ID}\,(1 - CV)$$

where FN = installed net thrust

FN_{ID} = ideal net thrust

ΔD_{INLET} = the throttle dependent portion of the inlet drag

$\Delta D_{AFT-END}$ = the throttle dependent portion of the aft-end drag

$\Delta FN_{\eta R}$ = the thrust loss due to inlet pressure losses

FG_{ID} = ideal gross thrust

CV = nozzle velocity coefficient

Inlet Performance

A two dimensional, triple ramp, external compression, Mach 2.5 inlet was used for this study. This type of inlet is typically used for tactical fighters with high maneuverability requirements. Inlet performance consists of inlet pressure recovery and throttle dependent inlet drag.

Inlet pressure recovery trends are shown in figure 5 as a function of flight Mach number. The pressure recovery is a function of shock system losses and subsonic diffuser efficiency. Generally, pressure recovery is considered to be a function of inlet mass flow ratio as well as flight Mach number. However, in order to simplify calculation procedures, the variation with mass flow ratio was neglected since differences between the three engines in pressure recovery due to this variable should be small, and therefore not alter the results of this study.

In this study, the reference inlet drag has been defined as the inlet drag occurring at maximum allowable inlet airflow at each flight condition. This drag is included in the airplane drag polar. The throttle dependent inlet drag coefficient for two Mach numbers is shown in figure 6 as a function of the ratio of engine demand airflow to inlet supply airflow. Each engine had an inlet sized to meet the maximum airflow demanded by that cycle throughout the entire flight envelope. Figure 6 shows the characteristic throttle dependent drag coefficient increase with decreasing mass flow (engine demand). Knowing the engine operating corrected airflow and the inlet capture area, the throttle dependent inlet drag, chargeable to the propulsion system can be simply calculated by the following equation:

$$\Delta D_{INLET} = \Delta CD_{INLET}\, q_\infty A_C$$

Several typical operating points are plotted on Figure 6 for the three different engines at .9 Mach number. The solid symbols are representative of operation at intermediate power and above. At this condition, all three cycles are clustered close together at relatively high mass flow and low drag level . However, noticeable differences appear as the engines are throttled back to cruise power settings, as shown by the open symbols. Both fixed turbine geometry cycles show reductions in mass flow and sharp increases in throttle dependent inlet drag. However, the capability of the variable geometry turbine turbojet engine to maintain near maximum airflow at cruise conditions results in substantially less drag. In fact, there is nearly a 83% drag reduction relative to the fixed turbine turbojet and a 79% reduction relative to the fixed turbine turbofan.

Aft-End Drag

A variable convergent-divergent nozzle was selected for all three engines used in this study. In this type of nozzle, the throat area, A_8 (see figure 7), can be varied to provide the desired engine matching. The nozzle exit area, A_9, can also be varied to obtain the maximum thrust minus external drag for the exhaust system. A close spaced twin engine installation was assumed.

The internal nozzle performance was calculated in terms of a nozzle velocity coefficient, CV, which is the ratio of actual gross thrust to ideal gross thrust. The internal nozzle losses include leakage, flow separation on internal nozzle surfaces, shocks, internal wall divergence, and frictional losses along the internal nozzle walls. These losses are calculated using an empirical correlation which is a function of nozzle area ratio (A_9/A_8) and nozzle divergence angle. Nozzle losses associated with either overexpansion or underexpansion of the flow were also included. These were calculated assuming isentropic, one-dimensional flow. The overall nozzle velocity coefficient is displayed in figure 8 as a function of nozzle pressure ratio and nozzle area ratio. The peaks in the curves represent fully expanded flow conditions.

The external aft-end drag is made up of the pressure drag and frictional drag over the entire aft-end of the airplane excluding control surfaces. Because of the interactions of the airframe and nozzle flow fields, it is important to include the drag of the complete fuselage structure from the maximum fuselage cross-sectional area, A_{10}, rearward. This ensures a complete accounting of the engine installation effects. The reference drag level is defined as the drag resulting when the nozzle external flaps were open to produce a cylindrical contour ($A_9 = A_{11}$), with fully expanded internal flow. This drag is included in the airplane drag polar. The remaining drag is charged to the propulsion system and is the drag resulting from changes in the nozzle exit area from the cylindrical position and changes in the internal flow from fully expanded conditions. The throttle dependent aft-end drag can be calculated by the following equation:

$$\Delta D_{AFT\text{-}END} = \Delta CD_{AE}\, q_\infty\, A_{10}$$

Figure 9 represents a typical throttle dependent aft-end drag map for .9 Mach number. Throttle dependent drag coefficient is plotted as a function of the ratio of nozzle exit static pressure to ambient pressure for lines of constant nozzle exit area divided by maximum fuselage cross-sectional area. The static to ambient pressure ratio takes into account the effects of the exhaust plume on afterbody drag. The area ratio takes into account fuselage closure effects on the drag. In order to use this throttle dependent nozzle performance map, estimates of maximum fuselage cross-sectional area for an airplane using each cycle were made. These estimates reflected both engine size and shape.

Several interesting trends are apparent from the data in figure 9. First, the larger the nozzle exit area relative to the fuselage maximum cross-section area, the lower the drag. This is primarily due to less aft facing projected area and reduced separated flow areas. Secondly, as the nozzle operates more and more underexpanded (higher exit static pressure) the drag is reduced. Increased recompression due to the increased turning of the external flow to follow the underexpanded exhaust plume could cause this effect. The characteristics depicted by figure 9 are considered typical and were used for this study. However, exhaust system performance characteristics are sensitive to specific configurations and operating conditions and should not be generalized for use in other than trend studies such as this.

Several typical nozzle operating points for the three cycles studied are plotted on figures 8 and 9. The solid symbols represent a maximum afterburning power setting while the open symbols show operation at a typical subsonic cruise power setting. At maximum afterburning all the engines operate with the nozzle in a wide open position and at low drag levels. However, at cruise, the fixed turbine geometry cycles operate with the nozzle closed down and suffer a large increase in drag. Figure 8 shows that these cycles must also trade away internal performance in order to maximize thrust minus drag. On the other hand, the variable turbine geometry turbojet operates at cruise with a larger nozzle exit area which results in nearly a 40% throttle dependent drag reduction, relative to the other cycles. Note also that this was accomplished without paying as much of an internal nozzle performance penalty.

To give a better idea of the magnitude of the throttle dependent installation losses, figure 10 shows them as a prevent of aircraft drag at a typical subsonic cruise flight condition. Throttle dependent installation effects make up nearly 20% of total aircraft cruise drag for both fixed geometry turbine engines. The variable geometry turbine turbojet, by better matching the inlet and aft-end has reduced these losses to only 10% of total aircraft drag. This will translate to cruise fuel consumption payoff, relative to the fixed cycles.

Historically, the magnitude and impact of high performance aircraft engine installation losses are underestimated at the beginning of aircraft development. The magnitude of the losses resulting from airframe/engine interaction are not known with certainty until flight test. At this time it is too late to change to an engine cycle which is less sensitive to installation losses. Because the variable geometry turbine turbojet will better match the inlet and nozzle it is less sensitive to these losses and therefore, the risk and impact of installation losses on system performance will be minimized.

IV. INSTALLED PERFORMANCE

Combining the inlet and nozzle performance with the uninstalled engine performance characteristics permits calculation of the overall installed performance for each of the three engine cycles. Installed performance comparisons are presented at several important mission flight conditions. Installed specific fuel consumption relative to the turbofan ideal specific fuel consumption at maximum afterburning power is plotted versus installed net thrust nondimensionalized by the turbofan ideal maximum afterburning thrust at that condition.

Figure 11 presents the installed performance comparisons at a subsonic cruise flight condition. The improvement in uninstalled performance due to the addition of the variable geometry turbine coupled with the reduced installation losses result in a cruise fuel consumption for the variable geometry turbine turbojet which is about 22% less than that of the fixed turbine turbojet and 5% less than the turbofan.

Figure 12 displays installed performance at a subsonic combat flight condition, figure 13 at a supersonic cruise flight condition; and figure 14 for an aircraft acceleration. The fuel consumption at these flight conditions relative to the fixed turbine turbofan are summarized in table II.

TABLE II

RELATIVE FUEL CONSUMPTIONS AT CRITICAL MISSION POINTS

CONDITION	MN/ALT/POWER SETTING	FGT/TF	FGT/TJ	VGT/TJ
SUBSONIC CRUISE	.9/30K/PART POWER	1.0	1.22	.95
SUBSONIC COMBAT	.9/30K/MAX A/B	1.0	.94	.91
SUPERSONIC CRUISE	1.6/50K/PART A/B	1.0	.85	.85
ACCELERATION	.6→2.5/36089/MAX A/B	1.0	.96	.92

The data in this table show that the variable turbine significantly improves the subsonic part power fuel consumption of the turbojet without sacrificing the turbojet's high power and supersonic advantage. It shows that this type of cycle has the potential for significantly reducing fuel consumption in a mission where both high power and low power fuel usage are important.

V. VARIABLE GEOMETRY TURBINE DESIGN

One of the most critical aspects of this variable geometry turbine turbojet is the design of the turbine itself which must operate efficiently over a wider range of conditions than a conventional turbine is required to operate. Any undue turbine performance penalty can quickly wipe out the potential gains in cycle efficiency and installation effects.

As the variable geometry turbine turbojet is throttled below intermediate power, airflow and compressor pressure ratio remain constant while turbine inlet temperature is reduced. In order to maintain this constant compressor match to thrust levels near cruise power, the turbine must be able to produce sufficient power to drive the compressor at its design point over a range of turbine inlet temperatures of nearly 1400°R. Turbine exit conditions, such as swirl and Mach number and turbine stresses generated while maintaining the compressor design point over this wide range of temperatures required that the turbine have two stages. In this design, both the first and second stage turbine vanes are mechanically variable: about a 40% area variation from minimum area in the first stage and 15% in the second stage. The turbine exit guide vanes are not variable. At the maximum turbine inlet temperatures assumed for this study both the first stage and second stage vanes and rotor must be cooled.

The major point of the variable turbine concept is to improve the turbojet's part power subsonic cruise performance. The turbine should operate as close as possible to peak turbine efficiency at this condition. This is difficult since the cruise operating point occurs at low turbine inlet temperature where the stage loading and exit swirl Mach number and angle will be highest. Particular emphasis should be given to maximizing turbine efficiency level at this extreme operating condition.

Figure 15 shows the variable geometry turbine performance for the operating conditions where both turbine inlet temperature and turbine area are controlled to produce constant power to maintain the design compressor operating point unitl limit loading is reached. Percent of maximum turbine flow function is plotted as a function of percent of maximum work and percent of minimum flow area. The second part of the figure presents turbine efficiency relative to design versus relative velocity ratio, again for lines of constant percent of minimum flow area. Notice that as the vane area is reduced, (reducing turbine inlet temperature) the turbine efficiency increases.

Two engine operating points are plotted on this turbine performance map. The open symbol represents intermediate power and solid symbol is a typical cruise condition. The cruise operating point falls at the maximum turbine efficiency while the intermediate power point is at a significantly lower performance level. In otherwords, there is a penalty at high power settings for designing the turbine for the minimum temperature case. However, in this study the cruise condition was considered to be most important.

VI. SUMMARY AND CONCLUSIONS

The performance of a variable geometry turbine turbojet was compared with that of a fixed geometry turbine turbojet and a fixed geometry turbine turbofan at several different flight conditions. The variable turbine was used to maintain design compressor operating point as the engine was throttled back to a cruise power setting. This resulted in improved thermal and propulsive efficiencies. Equally important was the reduction in inlet and nozzle losses achieved by maintaining the engine airflow at the design level over a wide range of power settings. The overall result is a cycle which exhibits the good part power installed performance of a turbofan and still retains the good high power and supersonic performance of a turbojet. As a consequence, airplanes incorporating variable turbine geometry turbojets will have the flexibility to provide good performance for a wide variety of missions.

The design of the variable area turbine was shown to be very important. For this study a two stage turbine was chosen in order to increase the range of turbine inlet temperatures over which constant compressor match can be maintained. The turbine was designed to give peak efficiency at the minimum temperature, highest stage loading conditions. This resulted in the best turbine performance at cruise and a penalty at higher power settings.

The study also indicates that the variable geometry turbine turbojet has several operating characteristics which may have payoffs in areas other than performance. Since this cycle cruises at lower turbine inlet temperatures than a fixed geometry cycle and since it can vary thrust over a considerable range without changing the rotational speed of the engine it may have better life characteristics than either the fixed turbine turbojet or turbofan. Also, a system using a variable geometry turbine turbojet will be less sensitive to the impact of underestimating engine installation losses early in development.

This study does not represent a complete cycle selection study. Other parameters such as engine weight and dimension must also be investigated. Engine cycle and airframe variables must be considered interactively in order to properly pick the best cycle. However, this preliminary study indicates that the potential of a variable geometry turbine turbojet is large and it may offer a solution to the growing requirement for flexibility in the propulsion system.

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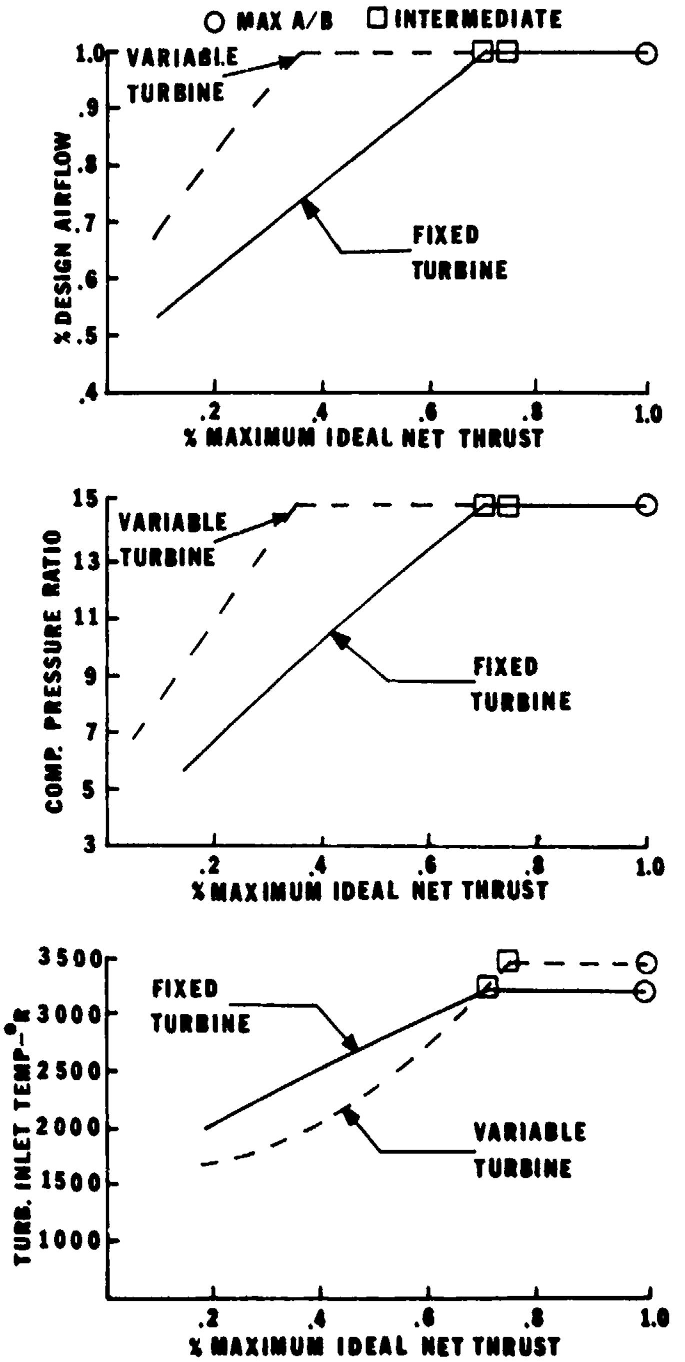


Fig.1 Part Power Operating Characteristics for Variable and Fixed Turbine Turbojets at 30,000 Ft. and .9 Mach Number

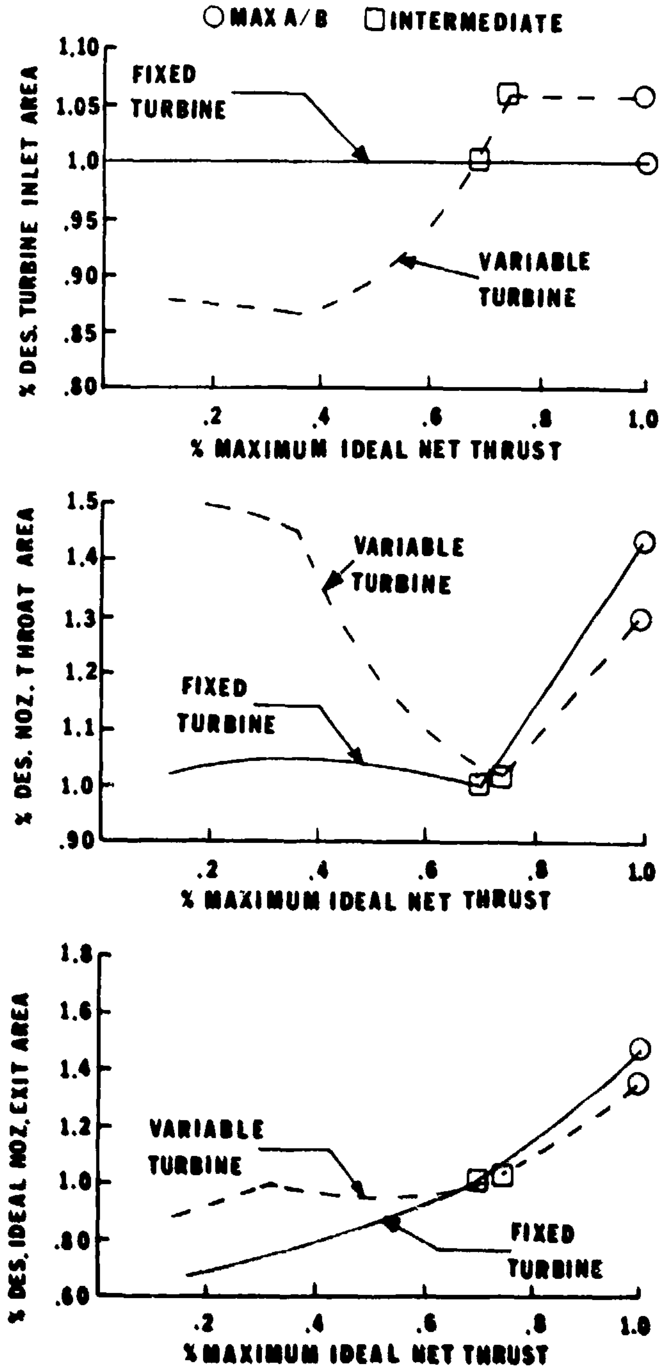


Fig.2 Flow Area Modulation for Variable and Fixed Turbine Turbojets at 30,000 Ft. Altitude and .9 Mach Number

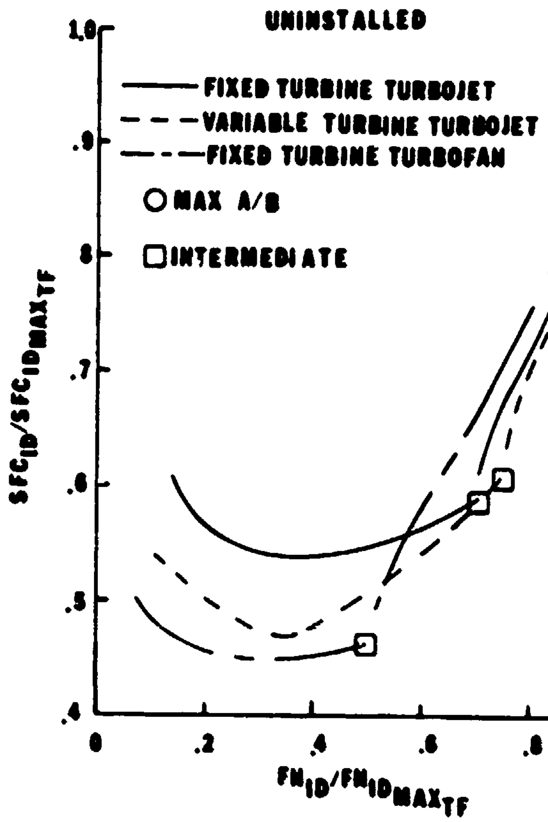


Fig.3 Uninstalled Engine Performance at 30,000 Ft and .9 Mach Number

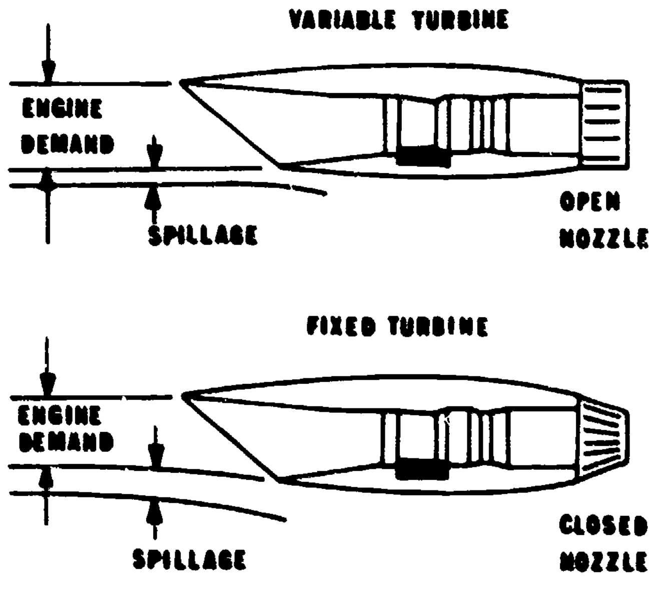


Fig.4 Schematic Representation of Engine Installation Loss Mechanisms

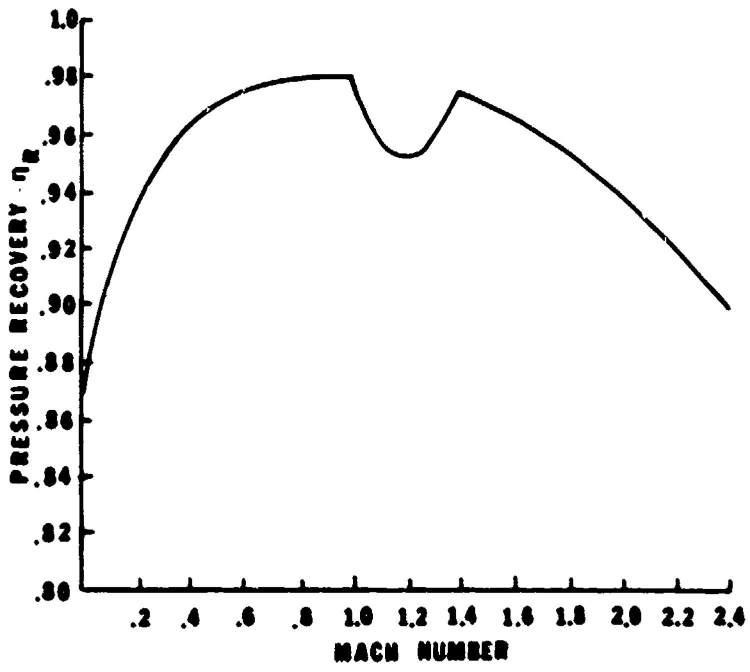


Fig.5 Pressure Recovery Trends for Typical Fighter Inlet

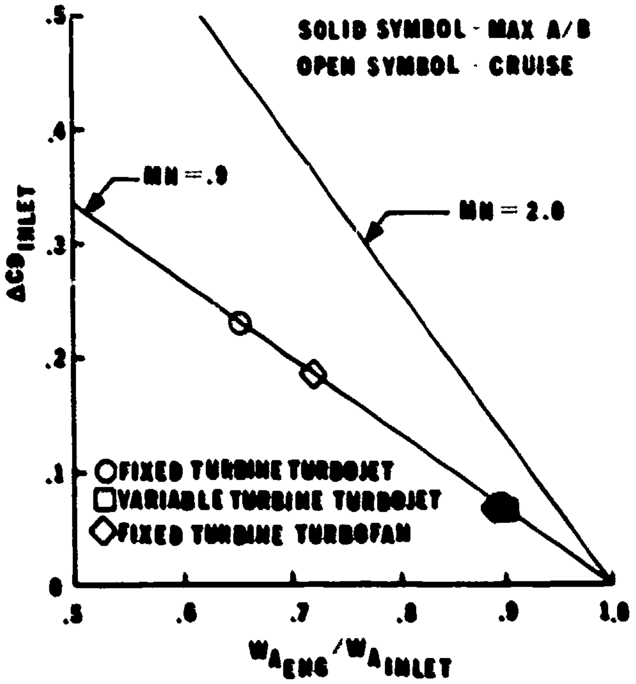
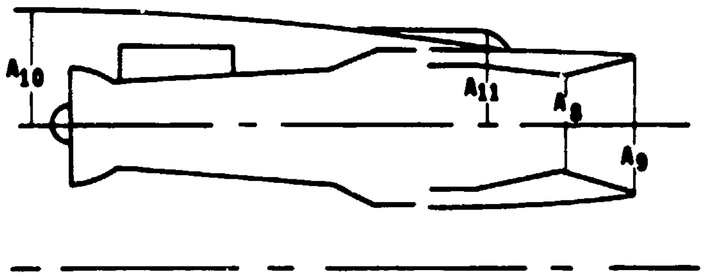


Fig.6 Throttle Dependent Drag Coefficient Trends for Typical Fighter Inlet



A_{10} - maximum fuselage cross-sectional area per engine

A_{11} - fuselage cross-sectional area at airplane connection point per engine

A_9 - exhaust nozzle exit area

A_8 - exhaust nozzle throat area

Fig.7 Afterbody Nomenclature Used In This Study

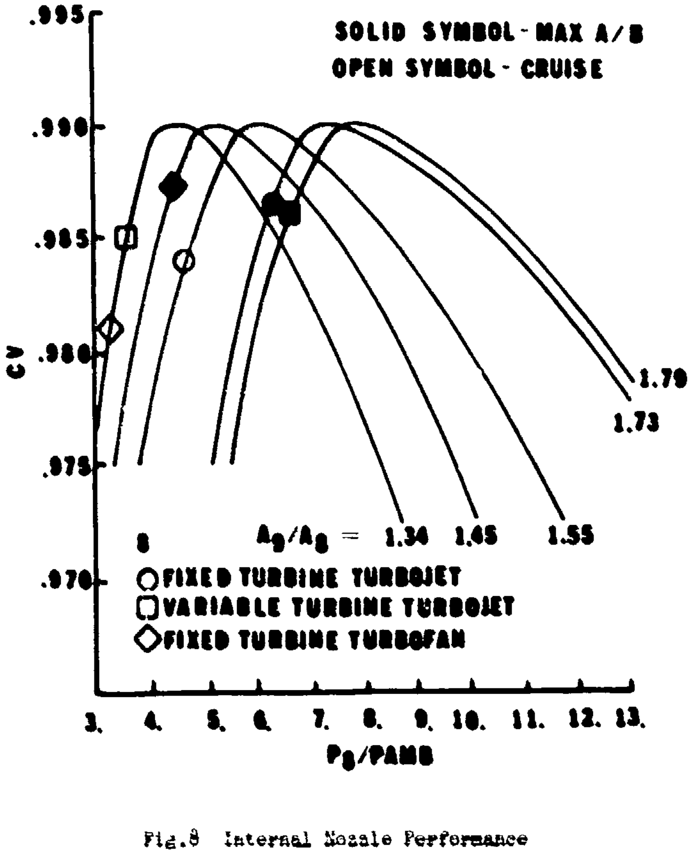


Fig.8 Internal Nozzle Performance

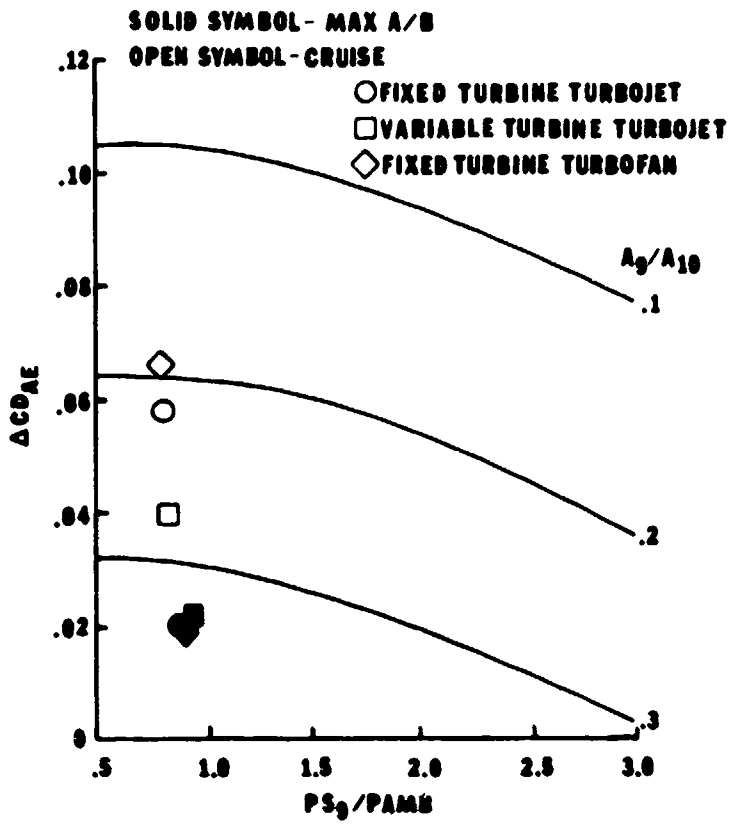


Fig.9 Throttle Dependent Aft-End Performance at .9 Mach Number

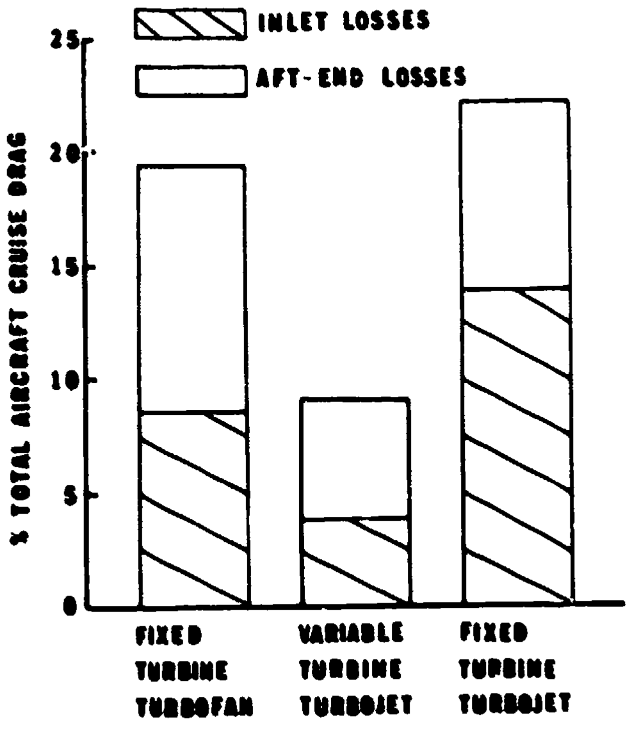


Fig.10 Subsonic Cruise Installation Loss Comparison at 30,000 Ft Altitude and .9 Mach Number

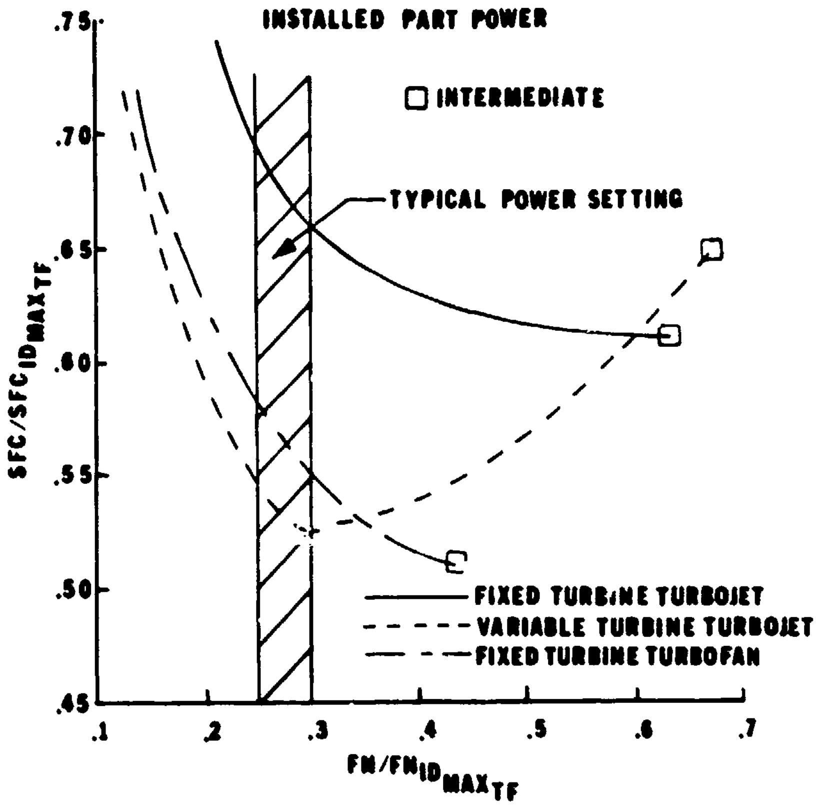


Fig.11 Installed Engine Performance at Typical Subsonic Cruise Conditions, 30,000 Ft. Altitude and .9 Mach Number

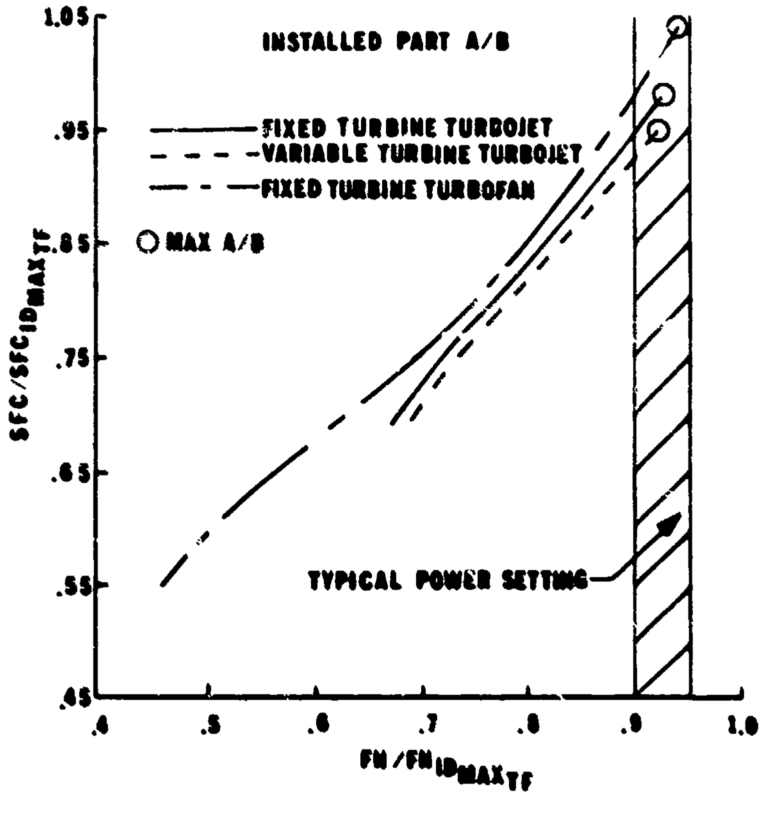


Fig.12 Installed Engine Performance at Typical Combat Conditions, 30,000 Ft. Altitude and .9 Mach Number

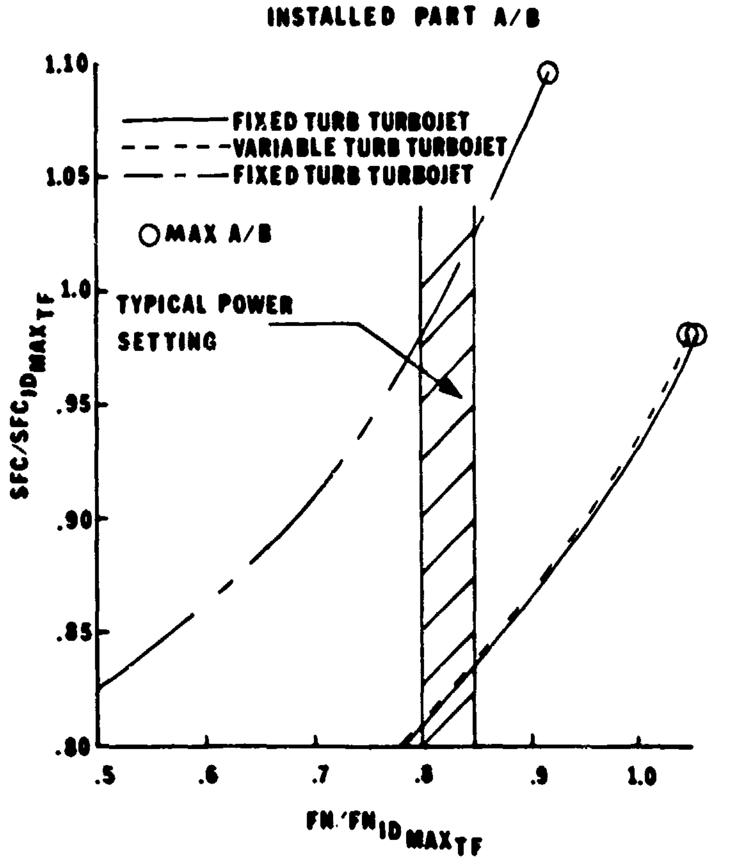


Fig.13 Installed Engine Performance at Typical Supersonic Cruise Conditions, 50,000 Ft. Altitude and 1.6 Mach Number

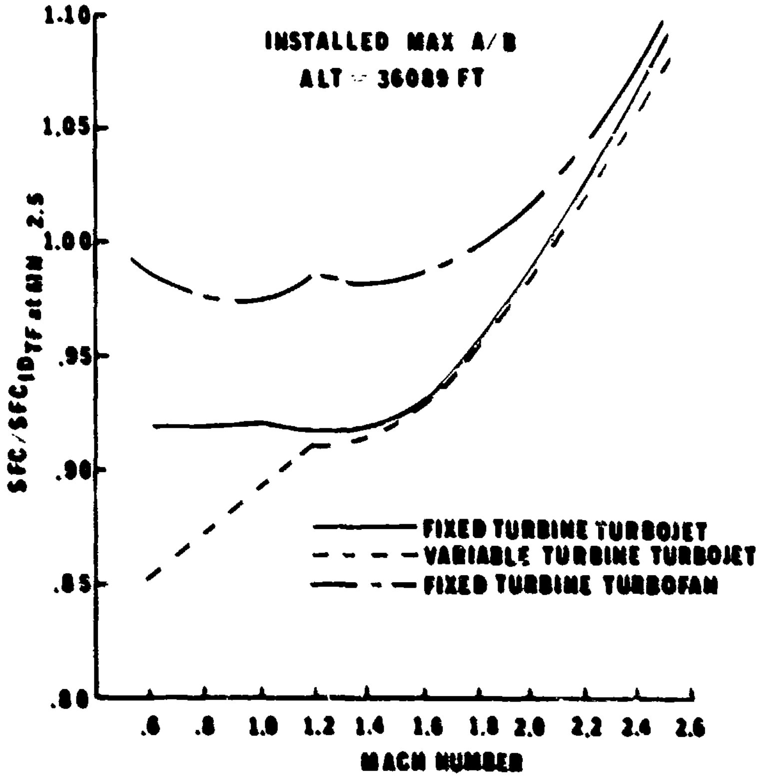


Fig.14 Installed Engine Performance for an Aircraft Acceleration at 36089 Ft Altitude

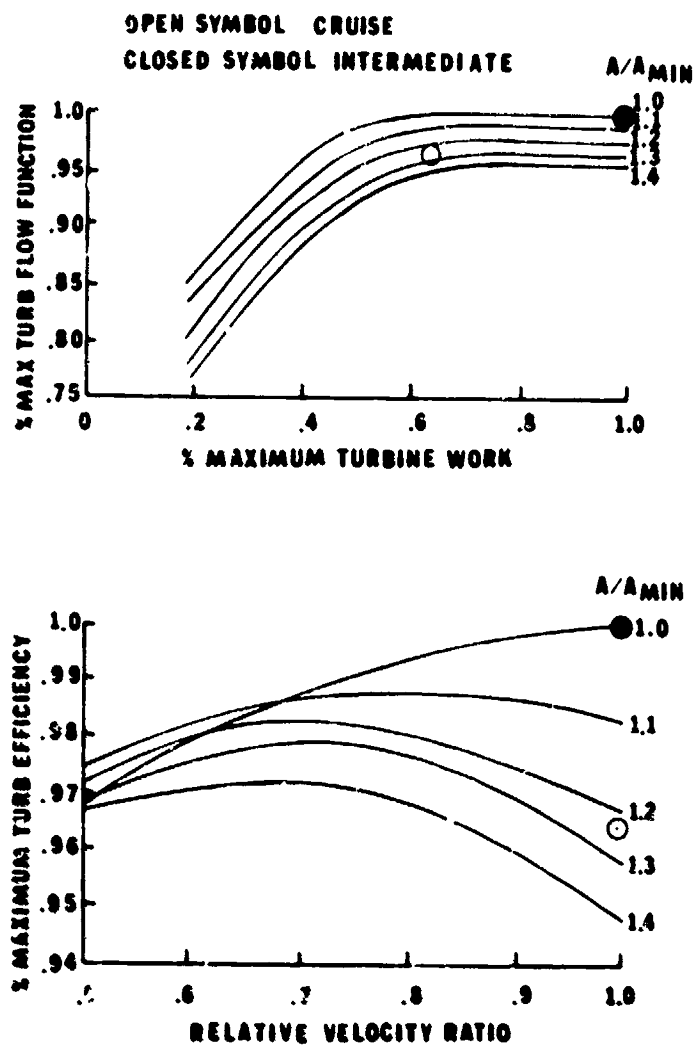


Fig.15 Variable Geometry Turbine Performance Characteristics

DISCUSSION

J.F.Chevalier

Si je me réfère à votre Fig 2, le moteur à turbine fixe qui sert de comparaison est utilisé avec une section de tuyère constante depuis la pleine poussée en moteur sec jusqu'au régime réduit. Les effets d'installation que vous avez calculés montreraient qu'il est déjà très important, avant de faire varier la turbine, de faire varier la section du col de la tuyère à régime réduit. C'est d'ailleurs ce qui est fait dans les avions tels que le Mirage III équipé d'un monoflux ATAR où l'on a optimisé justement la section du col de la tuyère pour réduire les pertes d'installation. Avez-vous fait le calcul avec un moteur dans lequel on ne faisait varier que la section du col de tuyère?

Author's Reply

Yes. That fixed turbine cycle had the nozzle throat area and the nozzle exit area varied in order to optimize thrust minus drag. However, this must be done very carefully. The engine rematches when the nozzle throat area is changed and it may move to a point of considerably poorer cycle efficiency. Changing the nozzle exit area will impact the internal nozzle performance. There is a trade then between external nozzle performance and internal nozzle performance and engine cycle performance. In this paper, each engine cycle had the nozzle areas set to obtain the optimum thrust minus drag at each flight condition.

W.Biehl

In your formula for installation effects, you considered spillage drag dependent on Mach number and airflow and recovery only dependent on Mach numbers. If recovery is considered to be also dependent on airflow, spillage and recovery have a partly compensating effect.

Author's Reply

In the preprint, I explain this in more detail, but basically yes, the pressure recovery varies with mass flow ratio. But we felt that it was probably a secondary effect. The differences in pressure recovery between the cycles we were comparing would not be that great. And it greatly simplifies the calculations if we just made the pressure recovery a function of Mach number and not also of mass flow ratio.

H.Grieb

In your figure 15 you show a turbine performance map. Is this performance map based on calculations or on measurements?

Author's Reply

Yes this is a purely analytical map, it was a predicted performance map.

VARIABLE FLOW TURBINES

by

R.J.Latimer
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Bristol Engine Group
Bristol, UK

1. Introduction

With fixed Geometry engines there is only one flow-speed relationship for the various components of the engine. Though it is sometimes possible to design the components so that this flow-speed relation is acceptable it is sometimes desirable to change this relationship in order to improve the performance of the whole engine. This may be because the performance of the individual component is unacceptable with the dictated flow-speed relationship or because the overall engine performance would be improved with a different matching between the various components.

Compressor variable geometry normally comes into the first class though variable stagger fans are an exception to this. Variable final nozzles and turbines normally come into the second class.

Turbine flow capacity can be changed in two ways either by restaggering the stator blades or by restricting the annulus area. Whilst the first method, restaggering the stator blades, allows the flow to be increased above nominal the second method is usually preferred as being mechanically simpler to introduce.

Though varying turbine flow capacity is not in general use on engines, it is often considered at the project stage and hence research into controlling turbine flow has been carried out at Rolls Royce (1971) Limited and this paper gives a survey of the results obtained.

2. Variable Stagger Blading

In order to determine the effect of varying flow capacity on turbine performance, a large range of research turbine stators have been manufactured with circular root fixings in order that the stagger of the stators may be varied.

The normal variation in flow capacity is quite small but on one turbine the flow was varied over a range of 17.5%.

Examining initially whether there is any general pattern we look at Figure 1 which is a selection of test results for five turbines covering a range of pressure ratios and loadings. It can be seen that no general pattern emerges except for the conclusion that the variation in efficiency can be large.

The results from the turbine with a large flow range (Figure 2)bear out this observation on large efficiency variation especially in the region of increased flow. It should be noticed from Figure 3 that this 17.5% variation in flow required a 26% variation in stator throat area. The variation of flow with stator throat areas was linear for reducing flow capacity, with a ratio of approximately 1:1 i.e. 14% reduction in area giving a 12.3% reduction in flow. With increasing flow area this was not the case it requiring a 12% increase in flow area for a 5.2% increase in flow with an apparent maximum flow change of 6%

3. Variable Annulus Area

There are two obvious ways of restricting the annulus area, either by mechanically introducing an obstruction into the flow or by introducing secondary air, ideally in such a way that the effective flow restriction is bigger than the amount of secondary air.

Both methods were tested, the mechanical system was simulated by attaching wedges to the casing inside the blade passages (Figure 4), the secondary air was injected through annular grooves both ahead of and behind the stator and through slots at the stator throat (Figures 5 and 6).

It can be seen that two turbines were used for these tests, an H.P. turbine with high hub/tip ratio and unflared stator and an L.P. turbine with low hub/tip ratio and a flared stator.

We will deal with the variation of flow first since this is the objective of the exercise. The wedges were an effective way of altering the flow since for wedges on the outer wall a 12% annulus blockage produced a 8% reduction in entry flow and for wedges on the inner wall a 12% annulus blockage produced a 11% reduction in entry flow.

The effect of secondary air injection is shown in figure 7 where it can be seen that upstream annulus injection and throat injection are effective in that they produce more blockage than the amount of secondary air. It can also be seen that downstream annulus injection is not very effective.

When we look at the effect of flow variation on turbine efficiency (Figure 8) the position becomes much more precise in that the only form of secondary injection that has an acceptable efficiency loss is injection at or near the stator throat, and that the efficiency loss for wedge control and throat injection is similar.

4. Discussion

If we examine all the efficiencies obtained for both turbines and include the variable stagger stator results on a similar L.P. turbine, several trends become apparent, Figure 8.

Firstly, the assumed prefered solution is the most efficient, i.e. variable stagger stators. These produce a loss in efficiency of 0.2% for every 1% change in inlet flow.

Secondly, annular slots are a very inefficient way of controlling flow, a loss of over 4% in efficiency being obtained even at very low flow changes.

Thirdly, throat injection and wedges seem to be equally efficient and after an initial deficit of 0.6% lose 0.3% in efficiency for every 1% in flow reduction.

Fourthly, there seems to be no difference in effect between the two turbines examined.

From figure 7, which is a plot of main flow variation against secondary flow, it can be seen that throat injection is a very efficient way of using secondary air a ratio of almost 5:1 being obtained, though at a high secondary air pressure ratio.

In all these tests with secondary air the efficiencies have been calculated using the total inlet enthalpy of both the main stream and the secondary air and all the flows have been corrected to standard temperature and pressure.

5. Conclusions

1. The most efficient way of controlling turbine entry flow is by stator stagger variations.

2. Flow control by restricting the stator throat either by mechanical or aerodynamic means is effective and reasonably efficient.

3. Controlling the flow by annular air injection can be ineffective and is highly inefficient.

6. Nomenclature

P Total Pressure

T Total Temperature

W Mass Flow

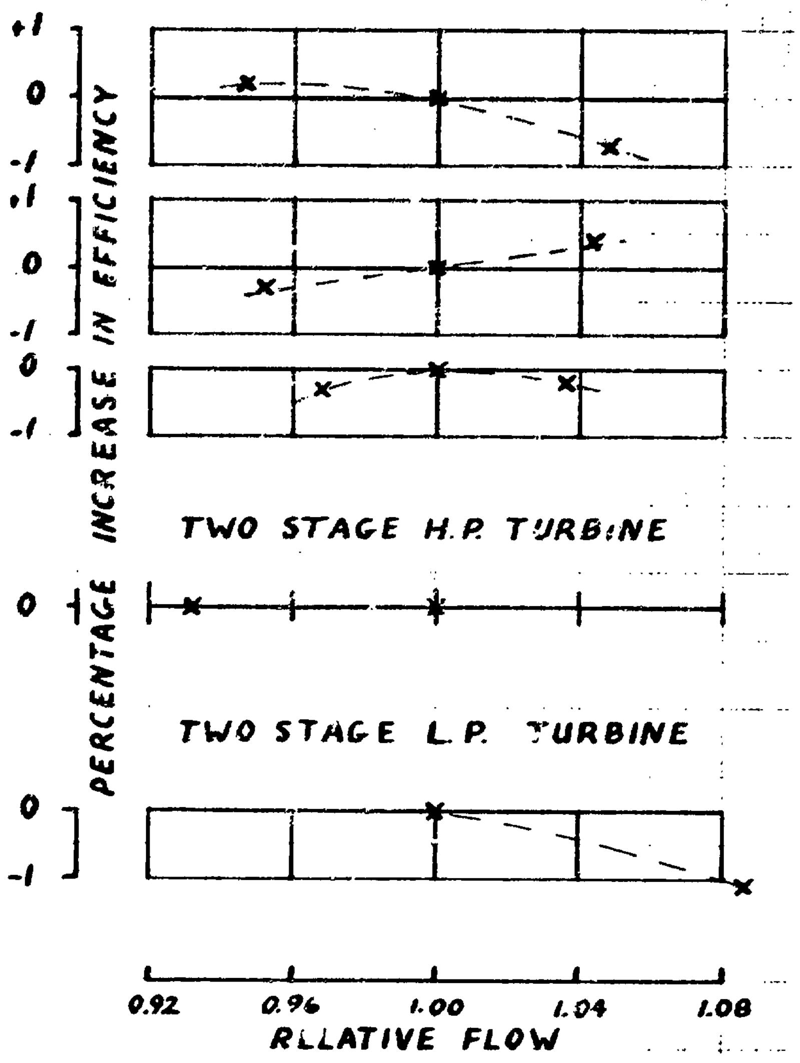
Subscripts

M Unconstrained mainstream

S Secondary

VARIABLE STAGGER STATORS

FIG. 1 EFFICIENCY VARIATION



VARIABLE STAGGER STATORS

SINGLE STAGE L.P. TURBINE

FIG. 2 EFFICIENCY VARIATION

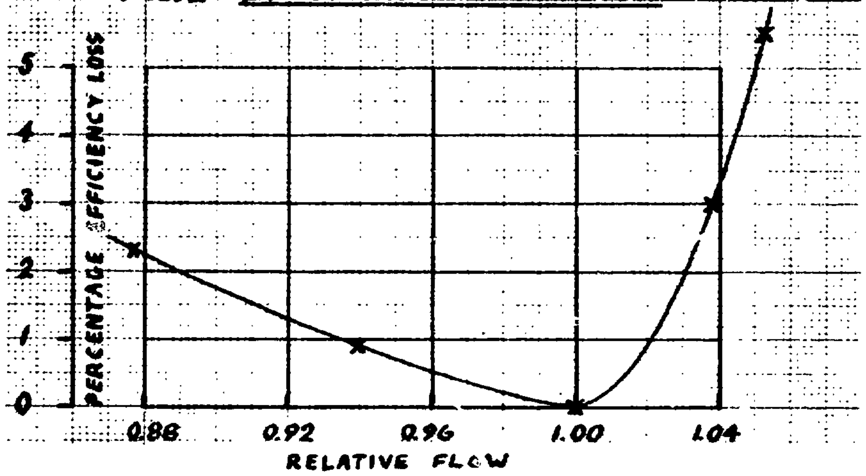


FIG. 3 FLOW VARIATION

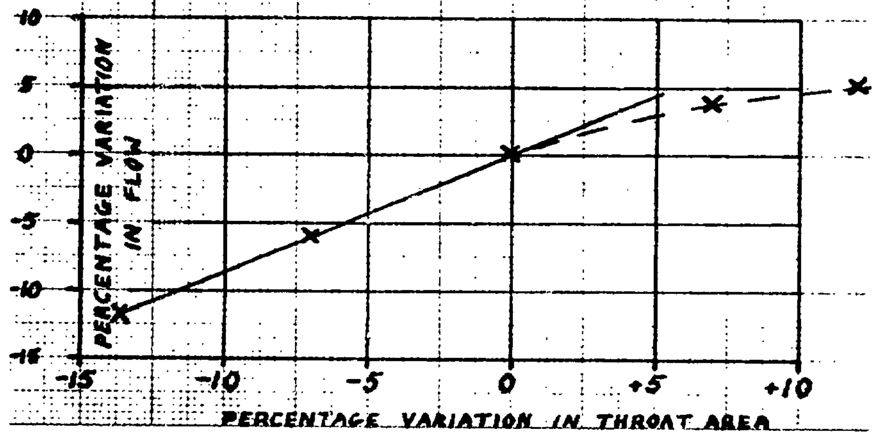


FIG.4 FLOW CONTROL BY WEDGES

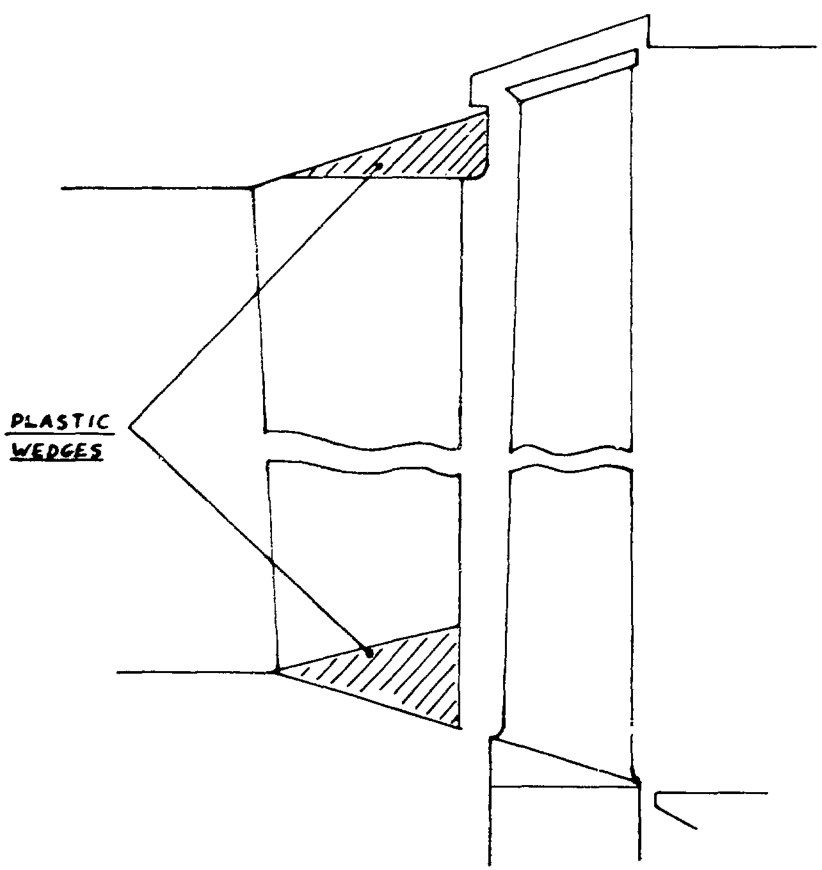


FIG. 5 H.P. TURBINE
SECONDARY AIR INJECTION POSITIONS

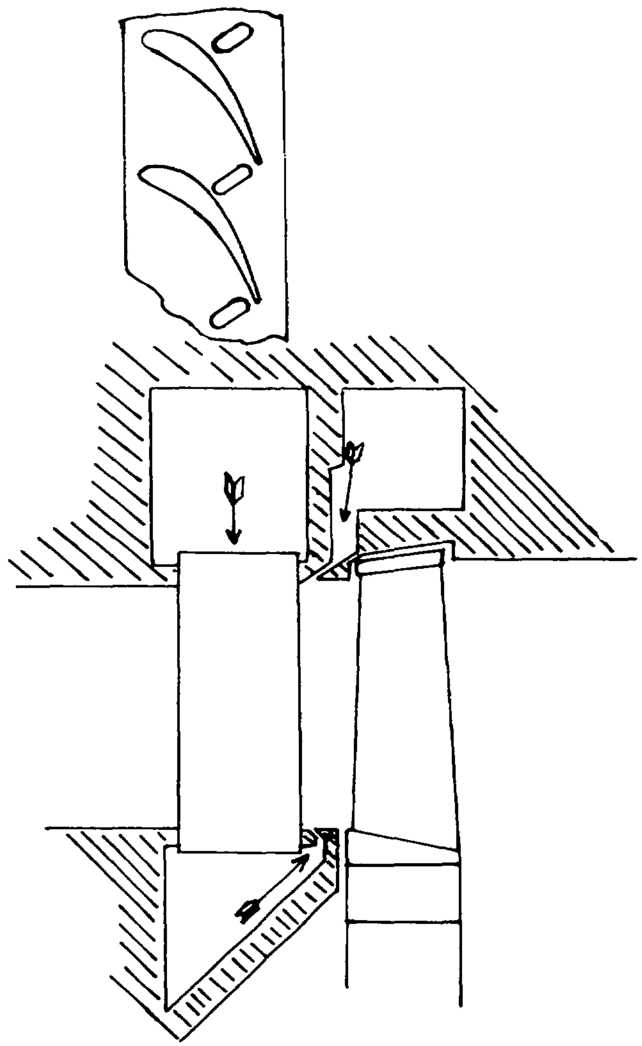


FIG. 6 L.P. TURBINE
SECONDARY AIR INJECTION POSITIONS

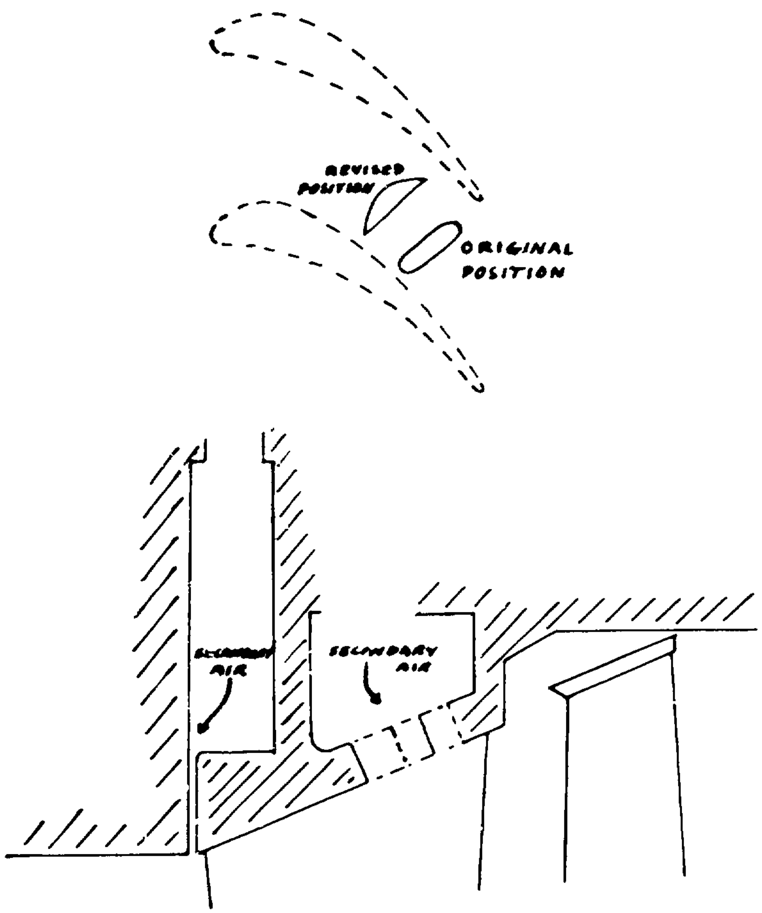


FIG. 7 SECONDARY AIR INJECTION
FLOW VARIATION

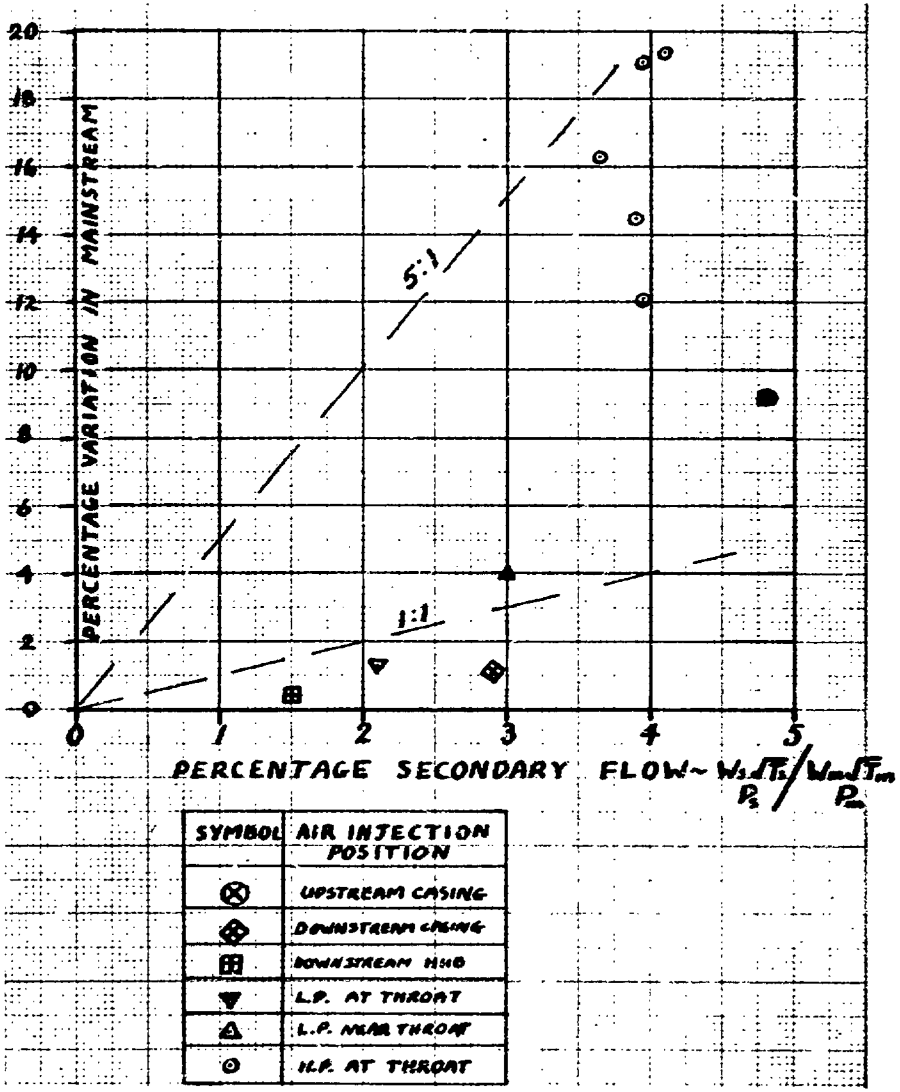
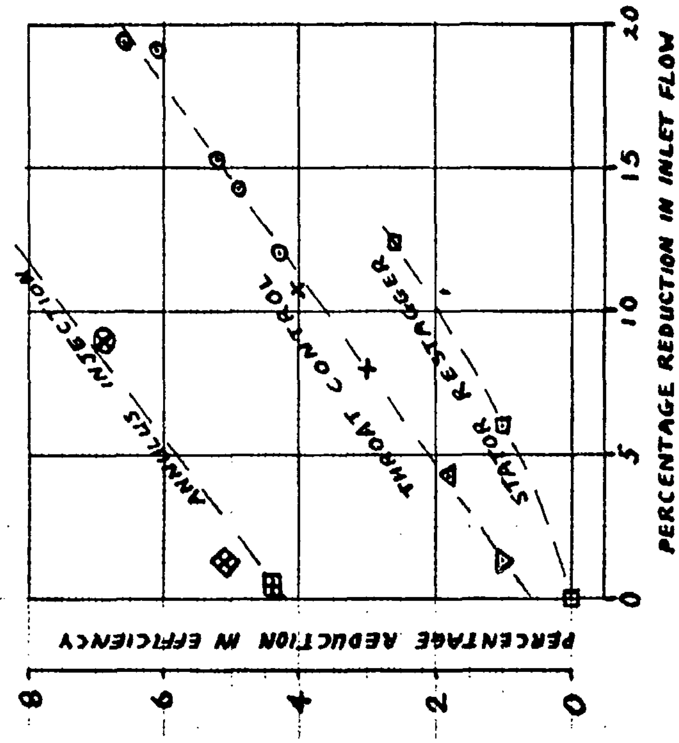


FIG.8 EFFICIENCY VARIATION



SYMBOLS AS FIGURE 7
WITH THE FOLLOWING ADDITIONS

SYMBOL	CONFIGURATION
⊡	STATOR RESTAGGER
+	WEDGES AT OUTER WALL
x	WEDGES AT INNER WALL

DISCUSSION

E.Willis

To what do you attribute the intial efficiency loss (at zero flow) for the air injection schemes?

Author's Reply

This is all due to boundary layer growth. The same as you get when putting bleed slots into a compressor. It is the presence of the slots. The slots were in fact opened at zero flow. So there was recirculation.

EXPERIENCE WITH A ONE STAGE VARIABLE GEOMETRY AXIAL TURBINE

by

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SUMMARY

Based on steady-state and transient performance characteristics of a two spool turboshaft engine, the possible improvements resulting from a variable geometry power turbine are discussed and the design requirements defined.

The aerodynamic design of a turbine fulfilling these requirements is described and the predicted power turbine characteristic is presented and compared with component test results. The design and development of the hot environment variable stator mechanism is described, including consideration of the vane positioning accuracy and response characteristics.

Performance test results and the problems encountered with the turbine operating in a full-scale engine are presented. Mechanical aspects such as the long-term endurance capability of the variable stator mechanism, as well as vibrational problems of the rotor blading caused by the variable geometry, are discussed.

SYMBOLS

f	Hz	frequency
H	J/kg K	specific work
M	kg/s	mass flow
M_F	g/s	fuel mass flow
N	1/min	rotor speed
P	kW	power
P_t	bar	total pressure
T	Nm	torque
T_t	K	total temperature
α	°	power turbine guide vane turning angle
η	%	isentropic efficiency
π_c	-	compressor pressure ratio
σ_d	N/mm^2	dynamic stress
$M\sqrt{T_t}/P_t$	kg $\sqrt{K}$/s bar	capacity

Subscripts

1	compressor inlet
3	gas generator turbine inlet
5	power turbine inlet
C	compressor
PT	power turbine

1. INTRODUCTION

Variable geometry components offer a possibility of significantly increasing the operating range of gas turbine engines and of improving fuel consumption and handling. Such improvements have already been widely achieved through the use of variable compressor stators and thrust nozzles.

With better materials technology, variable geometry in the hot parts of gas turbines appears feasible and has also been investigated in recent years (Ref. 1 to 3). The effects of variable turbines on jet engine performance were demonstrated with a modified Garrett-AiResearch TFE 731 twin spool geared fan engine (Ref. 4, 5). The development of a variable turbine advanced gas generator has been initiated with the JTDE-programme (Joint Technology Demonstrator Engine) in the United States (Ref. 6, 7).

Another promising application is the variable power turbine for two spool turboshaft engines. Such systems have been investigated primarily by manufacturers of regenerative gas turbines (Ref. 8, 9). The performance improvement in such an application is demonstrated in the compressor map of Fig. 1. At part load the fixed geometry engine operates with increased specific fuel consumption. With variable power turbine nozzles, however, the gas generator can be controlled to operate at a practically constant specific fuel consumption, for example along a constant turbine inlet temperature line.

This improvement does not apply to the non-regenerative gas turbine. The same part-load power (Fig. 2) can be selected either with variable turbine nozzles at high turbine inlet temperature and low compressor pressure ratio, or with fixed geometry at lower temperature and higher pressure ratio. In both cases, the specific fuel consumption is practically the same.

However, there are other advantages applicable to both types of engines. Increasing the power turbine nozzle area, lowers the working line of the compressor (Fig. 3) and results in higher gas generator turbine excess power during acceleration. This leads to a considerable reduction of acceleration time of the gas generator rotor. It should be noted, though, that there is a power lag of the engine during acceleration.

Varying the power turbine capacity permits a free selection of the gas generator operating point. This can be used to ensure sufficient surge margin for all operating conditions. The handling improvement obtained by this means can eliminate the necessity for variable compressor geometry or air bleeding.

Furthermore, the variable nozzle can be scheduled to improve starting characteristics, as well as idling fuel consumption at acceptable gas generator speeds.

If these advantages are to be obtained with a turboshaft engine incorporating a variable geometry power turbine, the following thermodynamic requirements must be fulfilled by the aerodynamic and mechanical design:

- Maximum change of nozzle area from closed to open (neutral) position.
- Low efficiency sensitivity to nozzle area variation.
- Variable geometry mechanism with low leakages and long life.
- Sensitive response to the control system at small vane outlet angles (closed position).

A power turbine to fulfill these requirements was designed and evaluated in rig and full-scale engine testing.

2. AERODYNAMIC DESIGN CONSIDERATIONS

A variable geometry turbine must have a range of some 60° through which the vanes can be turned, in order to meet all engine requirements stated above. This range provides continuous coverage from the smallest throat area at full load (closed nozzle) up to the neutral position with the largest throat area (opened nozzle) (Fig. 4). For aerodynamic reasons, nozzle guide vanes with two-part profiles and adjustable trailing edge segment cannot be designed to cover such a wide setting range. Undivided vanes were therefore provided for the nozzle assembly, such as have also been described by C.J. Rohnke (Ref. 9). To keep down the radial clearances through the entire turning range, the inner and outer nozzle shrouds are contoured with a spherical radius.

Changing incidence at the profile nose of the nozzle guide vanes is inevitable, owing to the vanes being turned and the flow direction upstream of the turbine changing with the engine operating point. To make efficiency insensitive to these effects, the vanes must be optimised for the most important operating range, and the profile nose must be designed so that it is insensitive to changing incidence. The same considerations apply also to the rotor blading, as the incidence is similarly affected by the setting of the nozzle guide vanes.

3. AERODYNAMIC RIG TESTING AND COMPARISON WITH PREDICTED CHARACTERISTICS

On the basis of the above mentioned considerations, a turbine was designed and its operating characteristics were investigated by means of component tests. The flow upstream of the test turbine was generated by means of inlet guide vanes, which simulated the exit flow of a turbine stage in front of the test turbine at the design point. These flow conditions were kept unchanged for all tests. Measurements were made on the turbine with the nozzle guide vanes set at the three angles $\alpha = 0^\circ$, $+7.5^\circ$ and $+17.5^\circ$. The setting $\alpha = 0^\circ$ corresponds to the design point, the positive angles represent opened-up positions of the vanes.

First of all, torque was measured as a function of the mass flow at the three turning angles with the rotor stationary and the results were compared with predicted characteristics. Fig. 5 shows good agreement between test and computed values. The torque curve for the vane angle $\alpha = -7.5^\circ$ (nozzle assembly closed beyond the design point) resulting from calculation only, was also plotted. Fig. 5 shows that torque builds up as the turning angle α decreases and the mass flow increases.

This characteristic of the variable geometry turbine can be used, for instance, to obtain an instantaneous boost of output torque with the power turbine stationary. The gas generator is run up to full speed and the variable nozzle guide vanes are brought to the neutral position. Owing to the small pressure ratio, the engine runs with low torque at a high power turbine capacity. By closing down the nozzle guide vanes and increasing the fuel flow, the gas generator can then be brought up to full load without lagging and full torque is obtained at the output shaft.

In the following series of tests, efficiencies and mass flows were measured at the three vane turning angles as a function of specific work at three different corrected speeds. Here, too, test results were compared with predictions. Fig. 6 shows efficiency as a function of specific work and vane angle for three corrected speeds. Efficiencies measured tally well with the predicted efficiency curves for all three vane turning angles. It is clear from this figure that efficiency falls off at all speeds as the vane turning angle increases. But it can also be seen that this drop in efficiency is greater at high speeds than at low speeds. Accordingly, it is an advantage to run idling range working points at low power turbine speeds and to raise speed as specific work increases.

In Fig. 7, the mass flow - likewise as a function of specific work - is plotted with the vane angle as a parameter for the three corrected speeds. As expected, the turbine mass flow increases with the vane turning towards the neutral position. Test results tally well in their trend with the predicted curves, but differences in level up to about 3 % do occur. Measured mass flow is affected less by the vane turning angle than the predicted values. This may be due to the fact that the change in mass flow caused by the change in vane angle and the associated incidence angle at the rotor blade leading edge is not sufficiently accurately covered in the prediction of the characteristic. Calculation of turbine performance at large incidence angles is namely difficult and problematical, in that flow separation may occur. The results shown in Figs. 6 and 7 agree with similar investigations (Ref. 9 to 11).

4. DESIGN AND DEVELOPMENT OF THE ACTUATION MECHANISM

Fig. 4 shows the design scheme of the adjustable nozzle guide vanes in the flow duct. The bearings for the adjustable vanes are on the outer perimeter of the flow duct. Five struts support the inner duct wall. As mentioned before, the outer and inner contours of the flow channel have a spherical configuration in the turning zone of the vanes, so that the top and bottom clearances between the vanes and the flow duct do not alter when the vanes are turned (see also Ref. 9).

The bearings for the vane shafts were optimised in an extensive development programme. They were first tried out on two test stands, in which 3 or 6 vanes with their bearings were set up. Each vane was turned by a windscreen wiper motor with a frequency of 60 1/min. Weights were used to simulate the operating load on the bearings.

Electrical resistance heating or a hot-gas stream from a combustion chamber was used to generate the operating temperature. In the latter case, it was possible to vary the temperature by intermittent lighting up and shutting off of the combustion chamber. This cycle lasted 6 minutes, the gas temperature being changed between 323 and 1123 K. The temperature in the vane shaft changed from 373 to 923 K.

Three of the versions tested are shown in Fig. 8. In the case of version 1, it was assumed that the shaft had to be cooled. For this purpose, cooling air was fed to the centre of the shaft and extracted on both sides throttled via labyrinth chambers. The shaft and the sleeves were coated. The test temperature was kept steady at 1073 K. The test had to be broken off after 147 hours. The anti-friction coatings on all parts had been ruined, the labyrinth fins had made deep scores in the sleeves and consequently the bearing clearance had become unacceptably large.

In version 2, a cylindrical vane shaft was used. Various materials for the vanes and sleeves were investigated. In all cases, the bearings jammed after several temperature changes. They only functioned satisfactorily with a very large bearing clearance, which resulted in an unacceptably large gas leakage.

A substantial improvement was obtained when the vane shaft in version 3 was mounted in two sleeves made of ceramic material. In these sleeves, vanes with uncoated shafts were tested at a steady temperature of 1123 K for 1200 hours and at changing temperature for 3000 hours. In operation in the engine, these bearings have so far logged 1800 hours.

5. PERFORMANCE AND MECHANICAL EVALUATION IN ENGINE OPERATION

The effects mentioned in the introduction of a variable geometry power turbine on performance were confirmed in the engine test. In addition, several secondary effects were ascertained. Variable nozzle guide vanes are the easiest conceivable means of compensating for any thermodynamic mismatching of the components. In the case of the fixed geometry engine, the entire nozzle assembly would have to be changed.

Compared with the component test, a loss of 1 % to 2 % in efficiency was analysed. It was put down to the different inlet flow angle and the bigger clearances in the engine. There were no thermodynamic problems worth mentioning associated with the setting accuracy of the vane turning angle, either in steady-state or transient operation.

Two mechanisms to actuate the variable nozzles were tried out with the engine in operation. In the design shown in Fig. 9, the vane actuating levers are connected to an actuation ring via ball-and-socket joints. The actuation ring must have axial freedom in operation, so that it can follow the circular movement of the actuating lever. In conjunction with the relatively large play in the actuating system, the actuation ring can tilt in the axial direction. This causes incorrect positioning of the variable vanes around the perimeter. The actuation ring is moved by two hydraulic cylinders, as shown schematically. This largely keeps radial forces away from the bearing of the actuation ring.

The vane pivot point is well to the rear, providing the advantage that the trailing edge of the vane swings only slightly when the vane is turned and there is only a minor change in the gap between the vane and rotor cascades. With this arrangement, the gas forces exert a continuous torque on the turning shaft, providing play-free following of the vanes when they are turned.

In the second design shown in Fig. 9, the actuation ring is guided in a wire-race ball bearing. The vanes are turned by toothed segments. Differing positions of the vanes around the perimeter, owing to the axial freedom of the actuation ring, as in the case of design 1, are no longer possible.

6. VIBRATION PROBLEMS

In what follows, characteristic phenomena in the vibration behaviour of the turbine blades resulting from the variable nozzle guide vanes are described on the basis of interesting results obtained from measuring blade vibration on the single-stage power turbine. The aspects dealt with are the sensitivity of the turbine blades to interferring objects in the flow path, the extension of the resonance ranges resulting from the many possible operating conditions, and the variability of the axial spacing between nozzle guide vane and rotor blade and its influence on the amplitude of resonance stresses.

The turbine blade investigated in full-scale engine operation showed unexpectedly high stress amplitudes of the first flap mode where it is excited by the 8th engine order (Fig. 10). The design of the engine provides no simple explanation for this effect. The flow upstream of the turbine is disturbed by the support struts, the pressure and the temperature probes shown in Fig. 11. A harmonic analysis of these interferring objects indicated that the phenomenon mentioned is to be attributed mainly to the distribution of the temperature probes, although they are fitted at a relatively great distance from the turbine. The distribution of the probes and the results of the harmonic analysis calculations are shown in Fig. 12 for two different arrangements of the thermocouples. It can be seen that removing two of the temperature probes should halve the excitation intensity of the 8th engine order.

Blade vibration with both arrangements of the thermocouples was measured under identical operating conditions.

The resonance stresses measured with two strain gauges at different positions on the blade surface are also shown in Fig. 12 as a function of the guide vane position. Test results do not agree with the analysis when the turbine guide vanes are closed. However, there is a significant reduction in the stresses when the vanes are in the neutral position. This phenomenon is easily interpreted, if it is borne in mind that flow is accelerated by the closed nozzle assembly, disturbances being reduced in the process, whereas the open nozzle assembly scarcely affects the velocity profile. This means, as a general consequence, that increased attention must be paid, right from the design stage, to the arrangement of probes and support struts as potential exciters of vibration. In the specific case described, a better distribution of the thermocouples was recommended as a provisional measure to reduce dynamic stresses.

Another aspect of variable guide vane geometry in relation to the vibration behaviour of the turbine is the extension of the scatter range of the speed at which resonance occurs. There is scatter in the natural frequencies of the blades of all turbomachinery, owing to dimensional deviations of the blade profiles from their ideal shape. Differing operating conditions, which result in a change in blade temperature, extend this scatter range still more. In a fixed geometry engine, the temperature levels in the power turbine are determined by the gas generator speed and the ambient conditions. In the variable geometry turbine, the same effect can be produced independently of the gas generator speed, by changing power turbine guide vane throat area (Fig. 13). It follows that when the dynamic properties of the rotor blades of variable geometry turbines are assessed, greater attention must be given to the scatter in frequencies and speeds at which resonance occurs.

A further special feature of variable nozzle guide vane geometry is the changing spacing between vane trailing edges and rotor blade leading edges (Fig. 11). Naturally, this effect depends on the point selected to pivot the guide vanes. The trailing edge of the guide vane can be considered a good pivot point vibration-wise, involving a spacing to the rotor blades which does not vary. However, as many aspects must be considered in designing a variable nozzle assembly, the best solution appears to be the compromise presented in Fig. 9. As resonance with the guide vane excitation order can certainly be expected in the low speed range, it should be ensured, irrespective of the guide vane pivot point chosen, that the minimum axial spacing obtained when the nozzle assembly is opened is not less than an empirical value of about 30 % of the guide vane chord length. This should substantially reduce the excitation intensity of the guide vane wake (Ref. 12).

Fig. 14 shows the resonance stresses of the guide vane excitation order as a function of the gas generator speed for two different positions of the guide vanes. The result is entirely consistent with previous findings on the effect of the spacing between nozzle guide vane and rotor blade on the amplitude of resonance stresses to be expected (Ref. 13).

A general consequence of ensuring a minimum spacing between nozzle assemblies and rotor cascades is a somewhat greater overall length of variable geometry engines. In the specific case investigated, on the one hand the axial spacing was enlarged by shifting the rotor and, on the other hand, the range through which the guide vanes can be turned was restricted to a minimum compatible with function.

7. CONCLUSIONS

The investigations carried out have confirmed that a variable geometry power turbine can increase the flexibility of a turboshaft engine and improve its performance. Turning of the nozzle guide vanes demonstrated the expected efficiency drop at off-design settings. However, this drop is less than 3 % for a turning angle of approx. 15°. Optimum efficiency for all vane turning angles shifts to smaller aerodynamic loadings as speed decreases. Particular care is needed in setting the clearances in the engine.

The actuation mechanism was developed to a stage ensuring high accuracy of settings and a considerable service life. It is evident that current technology can cope with a hot environment up to 1150 K. Engines with variable geometry nozzle guide vanes react more sensitively to interferring objects in the flow duct and their resonance speeds also have wider scatter. The resonance stress level of the rotor blades is influenced by the turning of the nozzle guide vanes and the resulting change in the axial spacing between nozzle guide vanes and rotor.

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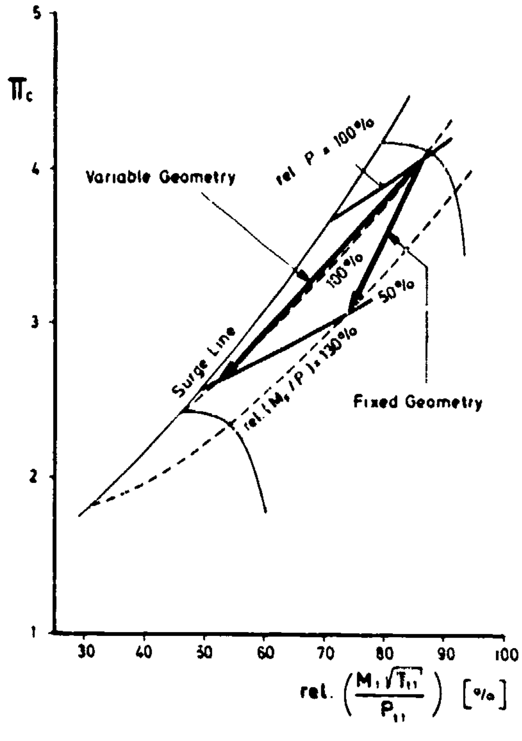


Fig. 1: Variable power turbine effects on regenerative engine performance

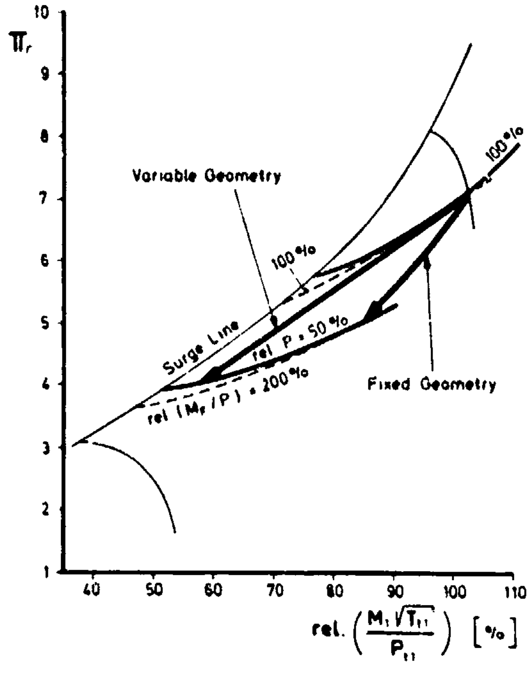


Fig. 2: Variable power turbine effects on non-regenerative engine performance

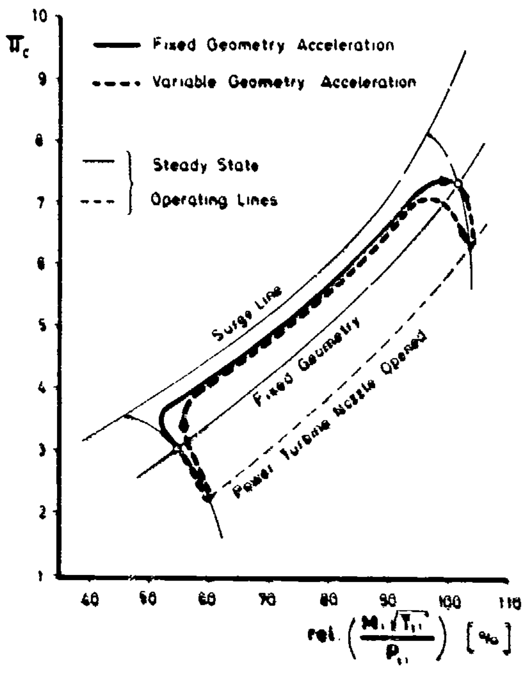


Fig. 3: Effects of variable power turbine on gas generator acceleration

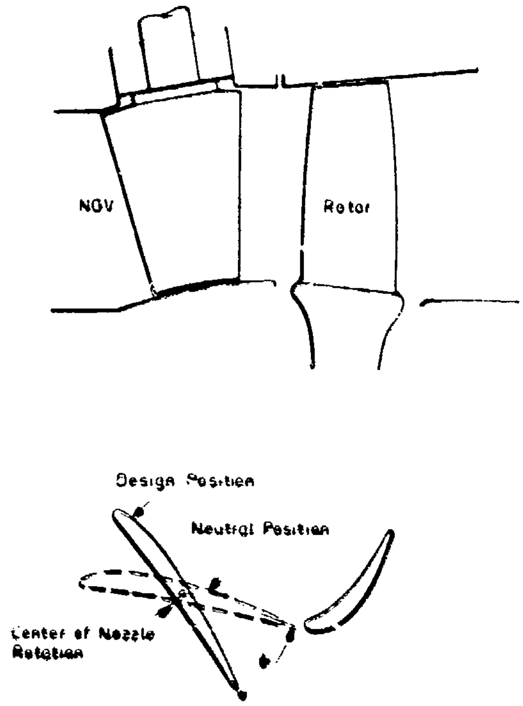


Fig. 4: Variable nozzle scheme

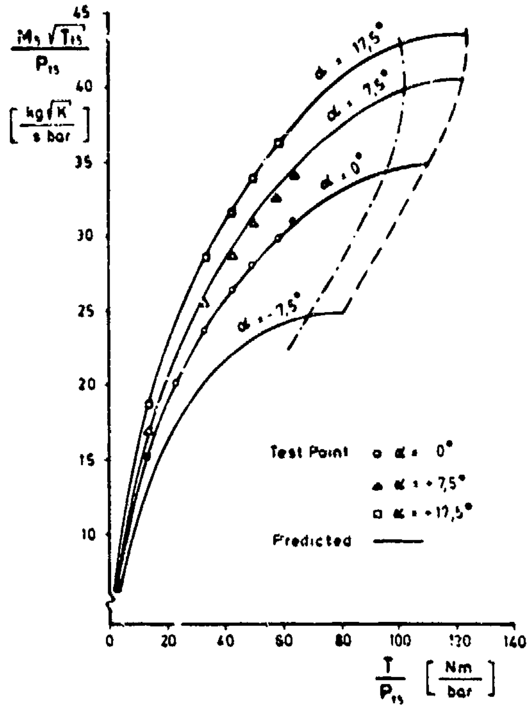


Fig. 5: Zero speed torque as affected by the guide vane turning angle

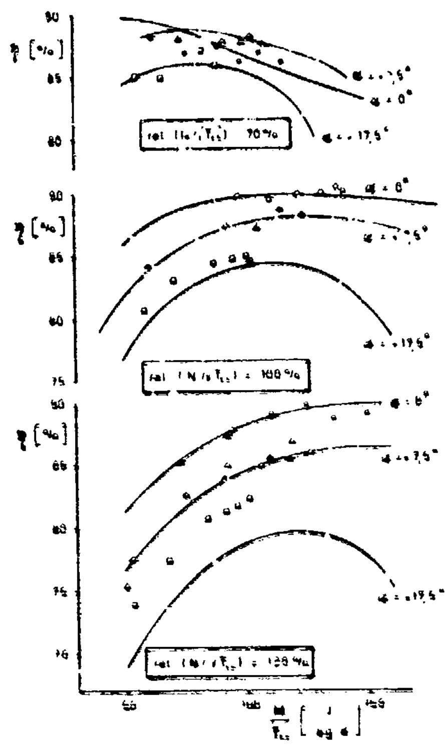


Fig. 6: Predicted and measured efficiency characteristics of the variable geometry turbine

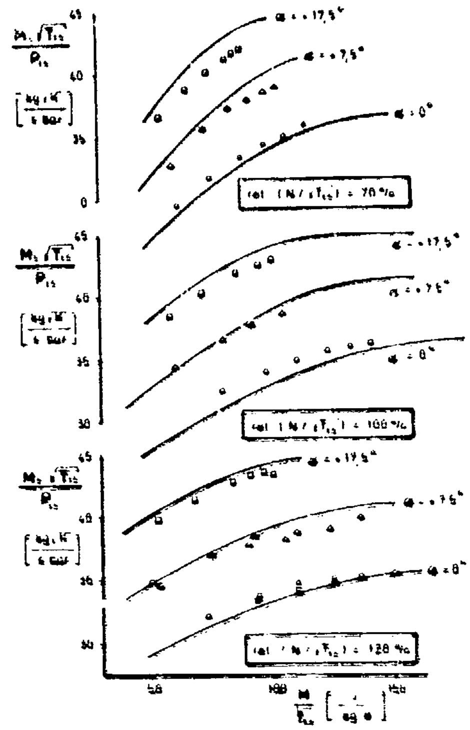
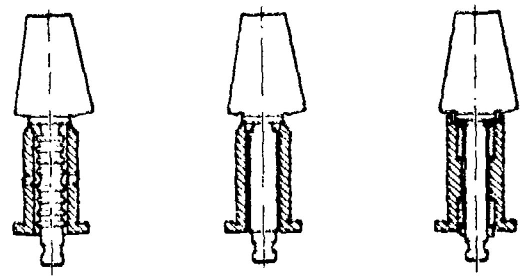


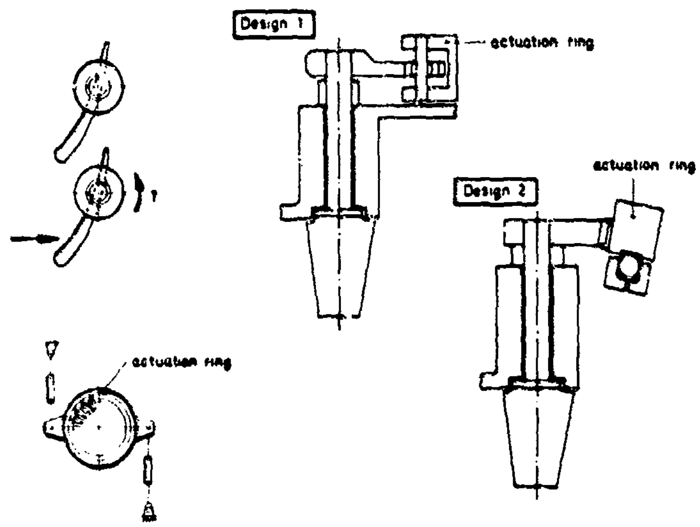
Fig. 7: Predicted and measured mass flow characteristics of the variable geometry turbine



Test Temp.	1073 K	1073 K	1173 K
Test Duration	147 h	120 h	static thermoshock engine 1200h 3000h 1800h
Test Results	strong sleeve abrasion coating destroyed large bearing clearance	bearing jammed too large bearing clearance necessary	only minor problems

Fig. 8: Guide vane bearing concepts

Fig. 9: Actuation mechanism schemes



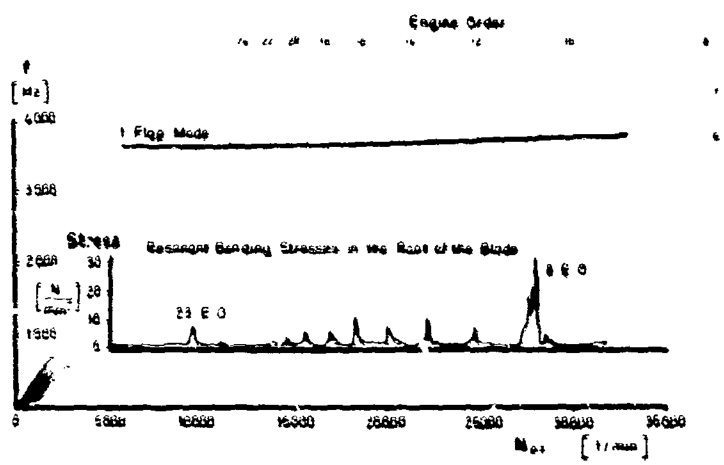


Fig. 10: Vibration interference diagram and dynamic response of the rotor blade

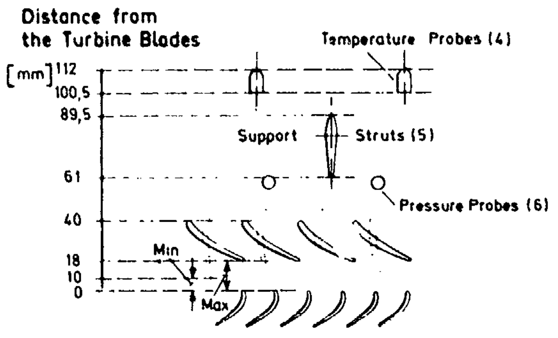


Fig. 11: Objects interferring with the flow upstream of the turbine

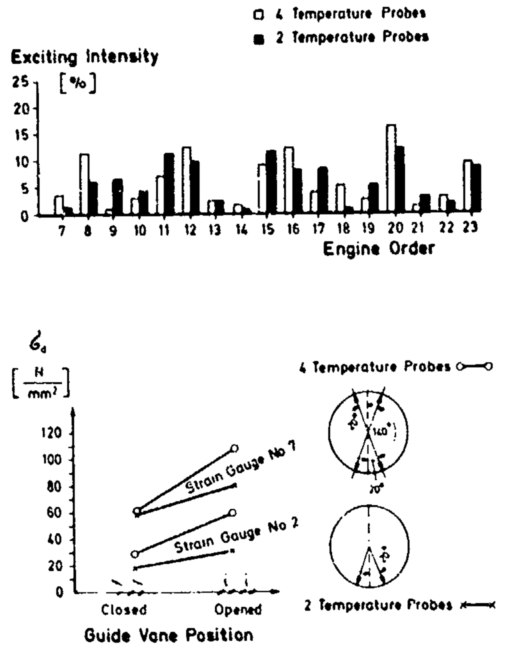


Fig. 12: Spectrum of the harmonic excitation caused by disturbances in the flow, and measured first flap mode dynamic stresses corresponding to the 8th harmonic

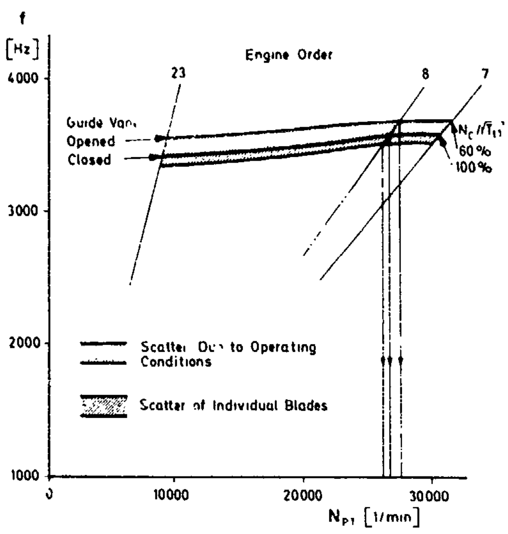


Fig. 13: Scatter of first flap resonances, as caused by operating conditions and vibration properties of the individual blades

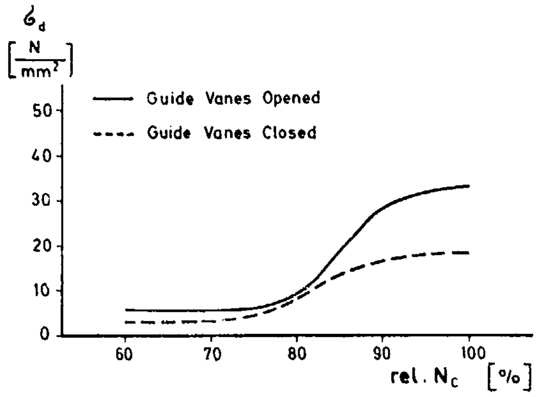


Fig. 14: Measured resonant bending stresses of the first flap mode as affected by guide vane setting and gas generator speed

DISCUSSION

P.A. Whiteman

The variable nozzle guide vane has been designed to have a pivot position to the rear of the blade to provide a continuous load on the mechanism. Have you considered what happens to the NGV in the event of failure of the driving mechanism?

Author's Reply

This depends on the failure position. If the failure is close to the NGV this would result in the NGV closing. This may be detrimental to the engine. Ideally the NGV's should be brought to a position, where the engine can produce shaft power without being overheated.

INTEGRATED PROPULSION CONTROL SYSTEM FOR FIGHTER AIRCRAFT

by

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SUMMARY

An Integrated Propulsion Control System (IPCS) has been designed, fabricated and flight tested on a F-111 aircraft. The IPCS combined the functions of the traditionally separate inlet and engine controls into one computational unit. This integrated approach has proven to be a most effective way to utilize the full potential of the propulsion components to achieve a high performance aircraft.

The program utilized the combined talents and special resources of the Air Force Aero-Propulsion Laboratory and the NASA Lewis and Dryden Research Centers to provide program management and altitude and flight testing. In the contracted portion of the effort, the Boeing Aerospace Company provided a unique tripartite approach that formed the IPCS team with engineering personnel from aircraft, propulsion and controls companies.

GLOSSARY

IPCS	Integrated Propulsion Control System
NASA	National Aeronautics and Space Administration
LeRC	Lewis Research Center
DFRC	Dryden Flight Research Center
USAF	United States Air Force
DPCU	Digital Propulsion Control Unit
PSU	Power Supply Unit
DCU	Digital Computer Unit
IFU	Interface Unit
HMC	Bill-of-Material Hydromechanical Control
BOMDIG	Digital Replica of Bill-of-Material Control

INTRODUCTION

Historically, the primary means of increasing engine performance has been through improvements in component efficiencies and increased cycle temperatures and pressures. Now that the limits of these methods are being approached, the primary means for enhancing overall airplane mission performance will be the use of variability in the engine cycle itself. Although there are many engine concepts under discussion at the present time, it is reasonably certain that the number of controlled variables will be increased when compared to existing operational engines. The trend in this direction has significant implications for the controls engineer. Previously, emphasis has been placed on improvements in accuracy and control logic to ensure that the engine was operating near peak efficiency and within safe limits. With the coming generation of engines, these goals will not be relaxed. However, additional criteria will be used to judge the adequacy of the propulsion control system. The primary control function will ultimately become that of continuous regulation of the variable geometry of the propulsion system in response to such specific airplane requirements and minimum fuel consumption or maximize thrust (Ref 1). This is not a trivial task, especially since it will require the involvement of controls engineers in propulsion and aircraft performance studies at a level far beyond customary practice and implies rather large increases in control computing power with the attendant difficulties with software management.

IPCS BACKGROUND

Major tasks in the development of control capability to meet future needs are (1) the demonstration of the use of the digital computer for total control, (2) the combined control of the entire propulsion system - supersonic inlet, engine and afterburner (Ref 2) and (3) a demonstration of the effectiveness of inter-disciplinary teams of controls engineers in solving the propulsion control problem. The desirability of accomplishing these tasks become apparent in the 1960's. However, it was not until the early 1970's that advances in digital electronics made this approach feasible, and the planning of a program to accomplish these task begun.

During this period of time the goals of the National Aeronautics and Space Administration (NASA) paralleled to those of the United States Air Force (USAF) in the desire to investigate the application of digital electronics to turbine engine controls. Consequently, a joint Integrated Propulsion Control System (IPCS) was established. The major goals of this IPCS program were:

(1) The demonstration of full authority digital control of a highly complex system,

(2) The demonstration of combined control of an inlet, engine and exhaust nozzle, and

(3) The demonstration of a methodology that would permit rapid and effective solutions to complex integration problems.

Since the achievement of inlet-engine compatibility in a supersonic aircraft is a highly complex subject not completely amenable to computer or wind tunnel simulation, it was further decided to provide a flight test which would demonstrate that the IPCS hardware and control modes were capable of functioning as designed in a realistic environment. The test vehicle selected for this demonstration was a F-111 equipped with TF-30 engines. The propulsion system in this aircraft required the control of eight (8) variables (Fig 1). These variables were: (1) inlet spike travel, (2) inlet cone travel, (3) 7th stage bleeds, (4) 12th stage bleeds, (5) gas generator fuel flow, (6) afterburner duct fuel flow, (7) afterburner core fuel flow, and (8) nozzle area.

Once the goals of the program were established in detail a division of responsibilities between the Government Agencies was established. The Air Force Aero-Propulsion Laboratory, in addition to having program management responsibility, would provide the control modes, software, control hardware, bench test and sea-level tests by means of exploratory development contract; NASA Lewis Research Center (NASA LeRC) would provide the simulated altitude testing; and NASA Dryden Flight Research Center (NASA DFRC) the flight testing. Each organization participated in the technical direction of the program by means of frequent meetings, phone calls, reports and letters.

THE IPCS PROGRAM

The contractual portion of the IPCS program was initiated in March 1973 with the Boeing Aerospace Co. as prime contractor and the Pratt & Whitney Aircraft Division of United Technologies Corporation and Honeywell, Inc. G&AP Division as principle subcontractor. Key features of this contractual effort were a three-phase approach that emphasized thorough, sequential testing, and a tripartite structure that stressed communication and utilized intercorporate engineering teams (Fig 2). Although the program details are discussed in Reference 1 and 4, a short outline is provided below.

Phase I (Ref 4, Ref 6) consisted primarily of analytical efforts. Initially, F-111/TF-30 data was gathered from all available sources. This data was then used to check and trim a combined propulsion system simulation of the inlet, engine and control. Verification of simulation accuracy at this point in the program permitted the analytical design of the control to proceed with a high level of confidence. Analog simulation of the propulsion system was also developed from this data, and used in the development of the digital control software. In addition, there was Phase I activity in the areas of (1) preliminary control design, (2) failure mode analysis, and (3) component selection.

In Phase II (Ref 1, Ref 5) the control hardware design and component fabrication was completed as were the TF30 engine modifications. Also, the software packages were developed, finalized and validated. Considerable use of the propulsion system digital simulation was made in this software effort.

In addition, two extensive sets of testing were accomplished. The first of these, flight assurance testing, was performed to ensure that the hardware was rugged enough for use in the aircraft and that flight safety would be maintained (Ref 7). The second group of tests was a thorough step-by-step evaluation of the functionality of both the hardware and software. This latter process began with software checkout on the digital simulation and hardware checkout on bench tests. The hardware and software were than joined and tested in a closed-loop bench test that utilized a real-time engine hybrid simulation (Ref 5). Only after completing this extensive test sequence was the IPCS used to control the TF-30 engine in a sea-level-static test which constituted the beginning of the Phase III effort.

In Phase III, the engine and control were shipped to NASA LeRC for software evaluation and development in their Propulsion Systems Laboratory altitude facility. At LeRC, NASA employees were responsible for total test operations, data instrumentation and recording. Contractor personnel were responsible for the controls-related equipment, software changes, and maintenance of software configuration control. Essentially, the same relationships were maintained during the flight testing at NASA DFRC (Ref 9).

Considerable intercorporate and interagency coordination was required to ensure smooth program flow. In response to this need, Boeing utilized their tripartite team approach to materially contribute to the success of the program (Ref 4). On the Government's side, the importance of the program was well recognized by upper level management and adequate facility priority was made available to ensure that no significant slippages occurred.

Once the basic facility scheduling was established the details of the testing programs were resolved. To accomplish this, there was extensive interaction between the contractor team and the responsible NASA LeRC and DFRC organizations responsible for the overall altitude and flight test sequences. Generally, a priority list of test conditions and points was established by the contractor team. This list was then reviewed with the testing organization to produce a plan that permitted efficient test operation while still observing the requested priorities to a high degree.

Many innovative control modes were investigated and flight tested in the IPCS program (Ref 8). These include direct distortion measurement and propulsion stability control, buzz detection and recovery, compressor exit Mach number surge protection, direct afterburner fuel/air ratio scheduling, airflow bias to match the engine to the inlet, and nozzle area control of the fan operating line. In order to accomplish an effort of this magnitude in 36 months, a highly flexible digital control system was utilized.

CONTROL HARDWARE

The basic control package was named the "digital propulsion control unit" (DPCU) and consisted of a power supply unit (PSU), the digital computer unit (DCU) and the interface unit (IFU). Due to cost constraints, these units were neither miniaturized nor hardened for engine mounting as is standard practice in the USA. Instead, they were mounted in the aircraft weapons bay and air cooled.

The computer was a 16-bit machine with a 16K core memory, a 1.2 microsecond add/subtract time and a 10.8 microsecond multiply time. The 16-bit word length was sufficient to maintain adequate precision for most of the computations, however, in some cases, double precision arithmetic was required. In converting the sensor data for use by the computer, twelve (12)-bit words were found to be adequate to maintain system accuracy.

The remainder of the flight hardware (Figs 3 & 4) consisted of (1) a computer monitor unit mounted in the cockpit, (2) inlet and engine controls that were specifically modified to allow complete computer authority, and (3) two transducer packages -- one mounted in the wheel well to service the inlet requirements and the other mounted on the engine (Ref 9).

This flight hardware was supported by a set of ground support equipment that consisted of (1) a teletype, (2) a high speed paper tape punch and reader, (3) a static simulation of the engine which provided simulated signal sources to the IFU, (4) a load simulated which provided loads for IFU output, and (5) digital-to-analog channels capable of driving analog recorders. This system proved invaluable in the development as it permitted significant levels of hardware and software checkout without the removal of the control from the aircraft and without the running of the engine.

CONTROLS SOFTWARE

Although the computer was capable of compiling software written in the FORTRAN computer language, assembler language was used exclusively in order to meet the 30 millisecond cycle time requirements of the control system. In total, two (2) complete sets of control software were generated (Ref 6).

The first software package consisted of a digital replica (BOMDIG) of the bill-of-material analog hydromechanical control modes This control was designed so that an evaluation could be made of the IPCS hardware without the added complexity of new control modes. The simulation accuracy of this software was quite good and is illustrated in Figure 5 where is is compared with the hydromechanical control (HMC) for an acceleration and deceleration. Initial flights were made with this software package installed.

The IPCS software, on the other hand, was designed to evaluate a number of innovative control modes. These modes in various combinations took advantage of: (1) the computational power of the system, (2) a high level of inlet-engine control integration, (3) nonstandard sensing schemes and (4) the use of computed airflow derived from a stored fan map to set fan operating line, trim inlet geometry and fan speed, and set afterburner fuel to air ratio.

A description of the entire set of control laws would be too lengthly to present here, however, two of the loops will be described in more detail. These are the control of the gas generator fuel flow and the exhaust nozzle area during supersonic flight. The combination of these two variables was used to provide inlet airflow matching at a favorable fan operating point.

In the production system an air data computer signal representing aircraft Mach number is sent to the main fuel control where it trims gas generator fuel flow and thereby engine speed and airflow. A schedule of a ratio of intercompressor pressure to turbine discharge pressure versus corrected gas generator speed provides the basis for setting exhaust nozzle area and therefore indirectly setting a fan operating point.

This mode has been successfully flown for several years, but is less than ideal since it is sensitive to minor cycle rematching and horsepower and bleed extraction. The consequences of this sensitivity is the routine requirement for field trimming while the engine is in afterburner operation, which of course, produced undesirable manhour and fuel expenditures and exhaust noise.

The mode tested in this program scheduled the fan operating line airflow directly as a function of overall engine pressure ratio and the loop was closed with computed fan airflow (Figure 6). The difference between these two airflows - airflow error - drove an integrator which produced an exhaust area trim signal. This signal was then added to a base area request to produce the final area request. Base area was established as a simple function of throttle angle. With this mode, fan operating line was held to close tolerances and rapid transient response maintained.

The computed airflow was also utilized to adjust the engine to match inlet requirements. In this mode a reference capture area was computed based on inlet local pressure ratio and compared to actual capture area ratio derived from computed corrected airflow and inlet geometry. The resulting error was passed through a deadband to reduce noise sensitivity, clamped to assure limited authority, and inhibited for subsonic operation. It was then applied as a correction to the fuel flow command signal, changed engine speed and, therefore, airflow. The combination of these two loops was designed to produce precise inlet airflow matching while maintaining the fan operating point near peak efficiency. It also eliminated the need for field trimming of the afterburner.

FLIGHT TEST EXPERIENCE

The comparison of these two control modes began at the airplane trim pad. With the necessity for retrimming the conventional control mode in the field, realistic trim tolerances have to be established due to control hysteresis and due to the physical endurance capabilities of the trim crew working in close proximity to an engine operating in afterburner power (Fig 7). With the IPCS nozzle area system, the afterburner trim would be performed under the same trying circumstances, but it would only be performed once -- when the control is mated to the engine and the aircraft.

In flight the nozzle area mode consistently held fan operating points to a closer tolerance than did the BOMDIG control (Fig 8). Although airflow is a particularly hard parameter to measure, we estimated that the fan operating point was held to within $\pm 1\%$. This improvement can also be inferred indirectly from the transient response of the afterburner system. The IPCS mode consistently utilized less fan surge margin than the BOMDIG control, and in fact, produced surge-free transients at extreme flight conditions where the use of the BOMDIG control mode often resulted in surge.

Performance of the airflow bias loop was in general satisfactory, in that a high degree of airflow matching was obtained. However, when the inlet geometry was deliberately run far off schedule at various flight conditions in order to study loop dynamics, the transient response of engine speed indicated that gain compensation would be required. For this application, burner pressure would not be suitable since it does not vary in the required manner. Engine pressure ratio is a promising parameter for gain compensation but the flight schedule did not permit a thorough evaluation.

Other significant results from the flight test were (1) improved engine acceleration times at almost all flight conditions - the one slower acceleration occurred as the result of a turbine inlet temperature loop, (2) increased supersonic rate of climb, and (3) increased supersonic range.

The flight test portion of the program proved to be a valuable investment since some additional control development was required as a result of flight test data, and the successful accomplishment of these changes proved that the control concepts developed on simulations and in altitude cell tests were sound and could be adapted to accommodate aircraft installation effects. Furthermore the combined inlet-engine control features can now be incorporated in future production systems with significantly reduced risk.

APPLICABILITY TO FUTURE ENGINES

Since the trend toward higher performance propulsion systems is continuing and is generally accompanied by increasing levels of engine complexity (Fig 9) the requirements for more sophisticated control logic and more accurate hardware will continue. These requirements will dictate the use of digital electronic controls in an increasing number of future high performance aircraft.

The IPCS program demonstrated a number of items that are expected to appear on fighter aircraft engines developed in the 1980's time period.

The first of these items was the use of a full authority digital computer itself. Although the speed and memory size of the computer can be considered large by today's standards, the history and projections of semi-conductor electronics indicate that the size and cost of computers will continue to decrease and become relatively minor considerations in the next decade. Consequently the IPCS provided a realistic demonstration of benefits from the use of the computational power expected in future control systems.

The second feature is the effective manipulation of a many variable system. As the number of variables increases, the control complexity grows at a far higher rate due to the interactions between the variables. These interactions must be accounted for in the control system in order to produce the levels of performance expected from the variable engine cycles. In the IPCS program a control for an eight variable propulsion system, with some rather strong interactions, was successfully analyzed, developed and tested.

Each of these features were significant in themselves; however, the true benefit of the program was their integration into a system that was able to enhance propulsion system and aircraft performance. This integration process occurred at all levels in the design process. Aerodynamically, fan-corrected airflow which is a key parameter in judging propulsion system performance was employed as a unifying parameter that involved inlet geometry, gas generator and afterburner fuel flow and nozzle area. With respect to the hardware, sensor accuracy and the precision of analog to digital conversion and computation were all selected to meet system needs. In the software area a modularized approach was taken and the modes and logic were designed to take advantage of the logic capability of digital computers and were not in the case of the IPCS software mere digital representations of an analog control (Ref 6).

While these factors could possibly be considered logical or "good pract. ", they are often difficult to achieve in practice. The tripartite concept used in the IPCS program did much to enhance communication and permitted a broad base of technology input to the system design. With this approach, teams of working-level engineers from the aircraft, engine, and controls companies were co-located for two-week periods at critical points in the program and charged with the task of working out detailed control system design and integration problems. Final decision-making authority still remained with the appropriate authority; however, these decisions were made with rather complete knowledge of the inpact on the total system.

There is still a considerable amount of work to be accomplished in preparing for advanced engines, particularly in the area of algorithm structure; however, the IPCS techniques provide a sound starting point for these advanced efforts.

CONCLUSIONS

1. The use of a full authority digital computer is an effective means of controlling a complex propulsion system. Furthermore, a decision to utilize a computer for purely performance purposes will generally result in the additional benefits of control self-checking and some degree of failure tolerance due to the inclusion of additional software.

2. Aircraft performance can be enhanced when overall propulsion system needs are factored into the detailed control made design. In this way the goals to be met by the aircraft are taken into account when control involving detailed component performance are formulated. This approach proved highly effective in the IPCS program and resulted in improved engine acceleration times, increased supersonic range and climb, and lower flight idle speeds.

3. Intercorporate teams of control designers can promote improved propulsion system and aircraft integration. Co-location of working level engineers and their integration into a single design team resulted in early management visibility as to the system implication of control detailed design.

4. Control modes that do not require field trim can be developed. The benefits of this feature are several: (1) improved system performance, since the loss of accuracy due to human element is minimized, (2) reduced unproductive engine operating time which produces an increase in flying hours per overhaul, (3) reduced fuel usage, and (4) reduced pollution and noise.

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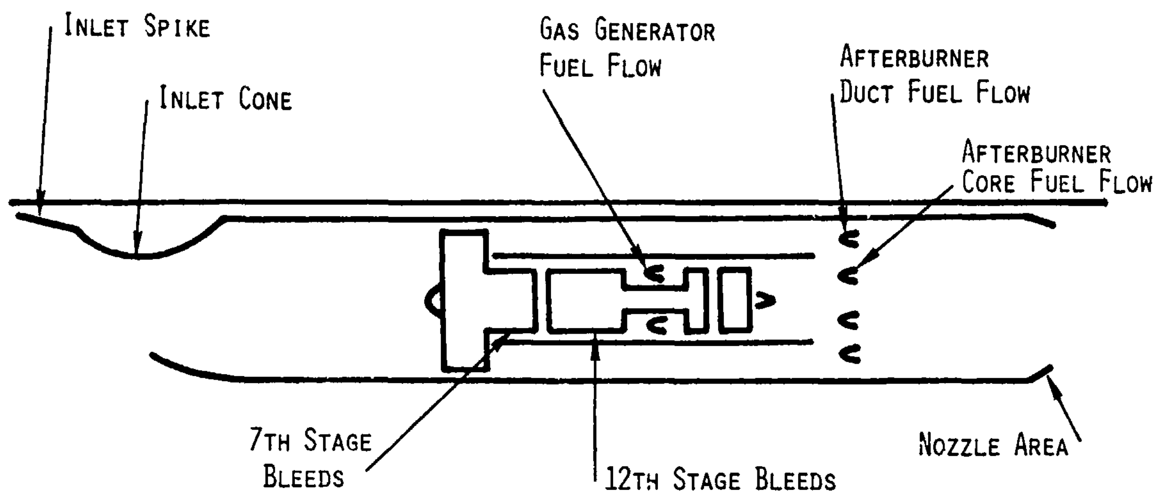
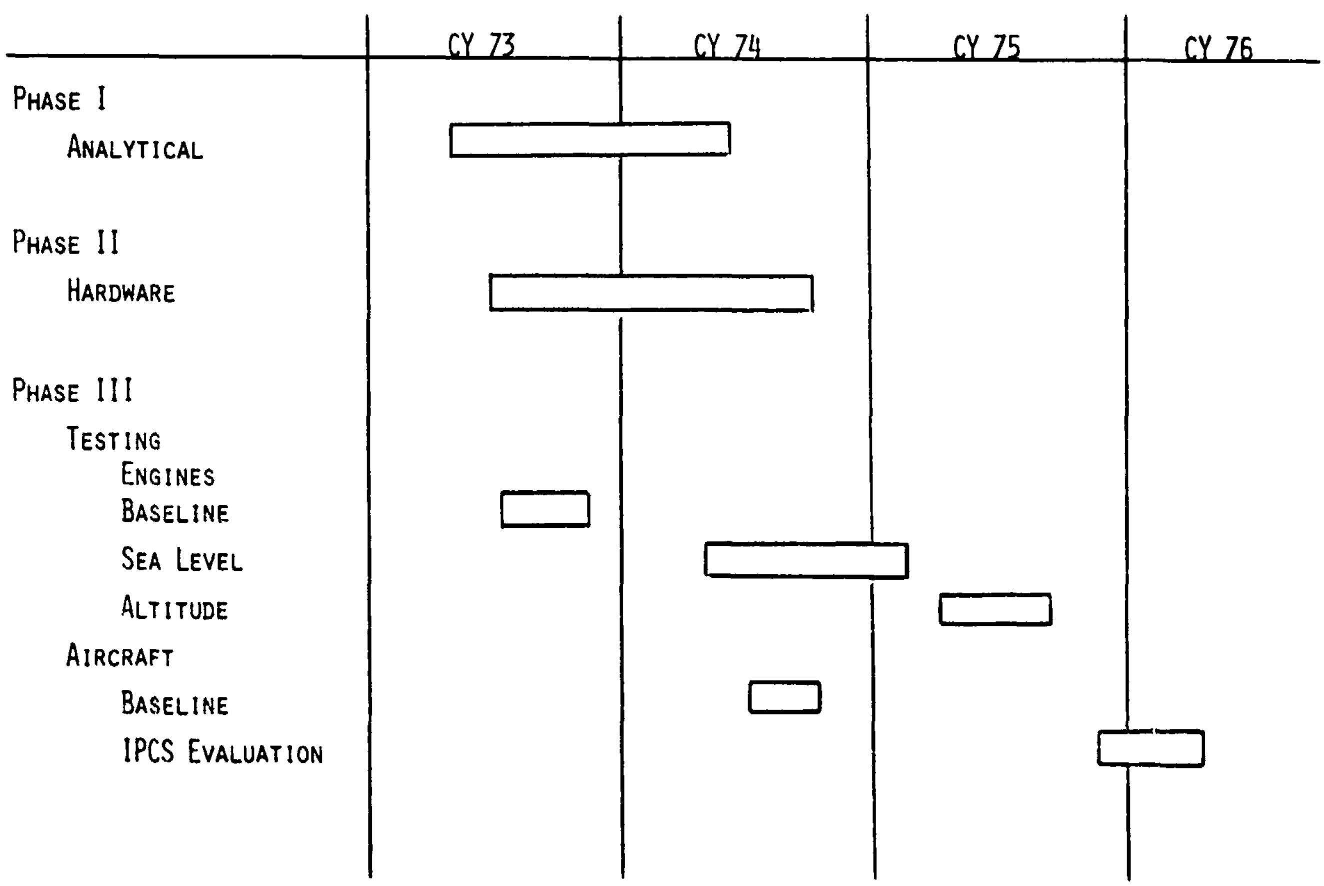
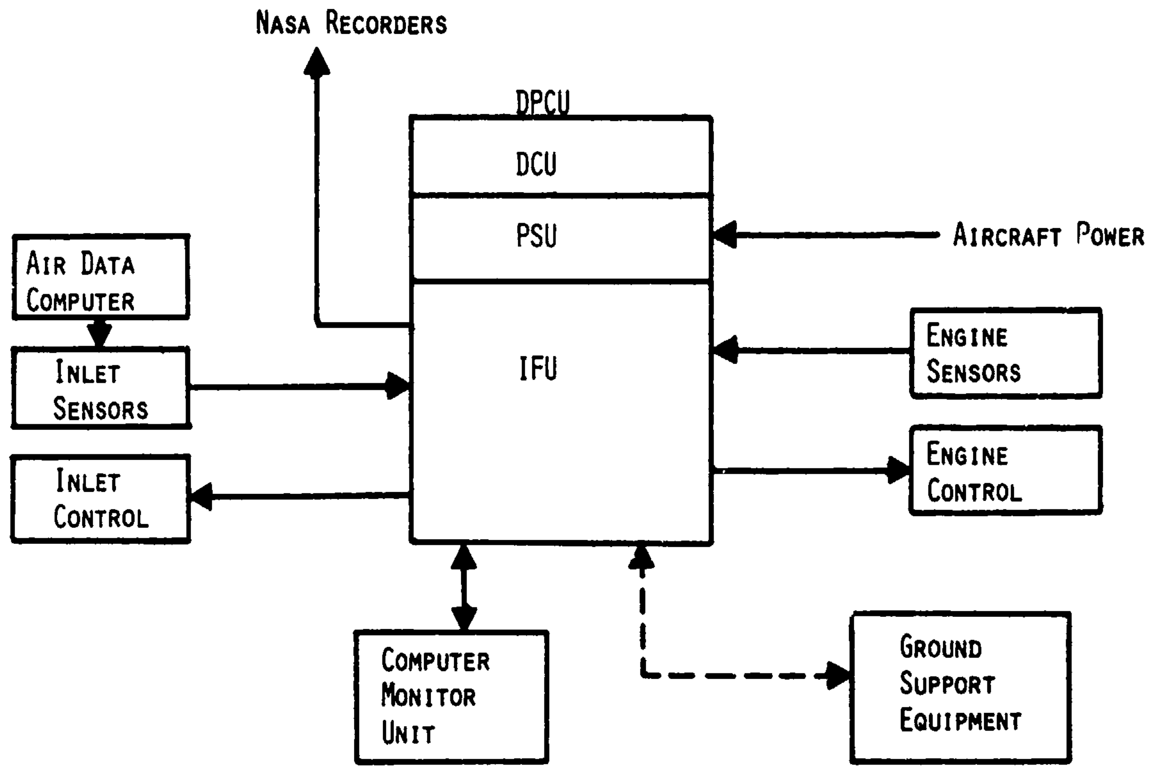


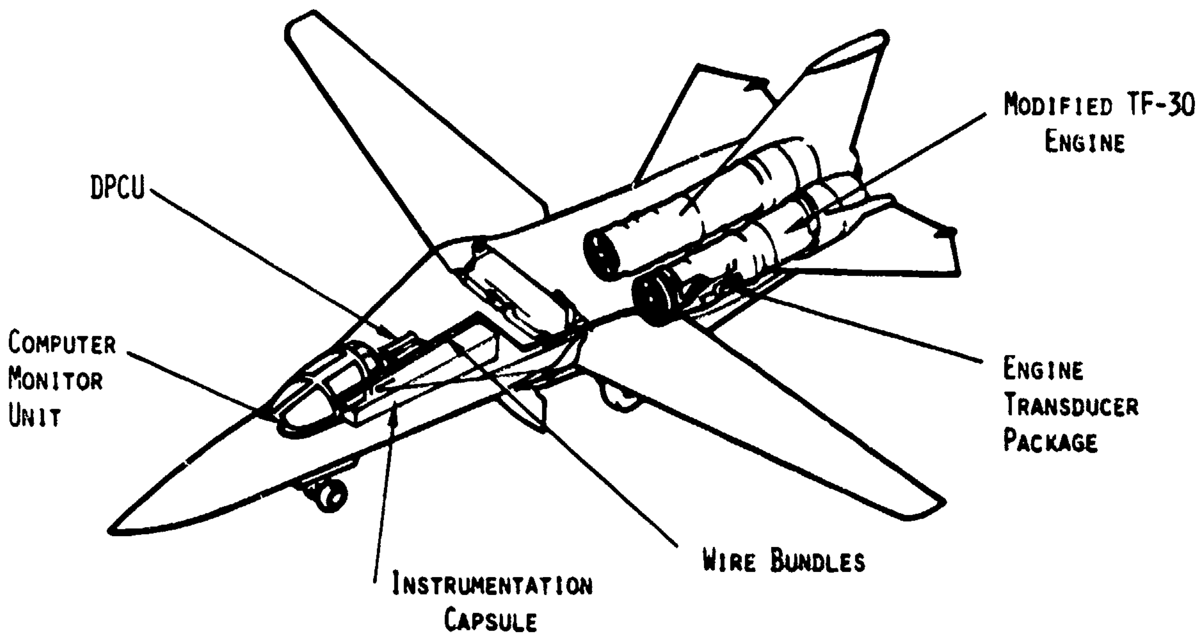
The IPCS Propulsion Variables
(Fig 1)



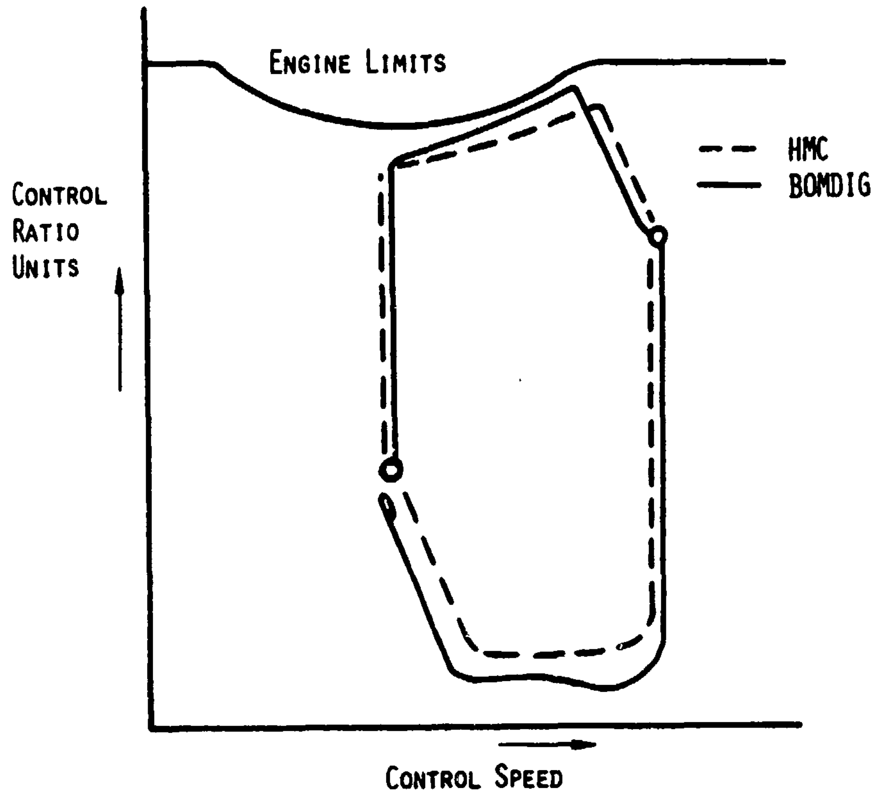
IPCS Program Schedule
(Fig 2)



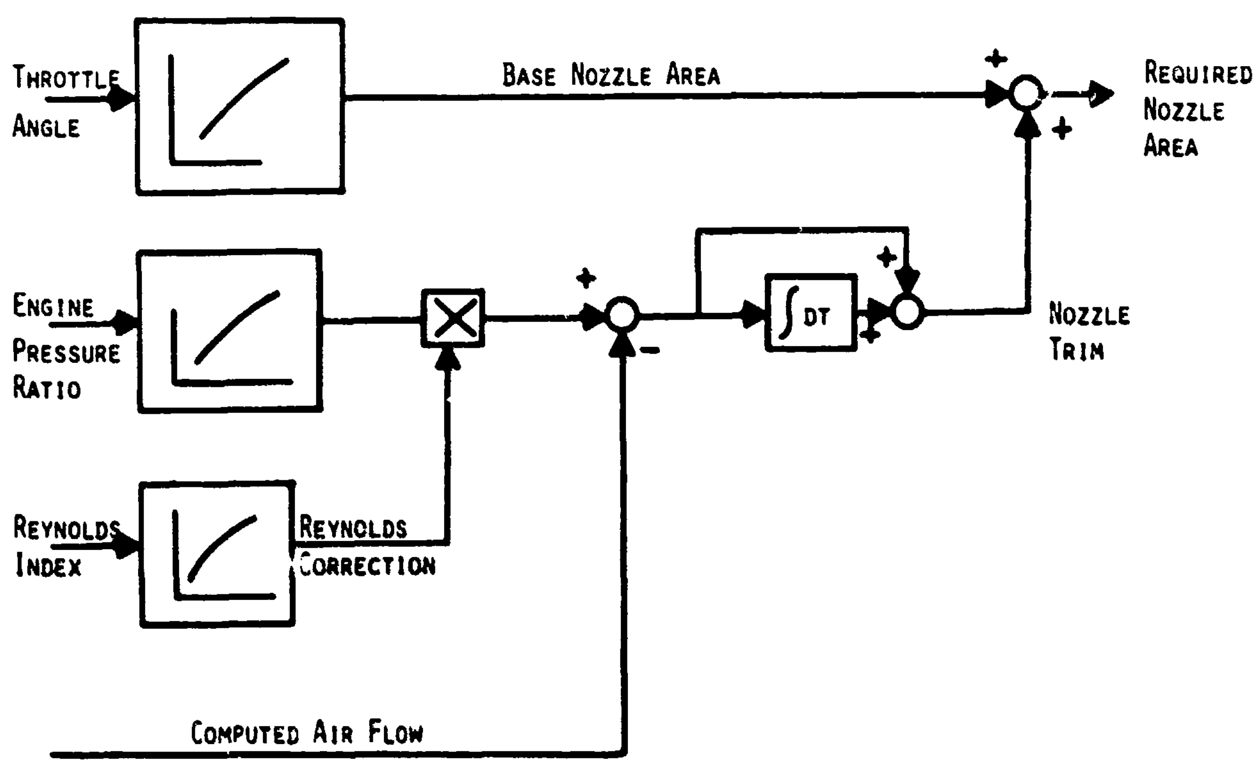
THE IPCS SIGNAL FLOW
(FIG 3)



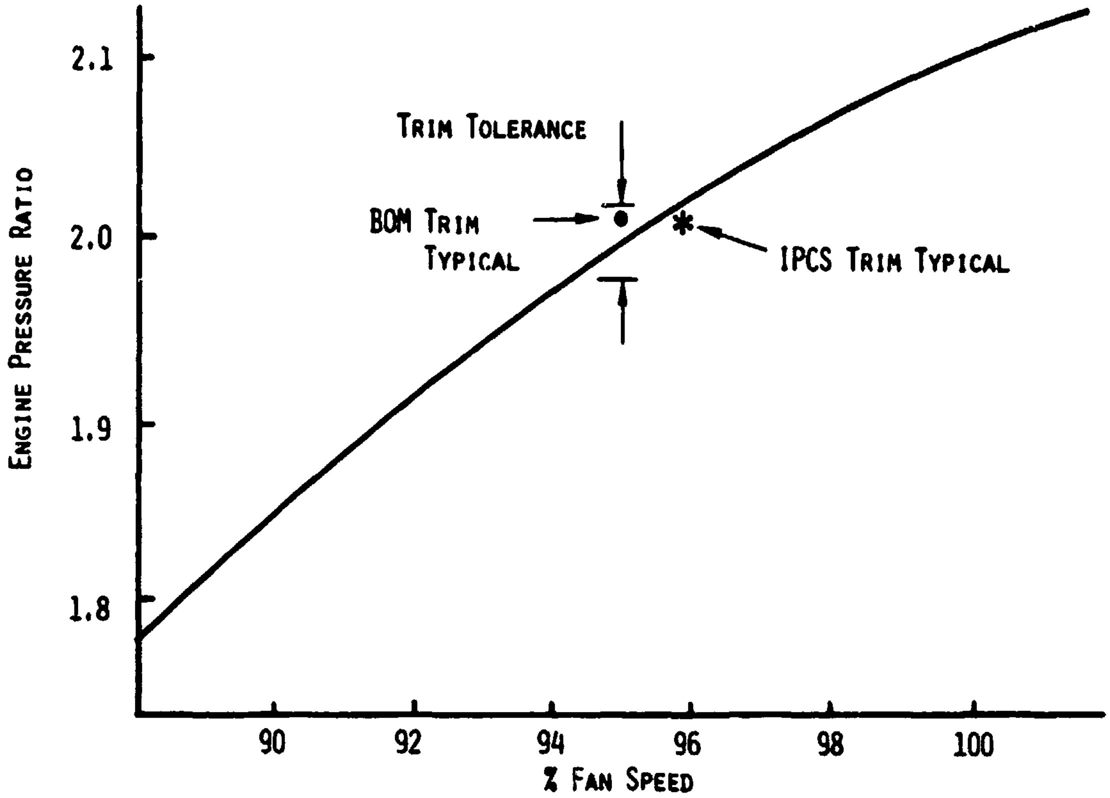
IPCS INSTALLATION
(FIG 4)



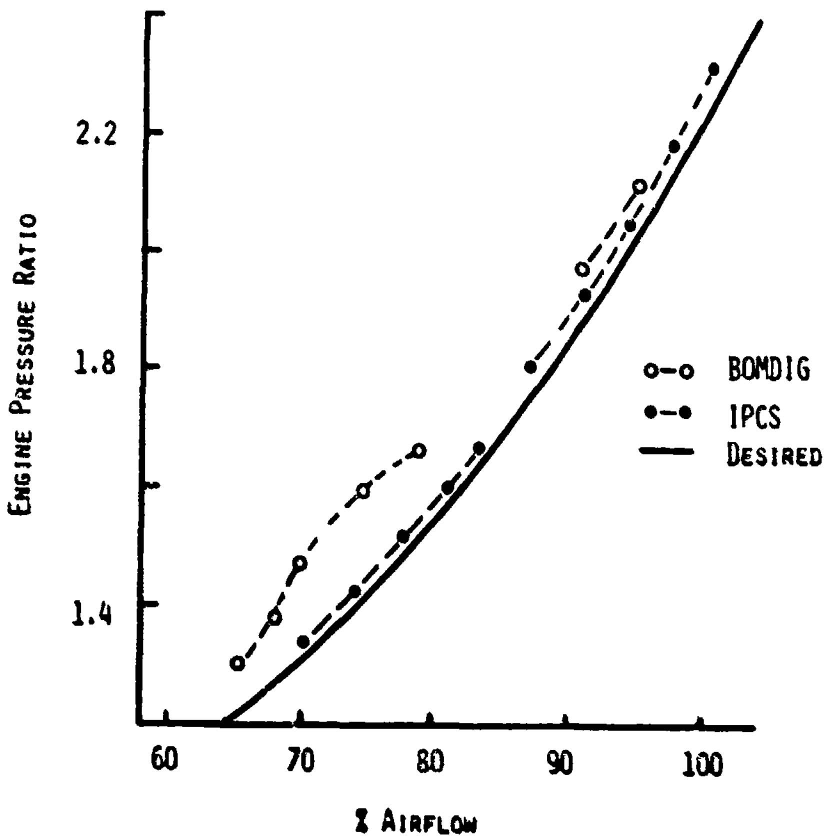
HMC/BOMDIG Comparison
(Fig 5)



Nozzle Area Control
(Fig 6)



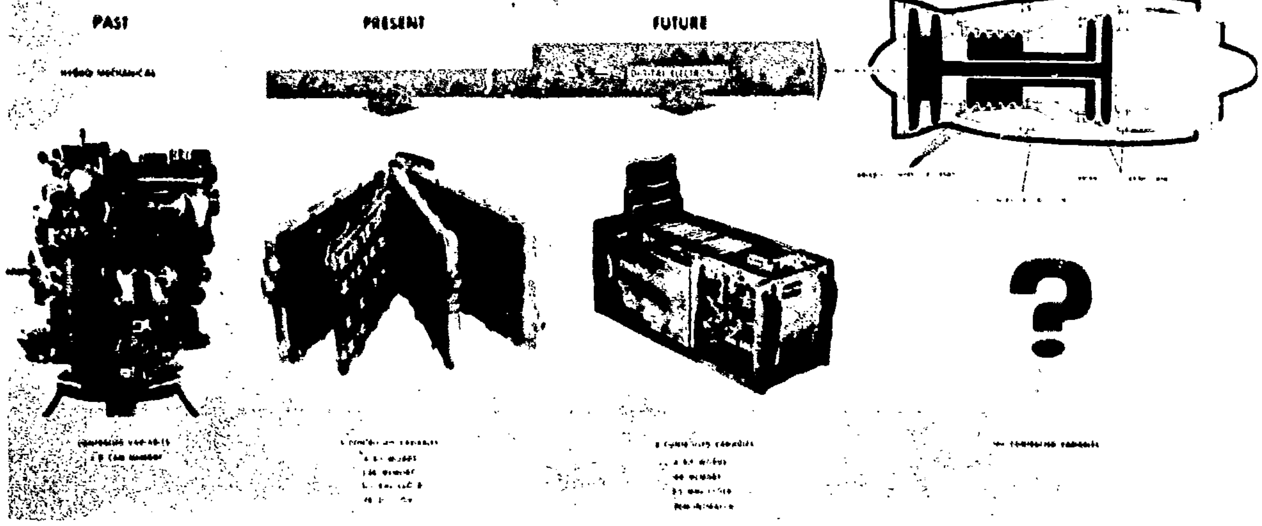
TF-30 Trim Curve
(Fig 7)



Fan Operating Lines
(Fig 8)

PROPULSION CONTROL TECHNOLOGY

- THE ULTIMATE GOAL OF THE AFAPL's CONTROLS PROGRAM IS TO RELIABLY AND INEXPENSIVELY OPERATE A COMPLEX PROPULSION SYSTEM IN A NEAR-OPTIMAL MANNER, FROM -54°C TO +495°C AT ALTITUDES RANGING FROM SEA LEVEL TO 100,000 FEET



- ADVANCED CONTROLS TECHNOLOGY OFFERS A 15% REDUCTION IN FUEL CONSUMPTION FOR MILITARY AIRCRAFT. SIGNIFICANT INCREASES IN NET THRUST ARE ALSO POSSIBLE. FUTURE CONTROL SYSTEMS WILL BE DEVELOPED MORE EASILY AND WILL BE FAR MORE RELIABLE.

(FIG 9)

DISCUSSION

J.C.Ripoll

In your comments of flight tests results you mentioned "extreme flight conditions". Does this include very high angle of attack?

Author's Reply

The term "extreme flight conditions" in the context of my paper specifically referred to low Mach number, high altitude operation, in other words low Reynolds number conditions. High angle of attack operation was also investigated in the category called maneuvering.

J.Kurzak

How fast were your reheat transients?

Author's Reply

That is a different question to answer with one number. It varied a lot over the flight envelope. I would say at one point sea level static for example we cut the time in half of what a standard system would do. We ultimately became limited by the speed at which the nozzle could be actuated and we could not afford putting in faster actuators because a stronger structure of the nozzle would have been needed.

J.Kurzak

Is that in the order of one, two, three, four seconds?

Author's Reply

Yes, certainly.

J.Kurzak

And it is in, say, 2 seconds you completed a full control cycle calculation with the mass flow matching it, with the nozzle area on the working line, and feeding the correct fuel rate to it. Was that a closed loop in these one or two second ramp times?

Author's Reply

Yes, this resulted from the basic computer cycle time of 30 ms. A new computation was done every 30 ms and in some cases there was a minor cycle for rather fast or critical computations that was accomplished in only 5 ms.

Ch Bourgarel

J'ai bien compris la plupart des avantages que ce système de régulation van donner et les trois gains de consommation spécifique qu'on peut en attendre. Est-ce que, en cas de panne partielle ou de panne totale du système il est prévu un système de secours hydro-mécanique très simplifié qui permet quand meme d'utiliser la machine?

Author's Reply

The basic program we ran was exploratory development which in the USA means that we are not preparing a system for production. In the case of the system that was actually flown we had a back-up hydromechanical unit on board the aircraft. And, in fact, the reason it was there let me add that it was used eventually was that it was very inexpensive to modify it to accept an electrical input. So any time the pilot felt the computer was not acting properly or any time the computer decided it was not acting properly it would automatically transfer to the hydromechanical control unit.

W.Biehl

Can you give the order of size of improvement in terms of thrust and SFC by (a) going to digital engine control, (b) integrating inlet and engine control?

Author's Reply

It is hard to break numbers down in that fashion. however, we did achieve, for example, a 16% improvement in supersonic range. At one point, we doubled the rate of climb just by having one engine modified and switched into IPCS, mode, which is in effect an increase of thrust. Ultimately, I think the payoff for integrating the inlet and engine controls together will be in reliability, cost, and weight of the control system. It would be possible certainly to handle the intercommunication between separate boxes, technically, however, the economics might dictate that we put them together.

W.Biehl

What problems do you see in applying an integrated inlet engine control system for new fighters to be developed as aircraft and engine developments often do no go synchronous time wise.

Author's Reply

The biggest problem I see in actually achieving this control and production are legal and corporate problems, not technical ones.

R.J.Eves

Can the author please enlarge further on which control loops and engine parameters required the faster sample rate (5 milliseconds?) and what the particular control problems were which necessitated this?

Author's Reply

One of the parameters was a high frequency intake pressure, utilized in determining engine surge margin requirements. Another was a pair of pressures at the rear of the high compressor used to determine available engine margin.

A.C.Willmer

If I have understood correctly, you have used corrected airflows from the fan as one of the control parameters. Can you say how this was calculated or measured in the aircraft during flight?

Author's Reply

Yes, in part of the IPCS program we were quite concerned about maneuver-induced distortion at the fan face. We ultimately used four pressure sensors at the fan face, specifically located for that particular aircraft to give an indication of distortion. At the same time as long as we were doing this and handling the information onto the computer, we also combined these with tail pipe pressure to formulate what came close to being fan pressure ratio. Given fan pressure ratio, fan corrected speed, and some measure of the Reynolds number index, we were then able to go back and calculate the airflow.

A.C.Willmer

There was no sort of correction for distortional effects on the fan itself either?

Author's Reply

An empirical correction for distortion effects was utilized.

B.Swann

How does the system cope with faulty sensors?

Author's Reply

In this control, action was taken due to failures of some sensors; the computer would check out of range, it would not check for slow drift. In some cases, we had sufficient information in the computer that if a sensor was detected as being out of limits it was switched out of the system and a back-up computation of the parameter was used. In fact this feature turned out to be quite useful during flight test. The wrong specification tubing was used in one place and broke up during supersonic flight. The sensor started reading out of range. When the first sensor read out of range the computer switched to the back-up computation. When the second sensor went out of range due to this tubing failure the computer turned itself off, and we went back into the hydromechanical mode automatically.

THE BENEFITS OF AN INTEGRATED DIGITAL
POWERPLANT CONTROL SYSTEM
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SUMMARY

The engine air intakes on Concorde are controlled by special-purpose digital processors. The flight experience gained on air intake control has been applied to a study of the feasibility of integrating the controls of a powerplant (air intake, engine, reheat, and nozzles) into a single digital system.

The paper describes the control system derived in that study, and indicates the benefits and problems of integration.

INTRODUCTION

The Olympus jet engines on Concorde require that variable geometry air intakes be employed to supply air at a relatively low speed and corresponding high pressure. In order to optimise the aircraft's performance over its whole operating range the geometry of the intakes needs to be controlled accurately to complex schedules of aircraft and powerplant parameters, both in steady-state and dynamic conditions. In 1970 it was decided that control of the Concorde intakes could best be performed by an airborne digital processor, and the Electronic Systems Group of the British Aircraft Corporation undertook the design and manufacture of the control system. A powerful tool used in the design was a mathematical model of the intake and engine which was developed on a hybrid computer, and which could simulate the response to atmospheric or pilot-induced transients over a wide range of operating conditions. The model not only gave design data for the control system but was also used to test prototype hardware and to back up flight trials which began in March 1973.

In 1974 the National Gas Turbine Establishment (part of the Procurement Executive of the Ministry of Defence) asked that the expertise gained on intake control be used to investigate the feasibility and cost-effectiveness of integrating the controls of the various subsystems of a powerplant, i.e. the intakes, gas-turbine, reheat, nozzles and reverse thrust mechanism, (see figure 1). Currently most of these subsystems are controlled independently and this gives an excess of control hardware due to the increase in parameter sensing and data transmission requirements; the diversity of equipment brings further problems of increased spares holding and maintenance facilities. As powerplant configurations take on a greater degree of variability, and as vectored thrust is further exploited, the need for an integrated powerplant control which can communicate with the main aircraft flight controller will be even greater. It was therefore necessary to show the technical feasibility, advantages and limitations of an integrated system with regard to current powerplant and controls technology. It was decided to use a civil aircraft as this has the more visible need for economic operation and in order to avoid the use of classified information; a Concorde-type powerplant was used in order that the study might benefit directly from past experience and an existing simulation. Furthermore to comply with present airworthiness requirements, it was decided to concentrate on integration "within" a single powerplant rather than "across" powerplants.

The systems which have been integrated are:-

Intake control, Primary nozzle control, Engine Starting Control, Engine fuel flow control, Noise abatement procedures, Secondary Air door control, Reheat fuel flow control, Reverse Thrust selection, Part of the Air Data System.

The derived systems have been kept independent of the aircraft's Central Air Data Computers in order to free the study from the constraint of having to configure the systems with regard to an interface with a second large system which is likely to be dependent upon the particular aircraft type.

The intake control system on Concorde has demonstrated that digital controls on aircraft are not only feasible, but highly desirable from the flexibility and reliability viewpoints especially for the more complex systems such as integrated powerplant control. Consequently an analogue integrated control system was not considered in the study.

The study was undertaken in three parts. The first was to investigate the feasibility of integrating the system by deriving a hardware configuration based on a centralised processing unit controlling all the powerplant subsystems. The second was to disperse this derived system into seperate units each associated with only one subsystem, and to compare the hardware parameters of the integrated and dispersed systems. (This seemingly inverted approach was adopted in order to isolate the benefits of integration from those which could be obtained by other means and so as not to be constrained by the form of the present equipment.) The third part of the study was to define the benefits that could be obtained from integration of the control algorithms, and from the generation of new algorithms subsequent to certain failures, made possible by the hardware integration.

DERIVATION OF THE INTEGRATED SYSTEM

The general shape of a supersonic airliner (figure 2) suggests the siting of the electronic control units near the rear of the fuselage to minimise the wiring runs to the powerplant whilst providing a reasonable environment and easy access. However, this means that cockpit signals and air data information must be transmitted over a long distance. In the case of pressure signals necessary for the computation of aircraft Mach number long piping runs are impractical and pressure sensors must be sited near the pitot probes. Once a unit to house these sensors has been established at the front of the aircraft it is advantageous to use them as data collection units for all cockpit and air data signals, and to transmit the multiplexed signals to the control units using digital format to interface directly with direct access memory of the control computer.

The control laws for the individual subsystems were assessed with a view to obtaining the requirements of total computer power for the integrated controller. An allowance was then made for the inclusion of cross-coupling of the control algorithms, for post-failure algorithms, and for the system monitoring task. The total computing requirements were estimated to be:

Word length	16 bits
Storage	10,000 words
Input/output	125 channels and 6,000 samples per second
Add time	5 microseconds
Multiply time	50 microseconds

The specification of a multiply time is derived from the large number of multiply instructions and function generators in the control algorithms: this largely determines the overall response of the control system. In a survey to find if these requirements could be met by a commercially-available microprocessor without recourse to special-purpose hardware, it was found that, in general, devices brought onto the market within the last couple of years could meet the need whereas older devices could not. Consequently, extrapolating into the near future, it is apparent that computing power would not be a serious problem for integrated controllers and that the problems of multi-processing would not be encountered. However, the large number of input-output channels needing servicing at high frequency indicated that asynchronous analogue-to-digital conversion and direct memory access to the processor was vital. This DMA facility would be common with that which inputs signals from the data collection units.

The system now takes the format of figure 3. Estimates of the failure rates of the units were made in order to indicate the level of redundancy necessary for the system to meet safety requirements. It was found that the control units would each have a mean time between failures (MTBF) of 2100 hours and so the requirements could be met by a system which had dual control units for each powerplant with one in control whilst the other stood in readiness to be switched in when the first indicated a fault; the data collection units, each with a MTBF of 6,000 hours, would need to be triplicated with all three feeding all the control units, as shown in figure 4.

Each unit has the capability to detect the failures within its own control circuitry and input and output interfaces. Furthermore each control unit can compare certain of its computations with equivalent computations made in the other control unit. In this way the vast majority of control system faults can be detected and isolated to a single hardware module so upholding system integrity and facilitating maintenance. This system configuration allows another maintenance feature to be easily included - namely an in-situ test system, which can test the whole control system in all its operational modes at maintenance periods or prior to each flight. At these times safety tests can be conducted which could not be carried out during flight. The use of a software-based control system means that the test sequences can be incorporated with very little increase in hardware and the use of data collection units which interface with all control units means that these test sequences can be stimulated with very little interface hardware. Consequently the major hardware increase for the incorporation of a test system is simply a unit to initiate the tests and to observe the results.

DISPERSAL OF THE SYSTEM

The dispersed system consists of four control subsystems as follows:-

- Air intake and secondary air door control
- Engine fuel flow, starting and primary nozzle control
- Reheat fuel control
- Noise abatement, secondary nozzle and reverse thrust control.

Each of these can be made independent of the rest except for their sharing of sensed inputs. However all subsystems, with the exception of reheat, require an input of some air data signals - Mach number, altitude, etc., derived from the aircraft pitot-static probes. Consequently the data collection units of the integrated system are required in a similar form for the dispersed system, although now the pilot's control signals will not pass through this unit.

Although the integrated system had used digital technology throughout, it was felt that justification was needed before this could be applied to the dispersed system. The life-cost/complexity curves for analogue and digital systems are shown in figures 5a and 5b. The analogue system shows the characteristic zero cost for zero complexity and the asymptote beyond which any increase in complexity is prohibitively costly, especially when fault identification and spares holding is considered. The digital system shows a relatively high initial cost, with a small increase in cost for any complexity increase over a wide range; the high sensitivity to the number of similar units reflects the commonality of spares and test equipment which is apparent only in digital systems. Taking a representative curve from each of these figures and cross-plotting them in figure 5c indicates that the thrust reverse and secondary nozzle controller could be implemented in either technology whereas the other three units should be digital. When it is considered that the data collection unit could also be based on a microprocessor the benefit of having the whole system in a common technology can be realised and the all-digital system can be compared to the integrated system using common cost synthetics.

Comparison of system failure rates with safety requirements indicated that the intake and engine controllers, with MTBF of 3,400 and 2,900 hours respectively, should be dual-lane, whereas the reheat and nozzle controllers (both with 4,400 hours MTBF) could be simplex for each powerplant. The data collection units should remain triplicated. Thus the dispersed system takes on the appearance shown in figure 6. The siting of equipment on the aircraft would remain unchanged.

THE BENEFITS AND PROBLEMS OF INTEGRATION

In comparing the integrated and the dispersed systems it becomes apparent that there are three major areas of cost reduction derived from system integration. These areas are:-

(a) Equipment costs due to reduction in the amount of system hardware.
(b) Maintenance costs due to an increase in the commonality of control equipment.
(c) Running costs due to an improvement in powerplant performance.

Against these benefits is the fear that loss of independence of the parts of the powerplant will hazard the aircraft.

Equipment Costs

In making a comparison between the hardware of the dispersed and integrated systems it is necessary to define reasonable limits on the system within which to assess size, weight etc. The limits chosen are the outputs of the sensors and the inputs of the actuators. The sensors and actuators themselves are either common in the two systems, or vary to a very small degree (for example the number of sensing elements in the intake temperature probe). Also all displays have been omitted in anticipation of the move towards an integrated cockpit design.

The simplest way to quantitatively compare the hardware of the integrated and dispersed systems is to use size, weight, power consumption and the mean time between any fault in the system. Estimates of these values are shown below.

	INTEGRATED CONTROL SYSTEM	DISPERSED CONTROL SYSTEM	UNITS
VOLUME	0.18	0.29	m^3
CABLE LENGTH	16	25	Km
UNIT AND CABLE WEIGHT	380	580	Kg
POWER CONSUMPTION	2.3	4.0	KVA
MEAN TIME TO TROUBLE	230	140	Hours

These figures indicate approximately a 40% improvement of the integrated system over the dispersed system. This would indicate a 40% cost saving for the purchase of the control system.

Maintenance Costs

Maintenance costs can be considered under two headings: capital costs, for the purchase of spares and test equipment (or ATE software) and the training of maintenance personnel; and the recurring costs of repairs to the aircraft equipment.

The capital costs are more dependent upon the number of different types of units in the system than on the complexity of each unit. In the dispersed system the units are of low-to-medium complexity, but there are five different types, whereas the integrated system has only two types of complex units. Allowing for the commonality of the central processing elements of the units in the dispersed system and the complexity of each unit, it has been estimated that adoption of the integrated system would give a 33% saving in capital maintenance costs.

For each fault in the system the cost of repair is related to the ease of isolating the fault, the cost of the element to be replaced, the effort required to make the replacement, and the ease of retesting the equipment. The middle two costs will be similar for the dispersed and integrated systems, but isolation of the fault and unit retest will be more costly in the more complex integrated units: it is estimated that this is a 25% cost increase for each fault. However, the integrated system only exhibits a fault once every 230 hours, compared with once every 140 hours for the dispersed system, and so the repair costs per unit time are, in fact, 25% lower in the integrated system.

Running Costs

The third benefit of integration is improved performance or fuel consumption that can arise from integrating the control algorithms of the various parts of the powerplant. No attempt has been made to quantify this improvement as many levels of integration are possible and the results would be applicable to only one aircraft type. However in order to indicate how these benefits can arise a study was conducted on a full-range areo-thermodynamic model of the whole powerplant. The model was an extension of that used for the Concorde air intake design and was validated by comparing its results with data from flight trials. The model was found to be within the scatter between powerplants for the same time record as can be seen from the throttle-slam traces in figure 7.

To date, the work on the integration of the control algorithms has been concentrated on trying to reduce the pressure fluctuations in the intake which are caused when reheat is ignited or when the throttles are moved rapidly. It has been found that by incorporating primary nozzle area and jet pipe pressure signals into the intake control these fluctuations can be reduced by about 40% without deteriorating the engine's response.

Although per se the reduction of transients in the powerplant have no economic benefit, they do give scope for improved performance in steady-state operation. This is because the engine running lines normally have to be displaced from their optimum to give an adequate surge margin, and this displacement is determined with regard to the magnitude of possible transients.

Control System Faults

Although the integrated control system can be shown to meet all its safety requirements, it is true that certain failures can have a greater effect on aircraft handling than would be experienced with a dispersed system. Although very little can be done to reduce the effects of some of the more remote faults (for example the failure of both control computers for one powerplant), integration can, in fact, alleviate some of the more probable failures. Two examples of this are: the modification of the control algorithm for one subsystem, following a failure in the control of another subsystem; and synthesis of a sensed parameter from other information in the powerplant following a sensor failure. As an example of this latter case, a study was undertaken using the powerplant simulation to synthesise a parameter which could replace the engine low pressure spool speed (N_1) should its sensor fail. This parameter was chosen as it affects the control of the intake, primary nozzle and reheat, and is thus an important parameter to maintain in an integrat system. N_1 was synthesised from the high-pressure spool speed, primary nozzle position and intake temperature, each with suitable lags. Figure 8 compares the effects of a throttle slam using the real N_1 with those using the synthesised N_1, by showing the time-histories of the excursions of intake recovery and real N_1 from their scheduled values. Although it can be seen that control is degraded in the acceleration case the powerplant is still maintained in a safe, surge-free condition.

CONCLUSIONS

Integration of the powerplant controls for a supersonic airliner is feasible using modern digital control technology and integrity targets can be met using a dual-lane type of system with triplex sensing of cockpit and air-data information. The benefits of adopting an integrated hardware design can be shown to be about 40% in control system purchase costs and 25% in maintenance costs. Integration of the control hardware permits integration of the control algorithms and this can have a beneficial effect on powerplant performance and hence aircraft operating costs.

Although systems integrity is apparently reduced by integration there are other integrity features-such as on-aircraft test systems and reversionary control modes subsequent to failures - which are greatly simplified in an integrated system.

ACKNOWLEDGEMENT

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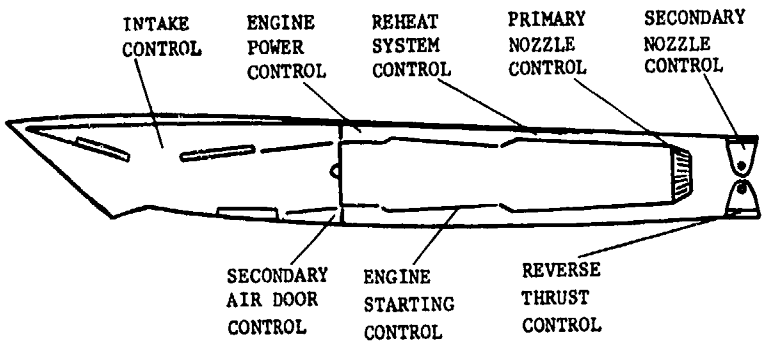
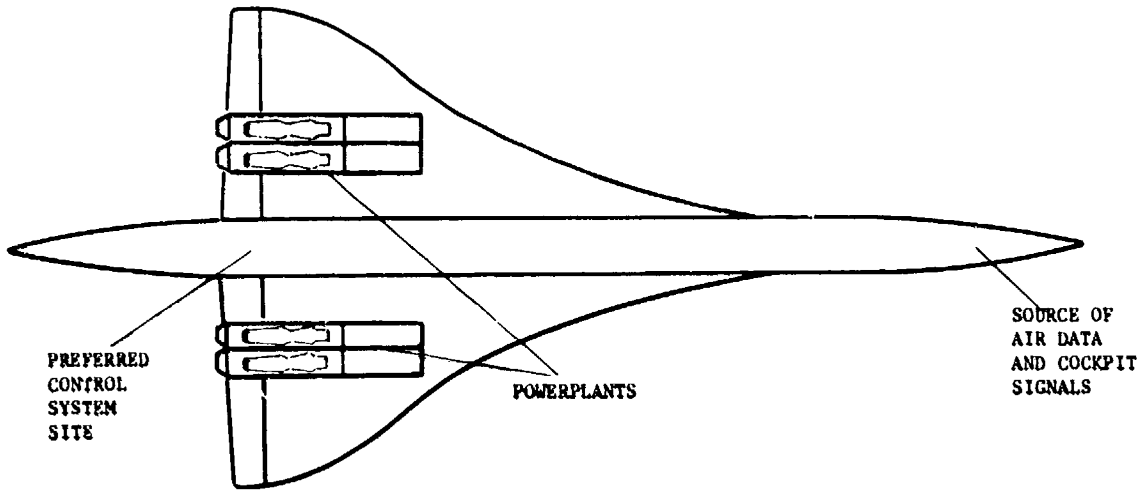


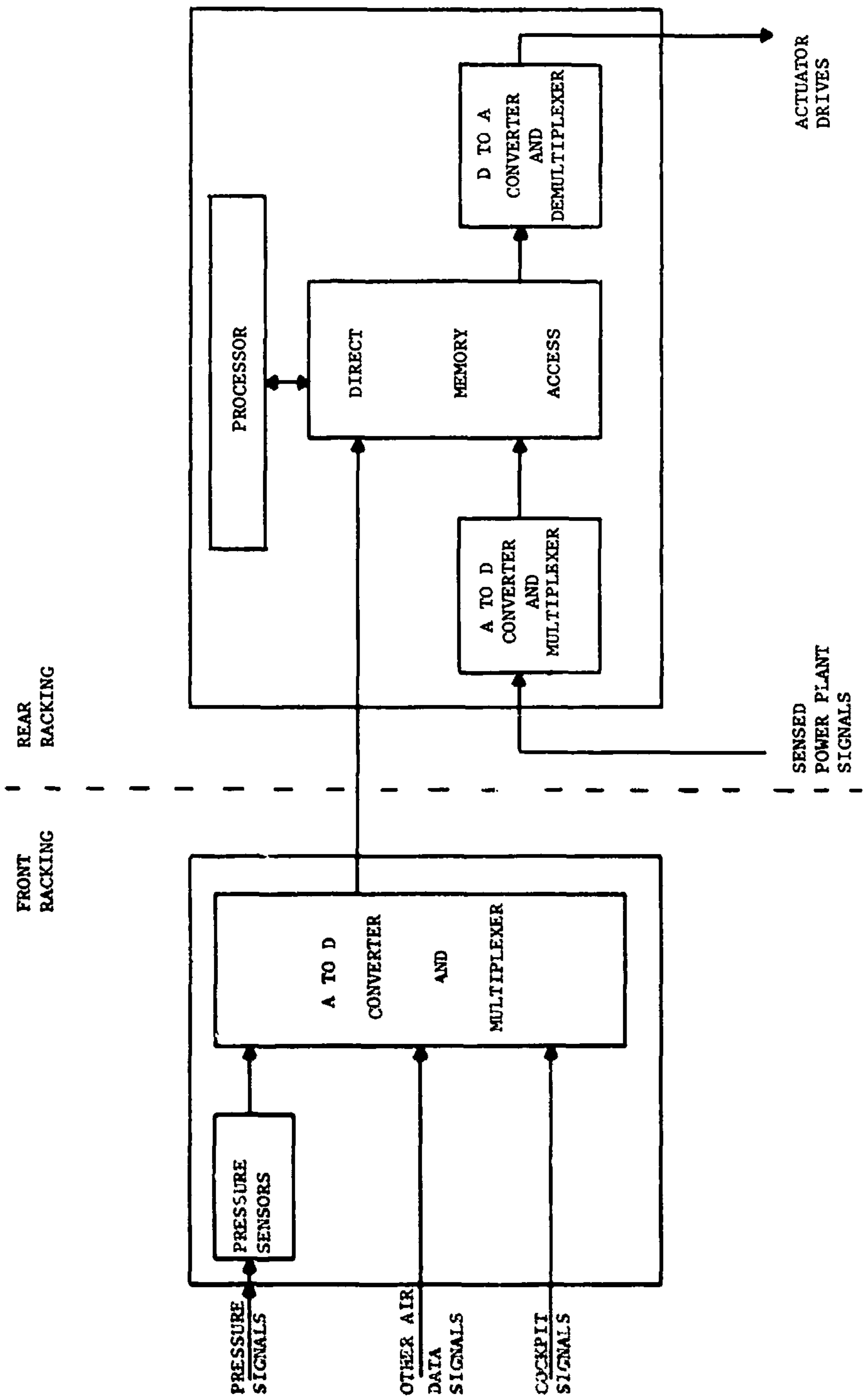
THE CONTROLS OF A POWERPLANT

Fig.1



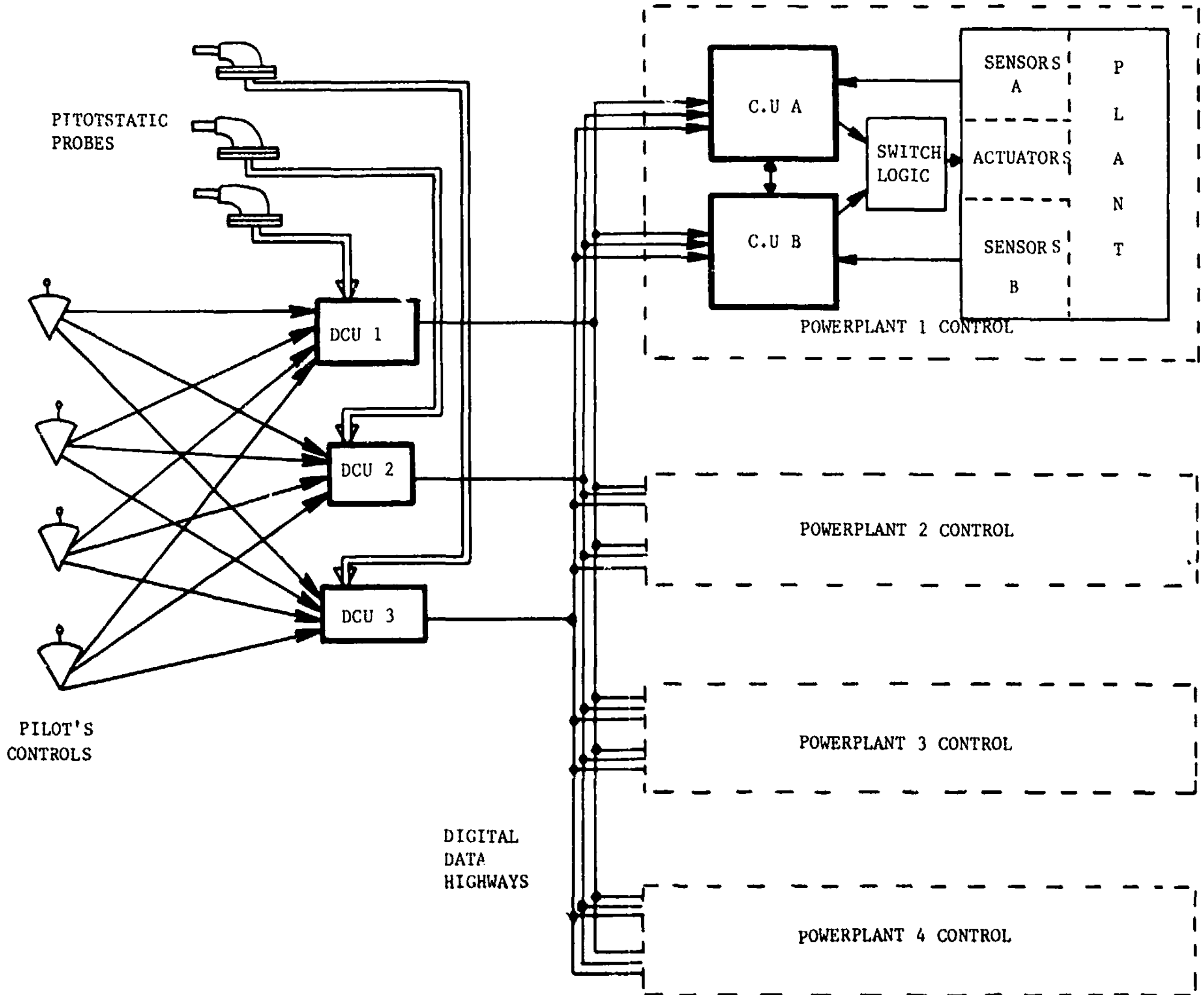
CONTROL EQUIPMENT LAYOUT

Fig.2

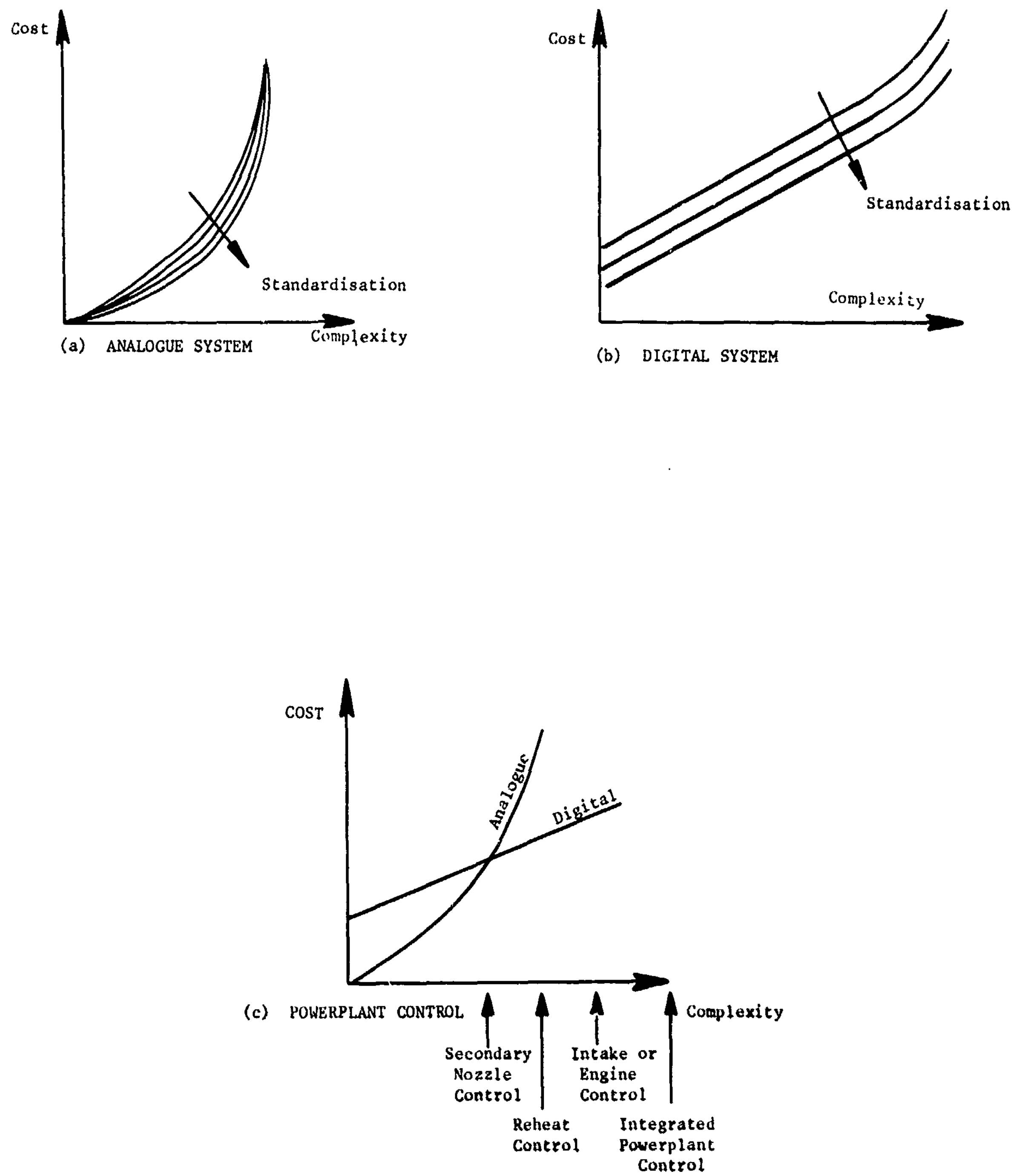


INTEGRATED CONTROL LANE SCHEMATIC

Fig.3

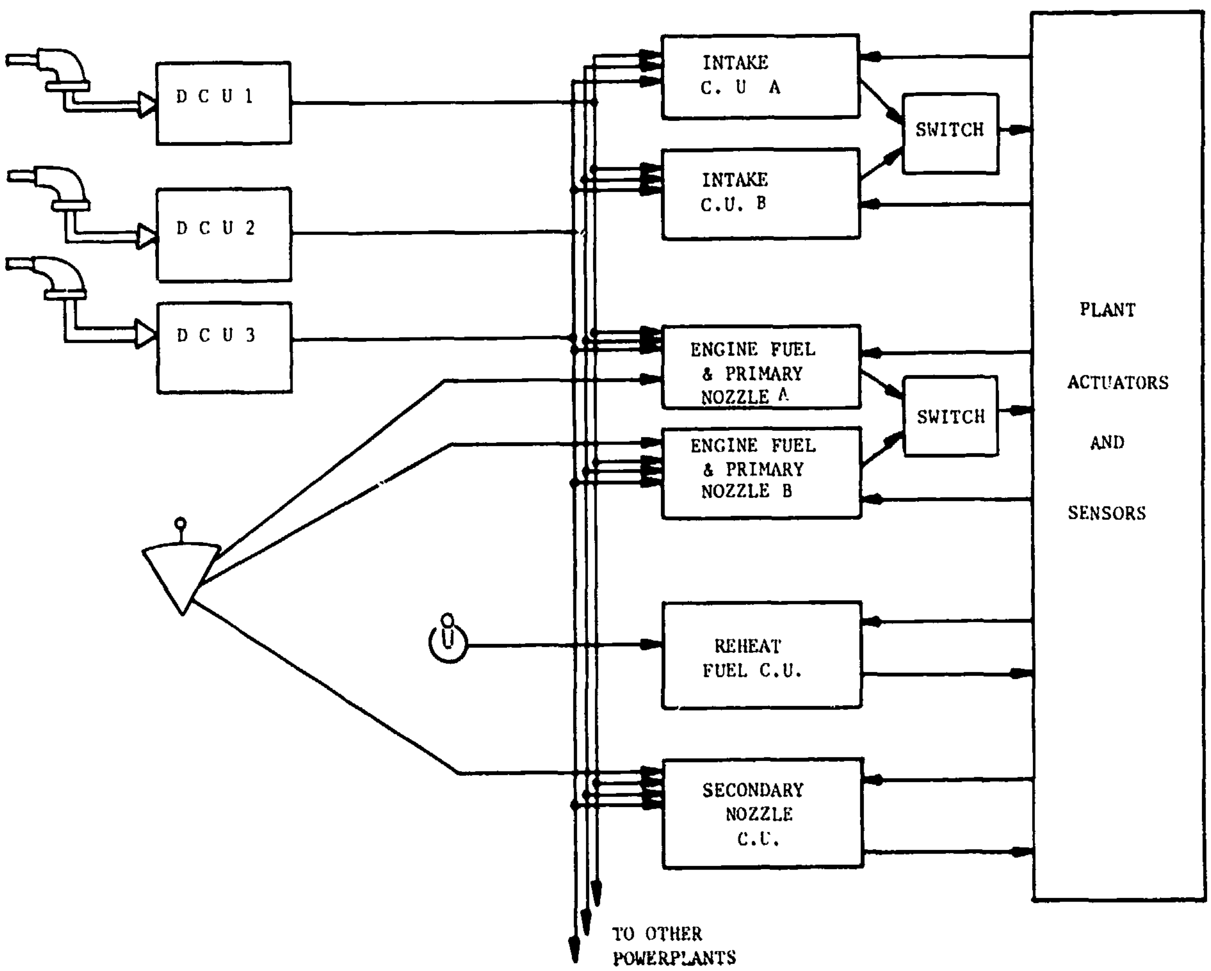


INTEGRATED CONTROL SYSTEM SCHEMATIC Fig.4



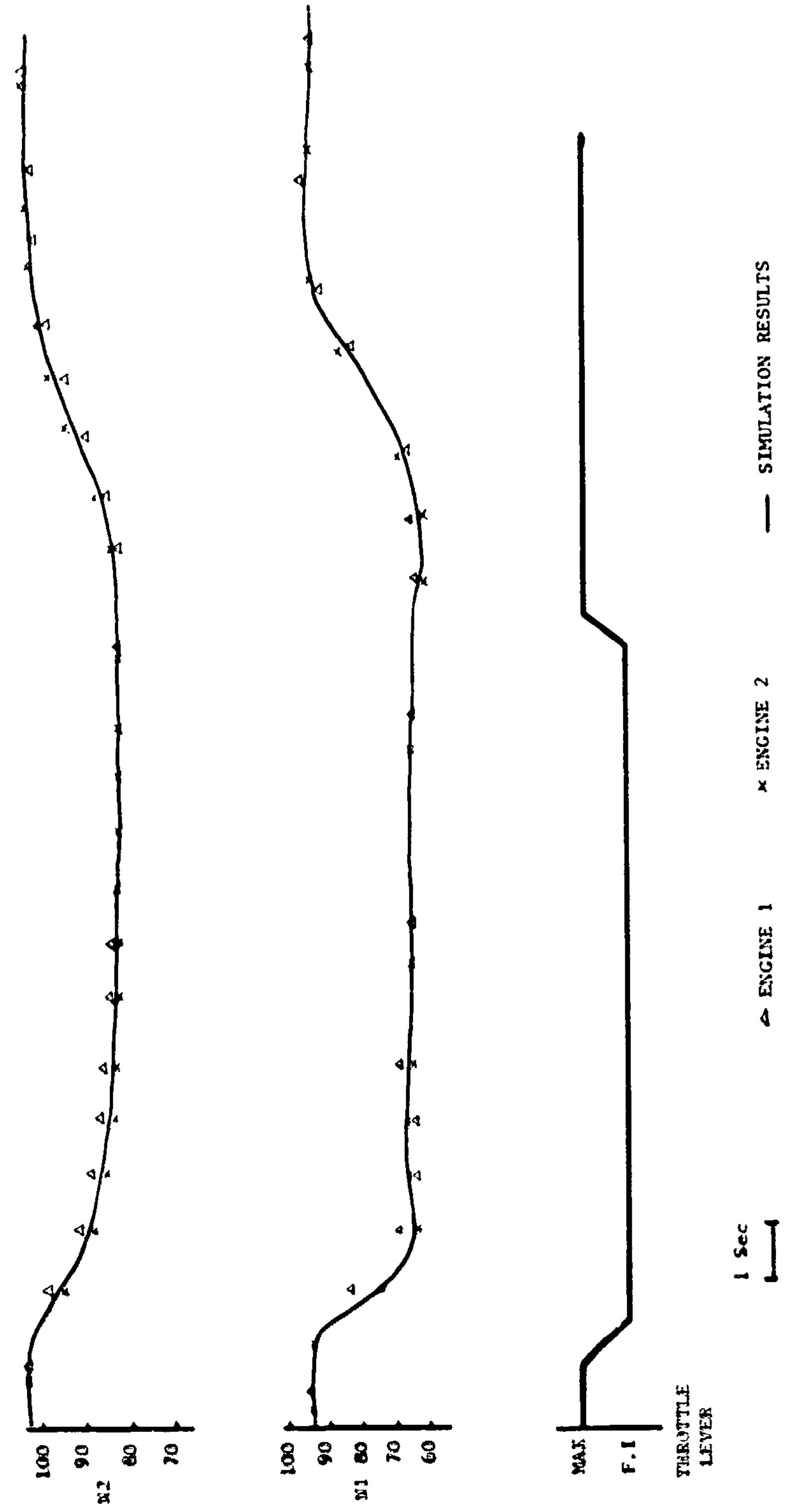
SYSTEM COSTS VERSUS COMPLEXITY

Fig.5



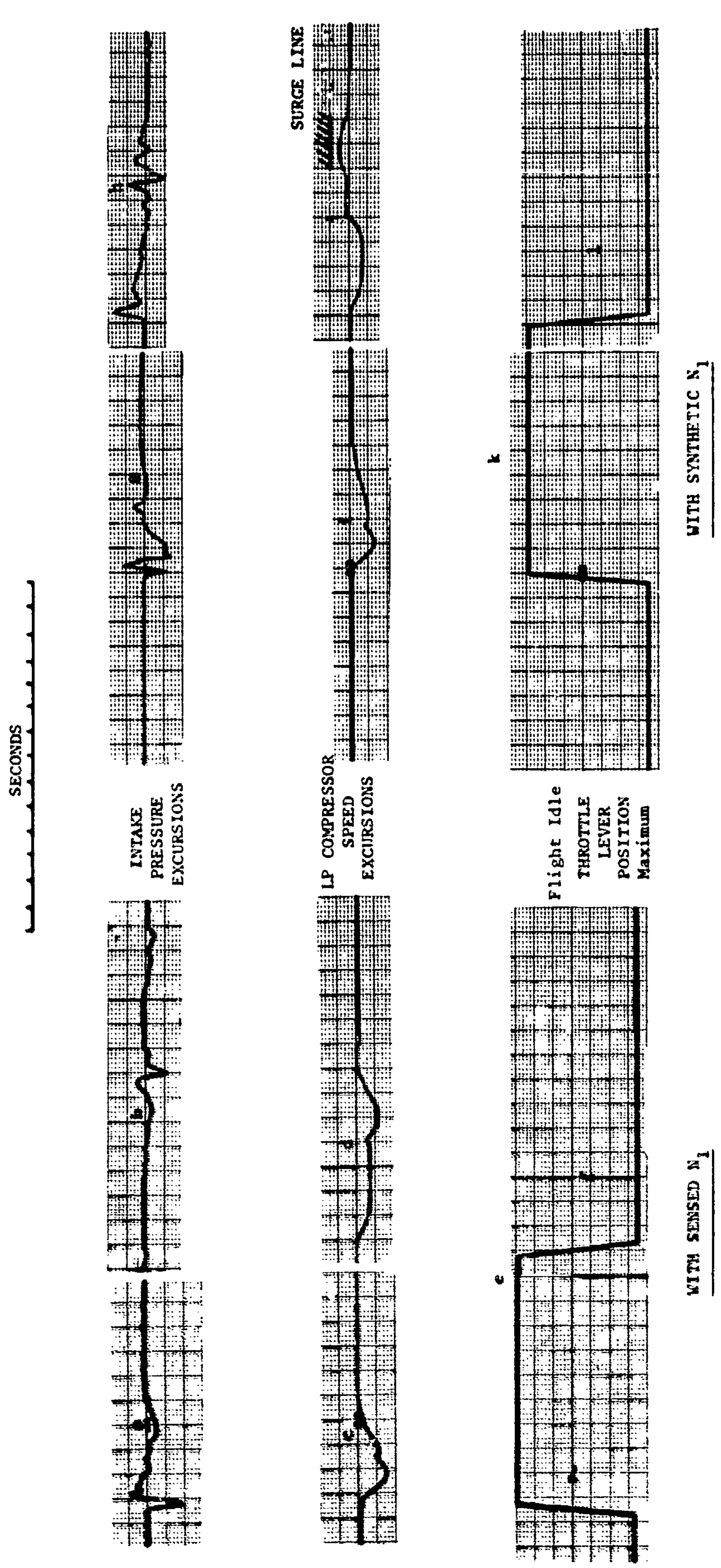
DISPERSED CONTROL SYSTEM SCHEMATIC

Fig.6



POWERPLANT MODEL VALIDATION

Fig.7



CONTROL SUBSEQUENT TO AN N_1 SENSOR FAILURE

Fig.8

DISCUSSION

W.C.Swan

You state that the digital system was dual with switching. Was the system fault monitored automatically and hence switching could occur due to some software judgement system?

Author's Reply

Yes it was. This has been a philosophy on our intake control system on Concorde and with slight modifications we took it across to this study. The main modification has been a limited amount of cross comparing of signals between the two lines of control in the power plant.

J.M.Hardy

Vous me dites que vous avez fait une étude de consommation spécifique. Je pense que vous l'avez particulièrement faite dans le cas où la température d'entrée varie à Mach 2. Pouvez-vous nous indiquer, par exemple, pour les conditions ISA 15 quel était le gain de consommation spécifique obtenu par votre système?

Author's Reply

Our model of the power plant was designed mainly to tell us the compatability between the components and between the sub-systems. It was not a model trying to predict specific fuel consumption, so I have no information on this topic. We considered that performance benefits can arise from integration of controls, but primarily this is derived from reduction of mismatching of the subsystems rather than a direct reduction of SFC.

R.J.Eves

Could the author please comment on how software reliability could be dealt with? Can he suggest any methods whereby the software design can be demonstrated to have met the reliability targets?

Author's Reply

This of course is something of very great concern for a lot of people at the moment. It seems that reliability of software as you have discussed it is no more difficult than getting the design of an analogue control in the first case. However, there are more complex control laws being used in digital control systems and consequently the problems are greater. I feel that the steps being taken at the moment with modular software and testing at modular levels, gradually building up to testing of the whole system, will prove as good as the testing of normal analogue or hydromechanical system designs.

R.J.Eves

Can you suggest a means of achieving this system of documentation or some other means that could be employed because software is all documentation and there can be a vast amount of it.

Author's Reply

To me as a control system designer, hardware is documentation as well. Somebody has still got to explain how it works. Granted that software is all documentation, but I would feel with the use of high level language now coming into the control field this problem is considerably simplified. The programs have become more visible to a lot more control engineers and to a lot of people on the specification side of control systems. It is my opinion that at the moment we are in an evolutionary phase which highlights all the problems. Eventually, this will settle down and everybody will agree on methods of documentation. I think that at this time the most promising system is the modular 'top down' approach to software documentation and to software writing. We should continue on that track and modify it as necessary as the technology develops.

R.M.Denning

In your comparison of dispersed and integrated control system for the Concorde you show a saving of 200 Kg (2 passengers) from the integrated layout. The current aircraft has an analogue engine control. Is this assumed in the dispersed system or are they both all digital?

Author's Reply

We have compared digital with digital. We did briefly undertake a comparison of analogue and digital for the dispersed control system and in our opinion the intake control and the engine control should certainly be digital on a cost effective basis. The reheat control and the secondary nozzle control is perhaps a borderline case. However, we feel that having a commonality of technology would give benefits and so we have treated all the units as being digital.

ROUND TABLE DISCUSSION

Round Table Panel

Mr J.F.CHEVALIER	SNECMA	France
Dr Ing. H.GRIEB	MTU	Germany
Mr D.R.HIGTON	NGTE	United Kingdom
Mr N.F.REKOS	NASA	United States
Mr M.STOLL	DASSAULT	France
Mr W.C.SWAN	BOEING	United States

Mr Nelson F.Rekos, Programme Committee Chairman: During the past four days, we have listened to a number of excellent thought provoking papers concerning the technical possibilities of variable cycle engines for future military and civil aircraft applications. We have heard how the variable cycle engines can provide economically attractive, environmentally acceptable advanced performance for supersonic conventional take-off and landing aircraft and also for vertical take-off and landing aircraft.

We have heard how adaptations of variable cycle engine concepts can provide us with high performance engine for air transport fuel economy. I think we are all aware that the world's petroleum resources are rapidly being depleted, and if we continue to consume these resources at our current rate of consumption, the consensus is that the world will essentially run out of fuel about the turn of the century. Therefore, the need for fuel conservation will be most pressing, if not mandatory, for the future of aviation as we know it today. We know that it takes about ten years or more lead time to develop a fully qualified aircraft, particularly when we begin with some new engine and/or aircraft design. So time is passing, let's not be waiting.

We have seen some provocative fuel conservative engine designs such as the high speed turboprops, which stirs up our imagination, if not, nostalgia. Perhaps we need not fly at higher speed just because speed represents a sign of progress. I'm not sure it may be much wiser to back off from Mach 0.85 (and higher) and fly perhaps at Mach 0.75, where we could use less sophisticated and less costly high aspect ratio wings rather than the expensive swept wings that we have for the Mach 0.85 transport.

We have also heard about the importance of integrated controls, both for the engine and the aircraft and how variable geometry can be applied to combustors and turbines to improve performance and also to reduce engine exhaust emissions. I think the way of the future will dictate, through appropriate regulations, that aircraft must be good neighbors, and not be intrusive on people, not only in the vicinity of airports, but also in the stratosphere where there is a great deal of concern regarding the deterioration of the ozone layer that protects us from radiation. This deterioration results from oxides of nitrogen emitted by aircraft operation, both military and civil. Perhaps, the ozone depletion fear may vanish as we learn more about the atmosphere, but then again, it may be only wishful thinking. I don't think we can afford to take too many chances on our future.

I feel like I am quoting the scriptures in that the text has already been published predicting a petroleum famine lasting for more than the proverbial seven years, predicted by Joseph; perhaps our experts on the Round Table can tell us what we should do in the aeronautics area in preparation for this famine.

I have asked our Round Table Members about their views on variable geometry engine cycle technology; also to identify limitations that may inhibit the development of variable geometry technology for military and civil applications.

I hope they may provide some insight as to whether variable geometry engines will provide such things as surge free engines under all operating conditions. Perhaps they may wish to tell us if the variable geometry engine will demand new types of ground test facilities. I would like now to give the other members of the Round Table their turn, beginning with Mr Swan.

Mr W.C.Swan: I would like to first observe that I have looked at this conference with mixed emotions. I was very pleased to hear the reports in the area of variable geometry turbines, an item I am very much in support of, and I was delighted to hear the positive side of developments of integrated controls because I am sure that this is a necessary supporting function if we are to get positive reliable variable cycle engines into our future aircraft. On the somewhat negative side, I felt that we are repeating or viewing programs that go back almost too far in history. I don't know whether you know it or not, but Gerhardt Newman, of the General Electric Company got his ideas for the variable stators on the J79 engine from me, because I designed and built a multistage compressor at Wright Field in 1948 which had a variable geometry eight stages and ran a two-foot x two foot supersonic wind tunnel. As far as I know, it is still running almost 30 years later. When I hear stories of great fears about variable geometry compressors I wonder how long we have to work at this before we will accept them as a thing that's here. You must expect that my inputs to this committee are provocative, because I feel we should be exploring and not reviewing works of the last 30 years.

Now I would like to make some comments on the meeting of things that I thought were only marginally covered and need to be worked harder before we can all fully appreciate this kind of engine.

I heard only one paper today in the area of nozzles and the effect of the use of variable geometry on nozzles and I thought that this author was beginning to touch a very interesting area. My own experience is that nozzles are that part of the engine to which the engine manufacturer pays little or no attention. Yet, it must be considered in detail in the cycle selection because it is a very high maintenance and drag penalty item. It is one of the most underdesigned items before the aircraft goes into production. It turns out that if you study variable cycle engines, it is conceivable that for certain missions, that include both mixed supersonic and subsonic, we may be able to employ fixed geometry nozzles, because the cycle provides both air flow and pressure ratio from the gas generator that accommodates the fixed geometry nozzle. This is a rather novel concept in life if you think of it, because both the intake and nozzle designs of today are principally developed to accommodate the gas generator. It would be interesting if the engine design would look at the entire system, including the nozzle and intake, and the result might be very simple nozzles and intakes with these engines. When you add up the total cost of the system, the designers should think of the total engine and not just of the compressor face to the turbine flange. We got very little of this thinking today and I think that mechanical design evolution can be greatly enhanced, with variable geometry in the gas producer section. I have alluded to intakes, something we did not look at all this week, and believe it is possible, in some uses of variable geometry engines to actually con[illegible] fixed geometry inlets and let the air spill when you don't want it. This could lead to the simplest and less costly [illegible] system. This is contrary, in part, to the thoughts of our Panel moderator. I would not suggest this idea [illegible] civil use, but for military use, it may very well be that you can afford to pay for complication in the gas producer and get away with an inexpensive, very cheap intake, so that the net cost comes out very favorable.

Another item I'd like to mention is that we all do studies on applications of these engines and we rack them up in aircraft comparisons. I think one should be very cautious about studying the influence of engine design parameters as to whether in your decision process you are just about the end of the cliff. For example, much of the advances in engines that weren't discussed this week are from increases in the combustor exit temperature which has moved up considerably since the work we did on this ten years ago. Some engine concepts are more sensitive to the limit on this value than others and rather than make a decision on the type of engine to build on the basis of some small temperature margin, we should have a look at the effect of deterioration in attainment of the temperature goal and find out how soon the particular engine you are talking about ceases to be a competitor where it was a winner before it fell apart. In other words, we must look at sensitivity of influence coefficients in the result.

Finally, I would like to introduce a movie, and with it I want to introduce the thought of concept validation. I have seen many engine manufacturers and Government developers who, when somebody mentions a new concept to be looked at, start thinking in terms of hundreds of millions of france, marks, dollars, etc., before the concept can be validated. I don't believe in this at all. I believe that if you have an interesting idea, you can validate a concept without an excessive cost. This is true, also, with relation to variable cycle engines.

I am going to show you a film that cost my company exactly $150,000 U.S. dollars to do the whole job, including the design, the engineers, the manufacture of the system, the tests, evaluation and shutting down of the project. I ask you, who work in Government installations and private industries, whether $150,000 dollars is a very small amount of money when it comes to doing such a job. You will agree from what you are about to see in the film that is will have answered at least several very important questions to us on the future of variable cycle engines and it was done at relatively little cost.

Dr H.Grieb: I would like to concentrate my comments on VCE [illegible] military aircraft, that means for aircraft with mixed subsonic and supersonic missions. Further, I will focus my co[illegible] on the VCE itself and its problems involved, whereas I will leave the discussion of the benefits in view of the [illegible] achievable with this type of engines mainly to the aircraft designers.

In accordance with publications on VCEs known so far, I got t[illegible] impression in the course of this conference that the flexibility in performance required can be achieved in several ways. When I am talking about "flexibility in performance" this does not only include favourable (not to say optimum) SFC in reheat and dry operation, subsonic and supersonic, but also substantial improvements in the matching of engine massflow to intake capacity under all flight and engine running conditions, as well as improvements in engine operating stability.

In the papers presented during this conference, many engine concepts have been described which, on the one hand certainly are capable to achieve the flexibility aimed at. But on the other hand, all these concepts look more or less complex and will probably result in considerable excess weight and cost compared with current engines. In fact, the weight and complexity problems to be expected with VCEs appear very difficult to be solved and eventually this would lead to a major draw-back for this type of engines in future. But here we should differentiate

for missions (mixed supersonic and subsonic) of low endurance (say in the order of one hour) my comment on the weight problem will certainly hold.

for missions of medium or long endurance (say of more than two or three hours), the excess weight of VCEs will be of minor importance.

A considerable part of the papers presented was devoted to the technology of variable components, including compressors (that means fans and multistage compre[illegible]ors), combustors, turbines, and not to forget, also thrust deflecting nozzles. Variable compressors, that means compressors with variable stators, as well as variable nozzles are already in use with current engines since a long time. But there is no doubt that variable turbines which are not yet applied in military engines, are of paramount importance in order to achieve the flexibility in performance in the comprehensive sense I mentioned.

Several authors who described VCE concepts made generous use of variable HP turbines. In my mind this is a critical point and it could possibly lead to a further draw-back for VCEs, if the variable HP turbine really would be an unavoidable component. We should realize that the trend to higher turbine inlet temperatures will be going on at least in military engines – and within the next decade a level of say 1700 to 1750 K will be reached. At the same time, this will imply very sophisticated cooling techniques which probably will not be feasible for variable HP turbine guide vanes. Moreover, the trend in turbine inlet temperatures will also worsen the situation at variable LP turbines because these will enter into a temperature region where turbine cooling is required.

It may be encouraging that variable turbines, running at moderate gas temperatures, are in civil service since several years. During this conference we got some further insight in the state of the art achieved in case of uncooled variable turbines. In my mind, however, it would be very worthwhile to hear more on the state of the art achieved with variable turbines for higher temperature that means with cooled inlet guide vanes.

The correct function of the control system under all flight and engine running conditions will be at least as important as the reliable function of variable components themselves. From the papers presented I got the impression that the technology being in hand with digital and analogous types of control systems will in principle meet the requirements given by the VCEs. But we should keep in mind that very probably the number of control variables, including the intake, to be integrated with the control system, will possibly raise to say 10 or 12 compared with now 7 or 8. I am not an expert in control systems but I could imagine that this increased number of control variables means a considerable increase in weight and cost, which has to be added to the excess weight and cost of the engine itself.

Once again I would like to refer to the problem of high development, production and maintenance cost to be expected with VCEs. When these costs are contrasted to those of current engines, it could be worthwhile to clear the question if by the introduction of VCEs, the number of different types of engines which are necessary to equip the required types of aircraft, could be reduced. If a higher degree of engine commonality could be achieved, this would result in some benefit in development and production and also maintenance costs.

Finally I would like to express that the aspects and results presented during this conference are rather encouraging. Certainly only a small part of efforts which will be necessary to create a VCE leading to a higher level of aircraft performance and effectiveness, has been made. In my opinion, however, it is necessary to continue the studies carried out so far in such a way that the work should be concentrated on the most promising concepts in order to make them really feasible. VCEs will certainly need intensive demonstrator programs. But my personal impression is that the time for starting a demonstrator program has not yet come, if the concepts presented here include all *existing* concepts. This holds only for military engines and not for the civil engine shown here in a movie.

Mr J.F.Chevalier: D'abord, je voudrais excuser Monsieur Wanner, Directeur Technique de l'ONERA, qui devait être présent à cette Table Ronde cet après-midi et qui n'a pas pu y venir. Il le regrette beaucoup parce que c'est un très vieux spécialiste des moteurs et il aurait certainement aimé discuter de ces questions.

Je vais donc essayer de donner quelques impressions que je peux avoir après ces quatre jours de session. Un premier point que m'est apparu est que l'on reconnait maintenant l'importance fondamentale des variations possibles de section de tuyères dans un moteur pour optimiser aussi bien le fonctionnement des composants que le choix du débit d'air du moteur pour l'adapter à la tuyère et à l'entrée d'air. Si je suis content de voir reconnaitre cette importance, je voudrais faire remarquer que de ce côté-ci de l'Atlantique ces problèmes là sont étudiés et les résultats des études sont utilisés depuis près de vingt ans dans les moteurs militaires. Dans un programme comme le programme Concorde je crois qu'on n'a pas ménagé les efforts pour adapter complètement l'entrée d'air et la tuyère de sortie à toutes les possibilités du moteur.

Sur un plan plus général, ce qui m'apparaît c'est que, après une période où l'on a essayé d'étudier des moteurs à cycle variable très ambitieux, par example capables de propulser un avion à Mach 3 en croisière, et d'être néanmoins très silencieux au décollage, l'orientation vers le domaine militaire nous conduit à des objectifs beaucoup moins ambitieux; d'ailleurs dans un des planches qui ont été présentées on montrait le compromis à faire entre l'amélioration de consommation spécifique et l'augmentation de poids du moteur et l'on voyait que l'on était beaucoup plus proche du succès pour une application militaire que pour une application civile à croisière supersonique. Cela veut probablement dire que l'objectif précédent était trop ambitieux.

Néanmoins, je me demande pourquoi on abandonne aussi vite l'objectif très ambitieux du moteur à cycle variable capable de faire une croisière supersonique et capable d'être silencieux au sol et au décollage. Je vois une raison à cet abondon, c'est une espèce d'espoir irraisonné dans les miracles de la technique acoustique. Ces temps derniers on a beaucoup espéré dans le système des "co-annular nozzles" qui effectivement, si il tenait les promesses announcées, éviterait d'avoir à faire des vrais moteurs à cycle variable. Or, je crois personnellement, et les analyses que nous avons faites de ce côté de l'Atlantique le montrent, que les gains annoncés sont très loin d'être même envisageables.

Je souhaite donc en conclusion qu'il se trouve des gens suffisamment imaginatifs et ambitieux pour relancer l'étude de véritables moteurs à cycle variable et ne pas rester sur une sorte d'optimisation des moteurs à double flux à post-combustion pour le domaine militaire.

Je m'excuse de la brièveté de mon intervention. C'est sans doute Monsieur Wanner qui aurait eu les choses les plus intéressantes à dire. Malheureusement, je n'ai pas pus recueillir le fruit de ses réflexions.

Mr D.R.Higton: I intend to keep my comments very brief. I agree with much that has already been said by the previous speakers and instead of reiterating many of the points already made I should like to just add a few comments and to raise a few points for subsequent discussion.

We have been presented with a very wide range of papers over the last few days. On the one hand we have considered papers advocating very advanced multi-mode variable geometry concepts and on the other papers dealing with fairly modest extensions of current technology to incorporate elements of variable geometry. There was surprisingly little in what might be called the middle ground, where the compromise between radical concepts and practical engineering constraints might ultimately lead us. I think Mr Payzer brought us closest to an appraisal of this compromise.

Mr Swan has just shown us a very impressive film of a dual-mode powerplant operating on a test bed, and this will obviously encourage us to seriously consider just what the prospects are for such devices. My feeling is that such radical schemes will not find ready acceptance unless the potential benefits remain very large after an extremely thorough and realistic assessment of the propulsion system as a whole, including the intake and nozzle, in relationship to the aircraft and its mission. We must remember that we are talking here of systems which involve very high technical risks and very high costs. I have had some previous experience with a dual-mode powerplant design study which foundered as a result of the engineering complexity involved, and this makes me somewhat cautious about the prospects for some of these current schemes. However, one undoubted benefit from this kind of activity, and I was reminded of this on a number of occasions during the week, is that it can offer new insights into how best to design powerplants which only incorporate a limited variable cycle capability.

Regarding variable turbines, I agree with Herr Grieb's comments concerning the difficulties of incorporating variability on HP turbine stages which at the same time require the use of very sophisticated cooling techniques. I would be interested in any comments on the prospects for variable HP turbines which the representatives of the engine industry might like to offer during the discussion that follows. However, I am left with the impression that variable turbines downstream of the HP system warrant further serious consideration for many potential applications.

One matter which has not been discussed during the week is the prospect for achieving large pressure ratio variations in multi-stage compressors using variable stators, or even limited application of variable rotor blading. I wonder whether any of the component specialists would care to comment on this later.

We have heard something of the case for very high pressure ratio cycles for subsonic transport, and I would agree that there is a strong case for moving in this direction. However, there are problems associated with high pressure ratios as we all know; the high temperature of the blade cooling air, and difficulties in maintaining efficiency in the later stages of the compression system due to aspect ratio and tip clearance effects. One further consideration which came up this morning was the impact of high pressure ratio cycles upon NO_x emissions and I wonder whether, even with variable combustion systems, we may not be forced to accept lower pressure ratios than we should otherwise prefer.

Another feature of the week has been the battle between the advocates of the short-cowl variable-pitch fan and those of the highly loaded propeller or prop-fan. I hope that during the discussion somebody will be prepared to comment on whether the high hub and gear-box weights which will be associated with the prop-fans might not exceed the drive system and cowl weights of the VP fan.

I feel that this conference has provided a stimulating forum for debating the potential benefits offered by variable cycle engines, and for reviewing the progress in variable geometry component technology which will ultimately determine the extent to which these benefits will be achievable in practice. The pace at which we can, or will, move towards variable cycle engines is clearly a matter which will continue to be debated vigorously, but I think that there can be little doubt that more extensive variability than is currently exploited will be a feature of future powerplants for particularly demanding applications.

Mr M.Stoll: J'aimerais commenter les exposés que nous avons entendus du point de vue de l'avionneur et de ses problèmes. Pour simplifier, je classerai les avions en trois catégories: les avions civils supersoniques, les avions civils subsoniques, les avions militaires.

Le SST semble être un domaine priviligié d'application de la géométrie variable des réacteurs, compte tenu des compromis à réaliser entre les besoins subsoniques, supersoniques et d'environnement. Mais n'ayant pas d'expérience dans ce domaine, je ne ferai pas d'autres commentaires sur ce point.

En ce qui concerne les avions civils subsoniques, on nous promet des propulseurs ayant toutes les qualités grâce à la soufflante à pas variable, carénée ou non. Les consommations seront diminuées de 30%. Les problèmes de 'réverse', de prélèvement ou de dégivrage en descente, de temps de réponse, seront tous élégamment résolus. Ceci nécessite bien un réducteur, mais on nous montre que c'est plutôt un avantage puisque celcui-ci permet de décaler la soufflante par rapport au corps haute pression pour placer l'un devant l'aile et l'autre en dessous, d'où une solution au problème d'installation de ce propulseur encombrant. Comme ceci permet de plus un excellent soufflage de la voilure, y compris à l'atterrissage: on réduit ainsi les longueurs de pistes. De plus, comme le bruit est insignifiant, du moins pour les riverains, on a résolu tous les problèmes des STOLs, du moins pour les avions d'affaires car je n'ai pas beaucoup entendu parler de gros moteurs de ce type. Toutefois, quand Mr Gray annonce un gain de DOC, j'aimerais être sûr que le coefficient par lequel il a certainement multiplié le coût d'entretien des réacteurs par rapport aux réacteurs actuels est raisonnable, pour tenir compte de l'entretien de quelques engrenages. Et le prix à payer comprend peut-être aussi une diminution de fiabilité et une augmentation de masse. De plus, il reste peut-être encore quelques problèmes d'intégration et je me demande si l'on a fait des essais à Mach 0,8. Je ne veux pas dire par ces remarques que les solutions proposées sont à rejeter, bien au contraire, je crois qu'elles méritent d'être validées par des essais et surtout poursuivies par des équipes mixtes motoristes/avionneurs compte tenu de l'importance de la propulsion et de la sustentation sur la performance globable.

En ce qui concerne les avions militaires, peut-être offrent-ils un domaine d'application plus large à la géometrie variable des réacteurs compte tenu des problèmes à résoudre. Le premier problème est de fournir la performance souhaitée dans un domaine d'emploi très vaste et, sauf cas particuliers d'avions très spécialisés – même si l'avion considéré n'est pas très polyvalent – les exigences d'un programme donné sont souvent presque inconciliables avec un cycle unique. Ces exigences en performances sont de deux types: d'une part, en un point du domaine de vol, on demande au moteur une forte poussée pour manoeuvrer et une faible consommation en croisière ou attente, donc à faible poussée, et la tendance est d'écarter ces deux points de fonctionnement l'un de l'autre car les avions deviennent plus fins et les exigences de manoeuvrabilité vont croissant. D'autre part, le domaine de vol couvre souvent des vitesses très faibles et supersoniques, de basses et hautes altitudes, de faibles et fortes pressions dynamiques.

Mr Ripoll sur une des ses planches a bien montré que, pour un cycle donné, on arrive rapidement sur une butée quelle que soit la direction dans laquelle on cherche à élargir le domaine de fonctionnement. D'ailleurs ces problèmes de performances conduisent déjà à une géometrie variable pour certains réacteurs actuels, que ce soit pour l'entrée d'air ou la tuyère, ce qui si j'ai bien compris Mr Swan est d'ailleurs périmé, et à des cycles variable, soit par stator variable ou décharge de compresseur, soit par emploi de post-combustion; on sera certainement conduit à aller plus loin, par exemple par l'emploi de distributeurs de turbines variables pour répondre à l'extension du domaine de poussée ou des variations importantes de cycles suivant le mode de vol. Ici encore, et Mr May l'a bien montré ce matin, l'installation joue un rôle essentiel et l'intégration de la propulsion à la cellule nécessite un travail très concerté entre motoristes et avionneurs. L'échange d'informations devant d'ailleurs se poursuivre en vol pour le plus grand bien de la régulation de l'ensemble propulsif. Mais ce ne sont pas les seuls problèmes à résoudre, et la géométrie variable aidée par une régulation savante devrait aider à améliorer les temps de réponse en poussée face à des sollicitations brutales, par exemple au combat ou en approche. Egalement, elle devrait rendre le réacteur plus tolérant face à une alimentation en air perturbée par de très grandes incidences ou le tir d'armes diverses – et bien sûr l'avionneur serait très satisfait si la géométrie variable permettait de diminuer la masse et l'encombrement du réacteur. Mais peut-être faut-il là laisser leur chance à d'autres innovation techniques.

Je terminerai par une dernière exigence: ces différents progrès dans les performances ne peuvent pas être obtenus au prix d'une diminuation de la fiabilité et de la sécurité et – je rejoints là certainement Mr Higton – il faudra accumuler beaucoup d'expériences, d'essais, avant d'appliquer effectivement les différentes techniques qu'on nous a présentées.

Mr A.H.Jackson: I'll be talking first on the overall weight of the system. The effect of cutting the diameter in half of a conventional propeller which of course is accomplished by going to the very high loading with the prop-fan, has a very substantial impact on the rotor weight and the gear box weight, both. The rotor weight is cut about in half on a conventional propeller producing the same thrust, and the gear box weight also is reduced almost to a similar amount due to the overall torque. When you compare it to the VP fan, Dave Gray from Pratt & Whitney went into this in considerable detail as did the General Electric Company in their studies for NASA, on their fuel conservative engine work. I'm not sure exactly how these comparisons came out, it does not show up very clearly in their final report either. But just looking at that particular pressure ratio fan, which was like in the neighborhood of 1.1 pressure ratio, the diameter is fairly high, and by the time you get a short shroud, then you got a long beam supporting it along the vanes – I think that the weight trade-off would not be particularly in favor of that kind of a fan over the prop-fan.

Now going to the higher pressure ratio fan of 1.25 and on the assumption that you could make that a comparable performance as Ralph Denning suggests, I would suggest it would be lighter, but there the argument has to boil down to the relative performance. I don't think the difference is going to be all that critical to the aircraft designer, I don't think the argument is basically falling into that area.

Mr N.F.Rekos: I have handed in a question to the experts and would like to repeat it here. Will variable goemetry engines demand a new type of ground facility? I think we talked about very high pressure ratios in the order of 45 to 1 and very high temperatures in the order of 3000°F. Should we be prepared to face this problem which will include the problem of funding as work on VCE will be very active in the future?

Dr H.Grieb: I have a question for you Mr Rekos, concerning high temperatures and high pressure ratios. Why do you tie up both the parameters with variable cycle engines. I mean, if higher temperatures will come because they are feasible from the standpoint of technology and if high overall pressures make sense from the thermodynamic cycle point of view, they will come, either VCEs will come or not.

Mr N.F.Rekos: I think it was indicated that high temperatures and high pressures provide the military engines with the high performance that they must have. How do you accommodate high pressure ratio like 45 to 1? We know you have problems there with compressor tip clearances. You'll find that's a requirement, and you'll have to look for it. But let's go back to the environmental part. If we go to these high pressures, they are not consistent with keeping the nitrogen oxide down. But nevertheless, we have to take a look at these facilities that are required there. We have very few facilities, for example, that can test a combustor at 40 or 45 to 1 pressure ratio within a reasonable size. We generally have to wait until we get an engine and test it as such, which becomes very expensive. So, I would indicate that you do need high pressure facilities for combustor testing in order to be prepared for some of this work. That was just one suggestion I have.

Now to answer your first question, I do believe that high performance is going to require high temperatures and high pressures. Obviously we can't go back to something like the JT3D or the JT8D very mild pressures. You're not going to be No.1 in the field of military airplanes if you resort to that level of technology.

Mr C.Ripoll: Je crois qu'il faut considérer trois points de vue différents: Si vous parlez de moyens d'essais en pensant instrumentation, il peut y avoir des nouveaux problèmes, je pense par exemple à des sondages fins derrière l'entrée des turbines qui est variable. J'imagine assez mal comment on peut analyser finement l'écoulement alors que tout est en train de varier dans toutes les directions. C'est la même chose certainement avec toutes les chambres de combustion variables. Il y a là un champs de développement pour des méthodes d'investigations.

Le deuxième point sur hautes pressions/hautes températures est parfaitement exact mais n'est pas nécessairement lié à la géométrie variable. Nous sommes déjà de toute façon sur le chemin des hautes pressions/hautes températures, et nous savons qu'il faut faire des essais extrêmement réalistes.

Un troisième point est je crois plus important, c'est celui de l'intégration. On a insisté beaucoup sur les gains globaux de performance qui sont obtenus grâce à une meilleure intégration du moteur et de l'avion. Si l'on considère l'avenir avec par exemple des avions construits en nouveaux matériaux qui pourront avoir des formes différentes, des moteurs qui pourront avoir des tailles et des allongements différents, nous pouvons être amenés à demander des installations qui permettent de faire des essais d'un moteur complètement intégré à un avion et là il y a aussi une question de taille des installations.

Mr M.Stoll: En tant qu'avionneur, je m'associe à cette dernière demande.

Mr N.F.Rekos: I had one thought on that subject: take Mr Swan's type of engine where we have a valve arrangement, where we get rapid changes from by-pass ratio of 1.1 to 3.9 where we get the rapid changes in nozzles also. Our present facilities are not capable of handling the altitude changes I am referring to. He resorted to an outdoor test and which might be o.k. for a sea level test. Some of the earlier studies that Willis and Boxer had presented here require operation of these variable geometry inlets and nozzles at altitude conditions. I don't believe we have facilities that will be able to test them.

Mr W.C.Swan: I'd like to change the subject. I'd like to provoke some arguments on the subject of conservatism versus being what would be called a radical, which is what I am. Let me quote some well known history: I remember that in 1957 we, at Boeing, were looking for an engine, other than a turbojet, and we went in the engine industry in Europe and in America and the Rolls-Royce Company in Derby was the only one brave enough to propose an engine. They were so brave that they proposed the 2/10 by-pass ratio engine. Our calculations suggested it ought to be at least one, and possibly two, so far, the radical group I represent, and Rolls-Royce's reasons for not going above 2/10 by pass ratio was that the discs would fall apart. They thought you could not hold an engine together at about 2/10th by pass-ratio. So they built the Conway and they sold a few of them. We finally broke back Pratt & Whitney in half with the JT3 and told them they could make a fan out of it. They came all the way up to 1.3 and they walked away with the entire business. General Electric who did not even have an engine at the time were dying to do something because they had nothing. They worked like hell to come up with an aft-fan engine and they would make a by-pass 3.4 or 5, anything I wanted. But at that more point in time, we weren't too sure of the aft fan engine programme and the guy in the middle won the business. But not by his own doing. I remember promoting the so-called high by-pass ratio engine back around 1965, long before a thing known as project forecasting group in the US existed. Finally that group of people got interested and supported the idea. I remember personally going to Pratt & Whitney with a high by-pass fan. They told me you had a pretty gear box in between the turbine and the fan and it could never happen. They are selling probably their 5000 JT/9D to-day without a gear box. So I think we are seeing in this room the conservatism of the designer and the radicalism of the buyer, and the buyer always wins.

Mr A.J.B.Jackson: I'am sorry about Mr Swan's troubles with Pratt & Whitney. I'd also like to comment at once to some of his earlier remarks about nozzles. He ought to, perhaps, buy his engines from companies which supply the whole power plant. There's quite a good company outside the United which does that.

I would however like to say that I am very much in favour of innovation and looking to the future. I'm not wishing to make an enemy of Mr Swan in any way. I would also like to say in support of this that we have offered one or two things in the past which have caught on such as sweep wings, and vertical take-off fighters so we have a history for being innovative. I hope that we can continue that way.

Mr R.M.Denning: First of all, it's nice to know that the aircraft companies are within the forefront of everything. I only wish that they would put their wings a little bit farther from the ground, so that when somebody wants to sell a high by-pass engine you can get it under the wings. Anyway that is my way of being humourous.

The other point I wanted to make was going back to the subject of higher turbine temperatures. Unfortunately I missed Mr Gray's lecture. It seems to me that the talk about the 45 to 1 pressure ratio raises a few questions about the cooling air temperature in the HP turbine blades. If you do your parametric studies with a constant blade surface temperature you will find that the cooling technology of the blade must be increased as you go up in pressure ratio. Now you could also cool the cooling air. But the contemplation of an air to air heat exchange, taking something like 5% of the core flow, cooling it perhaps in the by-pass and then bringing it back into the engine and in the rotor blade is a rather appalling thing.

Now if you take a constant technology of blade cooling, a very simplified assumption, you would then do a parametric study with a dropping turbine temperature as you increase the overall pressure ratio. If you do those sorts of calculations you can't come up with any justification for 45 to 1 pressure ratio. In fact it flattens out with the current level of turbine efficiency at a pressure ratio of about 30. If you assume a 90% efficient turbine at all time, it flattens out at about less than 40.

If I might be allowed a third point. It seems to me that variable cycle engines serve two rather distinct purposes – one is to improve the SFC in different parts of the flight plan. Now with variable geometry, due to the possibility of leaks and gaps and unfavourable geometrical situations at off-design you might lose that SFC advantage if you're not very careful. The use for variable cycles is to make the aeroplane do something it could not possibly do without it. One can think about a number of systems: the convertible rotor plane where you are making the engine to do two quite distinct tasks, and may be even on the SST where you are getting the noise down which otherwise would be impossible to do. If these co-annular nozzles results are provided with forward speed, this will alter the whole dialogue between the engine and the aircraft manufacturer and the noise authorities and will allow us to choose quite different cycles for SST engines.

Mr N.F.Rekos: I think what these studies do for us is to focus our attention. Mr Denning mentioned technology levels significantly above where we are. He is right, the technology level is to be much higher. We started out thinking of compressors with pressures up to 20 to 1 in six stages, but it requires some imagination, some thought, some stimulus. We feel that the studies were conducted with great sincerity, realism and we recognize the need for improving our technology in the compressor, in the combustor and in the turbine. So we are starting experimental research for it.

To avoid confusion between development which Mr Swan is primarily interested in as he wants to fly on the other side research from our point of view. We want to be able to provide this so-called technology but it is required of us to see that we are going along the right path. So we feel we are going along the right path, with the very high by-pass ratio turbo fan: by-pass ratio 8 or 9 to 1, without gears but requiring relatively high turbine inlet temperatures and a very high pressure ratio.

In order to obtain those benefits one must put attention in your particular research. And one of the intents, one of the AGARD meanings is to stimulate research in the AGARD community. We are not in the position to direct or dictate anything. So what we are trying to do is to stimulate some thought in this area and get your comments.

Dr J.Dunham: Despite the availability of the well established technology of variable geometry inlets, both the B-1 and F-16 aircraft have fixed geometry, owing to cost. Does Mr Swan consider the decisions wrong or does he see advances in technology reversing the balance of the economic considerations involved?

Mr W.C.Swan: I refuse to give you a direct answer, but I will discuss the subject. I do believe one of the higher maintenance items during the few months it has been in service has been the intake and the control system of Concorde. On the other hand, a much lower cost item, when looking at the variable geometry features of the gas producer itself. I look upon the inlet as a high cost item, and it is only there on the airplane because the engine manufacturer has never invented the supersonic axial flow compressor. Dr Antonio Ferri tried to do it some 25 years ago, at NASA Langley, but he found few disciples to follow him up. So we had to put this intake on to Concorde to accommodate the engine.

I believe that when such decisions as a fixed inlet are made, then planning should suggest totally different air flow characteristics for the engine.

We are smart enough in this room to do things with the engine that essentially make the engine a constant corrected flow machine.

Dr J.Dunham: Perhaps I could just comment having had a little to do with variable inlets and also with turbomachines. Yes, I agree that the variable intake and the control system are high cost items. But looking at other variable geometry bits inside the engine, I think they are probably more complex than the variable inlets and they have got another control problem. They are comparable surely.

Mr J.Hourmouziadis: I have a question for Mr Swan. He showed us a very interesting film on a demonstrator engine with a variable by-pass ratio. The demonstrator did cost about 150,000 dollars US. He did not tell us why he stopped the programme after having spent the 150,000 dollars.

Mr W.C.Swan: I didn't. I just told you what it cost to answer two questions: (1) Is it possible to put something in between a fan and a compressor that plays tricks with the flow without having the engine go to surge, and the answer was no, there was no problem. (2) It answered the question: "When you change from one by-pass ratio to another in an extremely short time or a very long time, does it make any difference?" The answer was no. That's what we paid $150,000 dollars for. It was worth it.

Mr N.F.Rekos: I think the reason was to show how low and small resources might be to try another innovative concept.

Dr H.Grieb: Mr Swan, if you would transmit the concept shown in your movie to a military engine for an aircraft with supersonic intake and certainly then with two or three stage fans in series and in parallel, would you consider the concept you have shown to be feasible for this type of engine?

Mr W.C.Swan: Military or civil, who cares. The answer is yes. This concept would, in my opinion, only make sense on a long-range airplane with multimission requirements, not a short-range airplane at any speed.

Mr J.F.Chevalier: Mr Rekos dans son introduction a beaucoup parlé d'économie de carburant et finalement, ça n'a pas l'air d'attirer grand monde c'est-à-dire que l'on est pas revenu sur la nécessité d'économiser le carburant. Personnellement je ne vois pas très bien dans le domaine militaire cette nécessité, mais je pense qu'il n'y a pas eu beaucoup de commentaires sur ce sujet.

Mr N.F.Rekos: I agree that there have not been many comments on that particular aspect. Since fuel apparently is plentiful right now, no one bothers with it. Many of us will wait until the problem becomes aggravated. When you are denied access to fuel you ll find you have a problem, but then it will be too late, you will not be prepared and you will not fly then.

Again I wish to thank the Panel Members of the Round Table for their time and effort in preparing the comments. Thank you.

CLOSING REMARKS OF THE PROGRAM COMMITTEE CHAIRMAN

Mr N.F.Rekos: We come to the close of what I believe has been a most productive, if not provocative and challenging meeting. The very fine papers that were presented and the subsequent discussions wich culminated in our Round Table Discussions, when put together in the final proceedings should result in a noteworthy and memorable document – a tribute to AGARD. It could very well be that by our efforts, we have laid out the ground work for the design of future aircraft, which I personally feel will be influenced by variable cycle engines for both military and civil applications.

I wish first to thank my colleagues on the Program Committee for their assistance in putting together the program for this 48th Propulsion and Energetics Panel Meeting. I wish to thank all the authors for all the time and effort that I know was required in order to produce the fine paper they presented.

Special thanks to the French National Delegates for inviting us to hold our meeting in these fine accommodations and providing the most welcome reception Monday evening; also Messrs Ripoll and Fabri for their assistance in making our stay here most pleasant.

Thanks to Mr Krengel, the Propulsion and Energetics Panel Executive, Mrs Scopes and the AGARD Staff for assisting us with our administrative requirements.

Last but not least, I would like to thank the Projectionist, Mr Leport, the electronic technicians, and the interpreters for their excellent and rapid translations.

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14. Abstract

These Proceedings consist of 33 papers including the discussions after each paper, and the Round Table Discussion at the end of the Symposium in Paris, 6–9 September 1976 (48th Meeting of the Propulsion and Energetics Panel). The papers from 6 nations (CA 1, IT 3, FR5, GE 5, UK 8, US 11) were grouped into four Sessions: Military Aspects of Variable Geometry Engines, Application of Variable Geometry Engines for Future Aircraft, Variable Geometry (Fans, Compressors, and Propellers). Variable Geometry (Combustors and Turbines). The session related to components included two papers on integrated power plant control systems.

Both military and civil aspects of variable geometry engines were analyzed, a broad variety of cycle investigations presented as well as on geometry variations of the engine components. Discussions of individual presentations and the final Round Table Discussion placed concern on weight and complexity problems of the engine under consideration and tried to identify what risks and costs are likely to be involved as tribute to an advanced performance during the different parts of flight missions.

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